Proposal for Two-part Recoverable Launch System for Unmanned Payloads to LEO and GTO

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I. Nomenclature

KSC = Kennedy Space Center, NASA Launch site in Cape Canaveral, FL

ΔV = Delta-V, measure of potential change in rocket velocity (m/s)

Isp = Specific Impulse, a rocket efficiency term (s)

LEO - Low Earth Orbit, ~ 500 km

GTO - Geostationary Transfer Orbit, Apoapsis ~ 42000 km

C_L - Coefficient of Lift

L/D - The ratio between the Lift produced by an aircraft to the drag encountered by it

II. Summary

Team 16 has created a viable system for a fully reusable launch system. This system consists of a 2-stage launch vehicle for the LEO and GTO payloads, and and a recovery aircraft. The launch vehicle, hereafter referred to as **Horus**, consists of two liquid fueled stages, capable of taking a 15,000 kg payload to LEO and a 5000 kg payload to GTO simultaneously. The first stage conducts a propulsive-landing, and the second stage goes through ballistic re-entry. The aircraft, hereafter referred to as the **Falcon**, launches to recover the payload fairings and second stage upon its re-entry, capturing and stowing these before landing.

III. Introduction

The RFP issued by NASA stated called for a payload delivery system capable of simultaneously delivering payloads to LEO and GTO in a single launch. This system is to consist of two vehicles, a fully reusable rocket, and a recovery aircraft for any upper stages.

A. ROCKET DESIGN REQUIREMENTS

The primary requirement for the rocket is complete reusability - the rocket may have no disposable parts - all fairings, parachutes, or other deployed elements must be recoverable. Furthermore, the rocket must be able to deliver its payloads from launch sites at NASA's Kennedy Space Center and the USAF's Vandenburg AF Base to a 500km posigrade LEO and a standard GTO, at various inclinations.

The first stage (and first stage only) is permitted to perform a propulsive landing, while the second stage must be recoverable after reentry at high altitudes - soft landing on terrain or water is not allowed, and any recovered pieces must be recovered before they fall into commercial air lanes (considered here to be 40,000 ft maximum altitude). NASA's safety margin is 10,000 ft, so the minimum recovery altitude is 50,000 ft. All propellants must be considered "safe" for handling and use by NASA, with Isp values provided for engines used. In addition, inert mass fractions must conform to historical parameters for previous rockets.

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B. AIRCRAFT DESIGN REQUIREMENTS

NASA's main requirement for the aircraft is mission capability - the aircraft must be able to recover all staged parts of the rocket before landing. This means that the aircraft must take off from the launch location, transit downrange to recovery locations, and recover all staged parts (excluding the first stage). It must have the loiter time to stay on station while the upper stages round the earth and deorbit, and then must perform the capture at the lowest possible speed. In addition, once the aircraft has recovered all elements of the rocket, it must then be able to land on existing runways at KSC and Vandenberg AF Base.

IV. LAUNCH VEHICLE DESIGN

A. DESIGN PROCESS

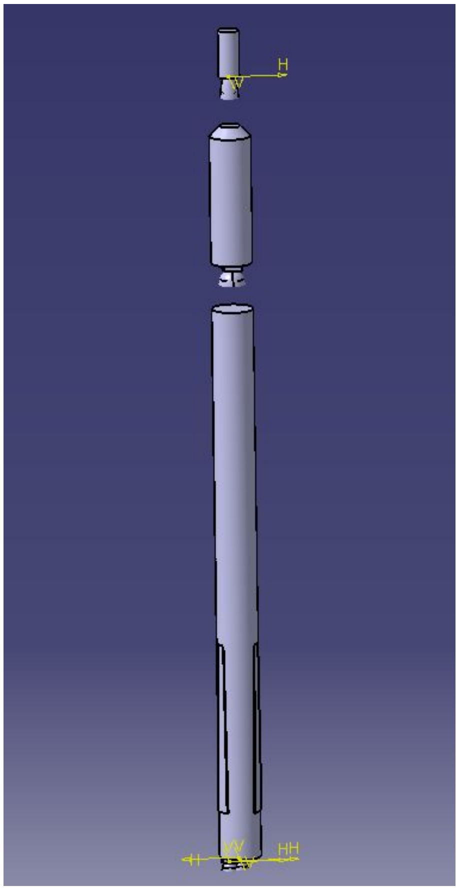
Our group modeled our launch system, Horus, using existing designs for commercial launch vehicles, to give us a starting idea of how modern designs achieve the necessary ΔV. As discussed in class, ΔV for spacecraft is a function of their engine efficiency, or Isp, and the vehicle's starting and final mass. These parameters affect their engine configurations, fuel loads, and staging parameters for spacecraft, based upon their mission requirements. It was found that a traditional multi series-stage system best fits our design, mainly due to the design constraints that require any additional stages to be recovered. This motivated us to pursue design simplicity with as few stages as possible, which has benefits for recovery as well as reusability - the fewer systems in the vehicle, the less inspection, processing, and refurbishing is necessary to re-use the vehicle.

For inspiration, we surveyed the current market for launch vehicles capable of carrying our payload, in order to have a baseline for our design. By surveying the current commercial market for launch vehicles, we identified several intriguing designs, mainly due to their payload weight to orbit. These were the United Launch Alliance's Delta-IV(M)[1] and Atlas V(531)[2], and Space X's Falcon 9 Full Thrust[3] (henceforth "Falcon 9"). We observed that these vehicles favor a minimalistic 2-stage design, proving to us that two-stage to orbit designs exist for our payload class. Upon further deliberation, it was decided to directly model our first stage off of the Falcon 9, as it is the only launch system to have ever practically demonstrated first stage recovery. While the Falcon 9 itself is not a fully recoverable vehicle, it provided a suitable base to work off of. To meet the launch requirements in this RFP, we had to essentially scale up the Falcon 9 to create the Horus. The Falcon 9 would not meet this RFP otherwise, since according to SpaceX[3], the Falcon 9 is capable of delivering 22,800kg to LEO and 8,300kg to GTO, but it cannot deliver two payloads to two different orbits, which is required by the RFP. The Horus had to achieve the RFP's requirements for payload margin while adding additional equipment for the recovery mission.

B. INTRO TO ROCKET DESIGN AND RATIONALE

Despite these changes, the Horus's design has a very similar outward appearance to traditional two-stage, liquid fueled rockets. The Horus has a traditional series-style "stacked" arrangement, with the payloads and stages stacked vertically on top of one another. The payloads are contained in an aerodynamic fairing at the top of the stack, with 1086 m^3 of internal volume for the LEO and GTO payloads, which we considered sufficient, as it provides 90% of the Space Shuttle's payload capacity{c} in a cheaper and more easily reusable system.

The Horus is capable of launching from both KSC and Vandenberg, as detailed in section C. It follows an optimal ascent trajectory, staging once the first stage depletes its fuel save for a reserve it uses for landing. The first staging altitude is at around 60 km/ 196,850 ft. After staging, first stage detaches and executes a propulsive landing on a droneship downrange, while the second stage fires and continues to LEO. We built in a sufficient amount of delta-v into the fuel budget for the first stage to account for this maneuver, based on the Falcon 9's landing. From there, the second stage deploys the 15000 kg payload as per mission requirements, and releases a second payload, the smaller booster with a 5000 kg payload. This smaller booster is contained under the fairing and strapped to the stack until LEO, from which it commences an injection burn to GTO.



C. ΔV REQUIREMENTS

With the spacecraft's mission planned and orbital profile planned, sizing the rocket began with accurate calculations of the ΔV requirements for all maneuvers. The first major maneuver is the ascent to an altitude of 500 km in LEO, which can be calculated using the equation for velocity in a circular orbit, henceforth Equation 1:

$$v_o \approx \sqrt{\frac{GM}{r}}$$

Since r = 6878km and GM = 3.986 * 10 ^5 km^3/s^2, the orbital velocity of a body at this altitude is 7.61267 km/s. Factoring in the assumed losses, the ΔV requirement to obtain a 500 km low earth orbit is 9.96267 km/s. When launching from Cape Canaveral and into an equatorial orbit, this requirement decreases to 9.55 km/s due to Earth's rotation. From LEO, the ΔV requirement to GTO can be calculated using the vis-viva equation, henceforth Equation 2:

$$v = \sqrt{\mu\left(\frac{2}{r} - \frac{1}{a}\right)}$$

Since $\boldsymbol{\mu}$ = GM and is the semi-major axis of the orbit, the velocity in GTO at that altitude will be 9.981 km/s. The ΔV required to enter GTO is the difference between GTO velocity and and the circular orbital velocity, which is 2.369 km/s. The total ΔV the launch vehicle must impart on the payloads, therefore, is 12.33 km/s. Other maneuvers, such as propulsive landing, cannot be calculated using orbital dynamics.

D. ROCKET SIZING

Since the ΔV requirements of the major maneuvers are known, the process of calculating the mass of each stages begins. The first step towards calculating stage mass is the selection of an engine, which provides an Isp value, which is used by Tsiolkovsky's Rocket Equation, henceforth Equation 3:

$$\Delta v = v_e \ln \frac{m_0}{m_f}$$

Ve = Isp * g0. From this equation you can find the mass ratio Mo/Mf, which is can be related to a series of other ratios to give estimates of propellant needed, the inert mass of the stage, and the initial mass. These ratios are described in these series of equations, where r = Mo/Mf:

- Basic mass summary

$$m_o = m_{pl} + m_{pr} + m_{in}$$

- Inert mass fraction

$$\delta \equiv \frac{m_{in}}{m_o} = \frac{m_{in}}{m_{pl} + m_{pr} + m_{in}}$$

- Payload fraction

$$\lambda \equiv \frac{m_{pl}}{m_o} = \frac{m_{pl}}{m_{pl} + m_{pr} + m_{in}}$$

- Parametric mass ratio

$$r = \lambda + \delta$$

m_o =initial mass
m_{pl} =payload mass
m_{pr} =propellant mass
m_{in} =inert mass

These ratios can then be related back to the mass values to give initial estimates of propellant mass, inert mass, and initial mass:

$$M_o = \frac{M_\ell}{\lambda} =$$

$$M_i = \delta M_o$$

$$M_p = M_o(1 - r)$$

An important constant to find is the inert mass fraction $\boldsymbol{\delta}$, which, based off of historical regression data, can be assumed to be .075 for Liquid Oxygen (LOX) /Liquid Hydrogen (LH2) stages, and .063 for Liquid Oxygen/RP-1 powered stages.[19]

At this point, the respective amounts of Oxidizer and Propellant can be found, based on the respective fuel ratio of that engine. Once this amount is calculated, the mass and geometry of the fuel tanks can be calculated using the density of the respective fuel:

- LH_2 tanks

$$\rho_{LH_2} = 71 \ \frac{kg}{m^3} \Longrightarrow M_{LH_2 \ Tank}\langle kg \rangle = 0.128 M_{LH_2}\langle kg \rangle$$

- LOX tanks

$$\rho_{LOX} = 1140 \frac{kg}{m^3} \Longrightarrow M_{LOX \ Tank}\langle kg \rangle = 0.0107 M_{LOX}\langle kg \rangle$$

- RP-1 tanks

$$\rho_{RP1} = 820 \frac{kg}{m^3} \Longrightarrow M_{RP1 \ Tank}\langle kg \rangle = 0.0148 M_{RP1}\langle kg \rangle$$

Once the dimensions of the fuel tanks are known, the stage can be sized. Additionally, both LOX and LH2 are cryogens and require insulation, which can be calculated via these equations:

$$M_{LH_2\ Insulation}\langle kg\rangle = 2.88A_{tank}\langle\frac{kg}{m^2}\rangle$$

$$M_{LOX\ Insulation}\langle kg\rangle = 1.123A_{tank}\langle\frac{kg}{m^2}\rangle$$

Finally, an intertank and occasionally payload fairing will be added to the stage, completing the sizing of the stage.

$$M_{fairing}\langle kg\rangle = 4.95\left(A_{fairing}\langle m^2\rangle\right)^{1.15}$$

At this point, the initial estimates are almost surely off, which can cause the stage not to produce the ΔV initially calculated. A second pass through the propellant and tank masses should meet the initial ΔV target.[18]

The first stage to be sized is the GTO Booster, which is also the smallest. Calculating its mass first is due to the fact that it will act as payload to the other two stages on ascent. Additionally, it has the smallest ΔV requirement, just 2.369 km/s. Furthermore, it was the simplest of the three major stages, as it did not require any extraneous components. The HM-7B, a LOX/LH2 engine was chosen for this stage because of the high efficiency and low mass, with an Isp of 446. Because the stage was small, there was not significant penalties for choosing a LOX/LH2 stage.

The second stage is to impart 5.535 km/s onto the stage, after the first stage cuts off. It is powered by a single Merlin 1D Vacuum engine, which is a RP1/LOX engine. This allows for a lower Isp of 348, but does not require as much cryogenic insulation as a LOX/LH2 engine.

The first stage is to impart 4.85 km/s onto the stack, which propels the vehicle out of the atmosphere. It uses 11 Merlin 1D engines, with an Isp of 282.

C. UPPER STAGE DE-ORBITING AND RECOVERY

The second stage, once free of its payloads, inflates a heat shield situated behind the payload stack and conducts a deorbiting burn to situate it on an initial 1.5 degree entry angle with the atmospheric interface. The entry angle was determined by referencing the ballistic entry angles for the Mercury program's orbital flight with John Glenn's flight in Friendship 7. In addition, we are assuming a targeted descent profile that places the deorbiting stage over the ocean (in case of system failure) and, if applicable, on the way back to base for the aircraft. This way, the minimum fuel is required for the aircraft as it returns from its downrange leg to recover the fairings.

In calculating when and where it would be possible to recover the second stage, we utilized equation 8.87 from Anderson's Introduction to Flight, 8th Edition (the course text) to model the spacecraft's velocity profile during reentry, with the intent of finding the altitude at which we could deploy our recovery parachutes. Based on some initial calculations for drag force, we assume that as long as the second stage is subsonic, we will be able to deploy chutes. Since we are not designing a heat-shield for the spacecraft, we took the liberty to assume some of its characteristics to do these calculations - based on NASA's own inflatable heat shield [19], it is likely to have a cone shape with a coefficient of drag of 1.28 (data taken from *Fluid Dynamic Drag*, Dr. Sighard Hoerner, 1965) and a surface area in the realm of 28 m^2 . Once these values were in hand, we were able to include the equations in order to find the altitude at which recovery would be possible. According to our calculations, the aircraft reaches a subsonic speed by the time it has passed 40 km, or 137,000 ft, by which time the atmospheric density has reached the point that the spacecraft is subsonic. The spacecraft then deploys its parachutes once decelerated. For weight calculations, of the second stage, these were closely modeled on the Orion MPCV's main parachutes for their proven technology and well known figures [20]. Once the chutes are fully deployed, the second stage awaits aerocapture.

D. FAIRING RECOVERY

The payload fairings will split apart soon after the first stage boosters cut off and stage separation occurs. The fairings will be equipped with small thrusters which reorient the separate shells into the atmosphere. As they undergo re-entry, the shells will passively slow themselves down to subsonic speeds because of the high levels of drag they will encounter. At a sufficiently slow speed, parachutes will deploy, and the fairings will await capture by the aircraft. This process is similar to the attempts at fairing recovery by SpaceX on their Falcon 9 launch vehicle.[5]

Interstage fairings will not be recovered by aircraft, but will remain attached to the first stage. During liftoff and the initial climb, the fairing will shroud the second stage. After the first stage cuts off, the second stage engine will slide out of the shroud and begin firing. The shroud will not be jettisoned, but will land on top of the first stage on the droneship. Some issues associated with this design are potentially adverse aerodynamic forces on the now-empty shroud on descent, and added complexity involved with stage separation.

V. RECOVERY AIRCRAFT DESIGN

A. Concept Selection and Sizing

The aircraft that is purposed to recover the second stage and fairings of our rocket must fly 10,000 feet above commercial traffic which makes its service altitude around 45,000 feet[6]. It should be able to loiter at this altitude for some time before it starts performing maneuvers to capture the fairings and second stage. We took several design inspirations from vehicles that featured high cargo capacity such as the Lockheed Martin C-5M Super Galaxy to blended wing vehicles that offered very high fuel efficiency and loiter times like the Northrop Grumman YB-35. We used an evaluation matrix that featured 9 different requirements which were weighted according to their rank which was determined by the mission profile. We compared Blended Wing body configurations, Traditional Body configurations, Fused Body configurations, Blended Body with Fuselage configuration and Blended Body with Canard configuration and found that the Traditional Body best suited our mission profile with the Blended Wing coming in at a close second as is evident in the design matrix presented below. The scoring system used is also provided below the evaluation matrix for better understanding of our matrix.

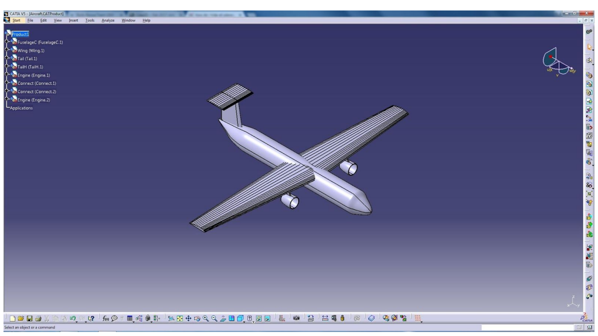
Aircraft Design	Weight	Blended Wing (YB 35)	Traditional Body (C-5)	Fused Body (Stratolaunch)	Blended Wing + Canard	Blended Wing + Fuselage
Safety Features	16	2	6	8	0	4
Capacity by Size	14	8	6	2	2	4
Loiter time	13	8	2	4	6	4
Fuel Required	12	8	4	2	6	2
High Speed Performance	11	8	4	4	6	4
Low Speed Performance	10	0	6	8	2	4
Landing Ease	9	2	8	6	0	4
Degree of Innovation	8	4	8	6	2	0
Length of Runway Required	7	6	8	4	0	2
Total Points	100	524	550	486	280	330

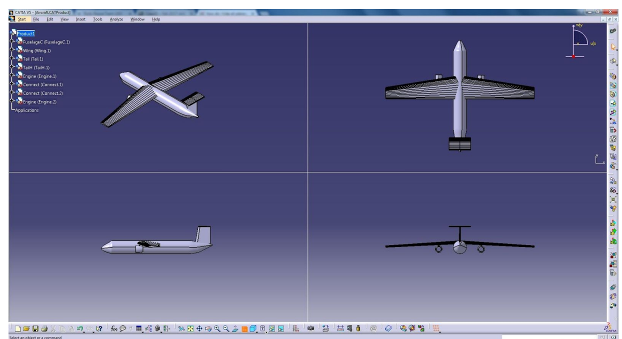
Scoring System	
Rating	System
Best	8
Better	6
Average	4
Poor	2
Worst	0

We found that the Traditional Body design which we wanted to base on the Lockheed Martin C-5M Super Galaxy to be the best for multiple reasons. First off it is a tried and perfected design which requires little to no major modifications which makes it the easiest to manufacture and design. It also comes in second place with the most amount of safety features present onboard in case something was to go wrong with the mission as it has pre planned emergency routes and doors along with flotation devices in case the aircraft was to splash down in water. It also has a large cargo door and several exits along the length of the airplane incase everybody on board had to be evacuated in a hurry. It also had the second best capacity by size due to its large cargo bay making up most of its length. It wasn't able to maximize on loiter time as the blended wing was definitely more fuel efficient but as the mission profile was well within the endurance range of a C-5M and our design was to be a scaled down version of the C-5M this was of no major concern. It provided us with a great low speed performance as it's large wing area and thrust could let it go at slower speeds to capture the payload of the fairings and second stage which would enable a smoother recovery increasing the overall safety of the crew and the mission. As the C-5M was designed for rough terrain and small airstrips despite its massive size, it proved to be well within our runway limit at either of the runways at Cape Canaveral and Vandenberg Air Force Base.

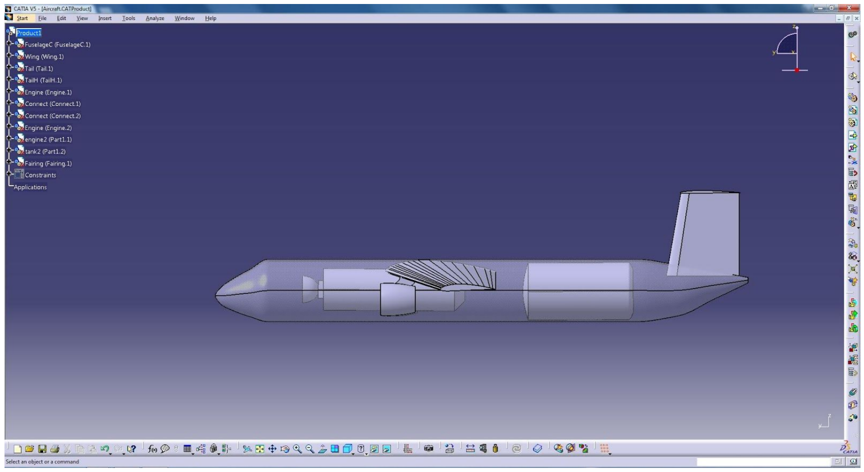
B. Aircraft Description

As we decided to base our aircraft on the C-5M Super Galaxy but specially designed it for recovering the rocket's second stage and fairing, we changed the size of the aircraft and added minor changes so that it can meet the high altitude heavy load recovery mission. We stuck with the high wing placement on our design as it helps give better clearance for recovering the payload and also offers more stability and structural support. We changed the wing design to be a straight wing as we did not need to go very fast on this mission and at higher altitudes a straight wing would provide much greater lift as when compared to a swept wing which would have been better if we needed high speed in our mission[12]. We also decided to give the wing a taper ratio of 0.5 as this would decrease induced drag significantly and since induced drag accounts for the most amount of drag at the mission's designated speed and altitudes. We also decided to make the wing slope down by 12 degrees making it anhedral by design to improve stability at higher altitudes and with heavy payloads which might interfere with the moments affecting the plane. We also decided to add winglets like the ones found on the C-5 Globemaster III as these provided it with a Oswald Efficiency Factor of 1.01 which greatly improved the aerodynamics of the plane. We added in a High T-Tail as the downwash from the powerful engines and turbulent airflow would've resulted in a very inherently unstable plane if the tail was placed conventionally. This also allowed us to add large horizontal and vertical stabilizers which could better support the plane's pitch and yaw with the massive payloads it would be carrying. We decided to change the material of the body to a lighter weight yet more durable Aluminum 7075 alloy which is treated with zinc to prevent corrosion and can be strengthened to a capacity of 600 MPa as compared to the Aluminum 6061 alloy out of which the plane was initially made.

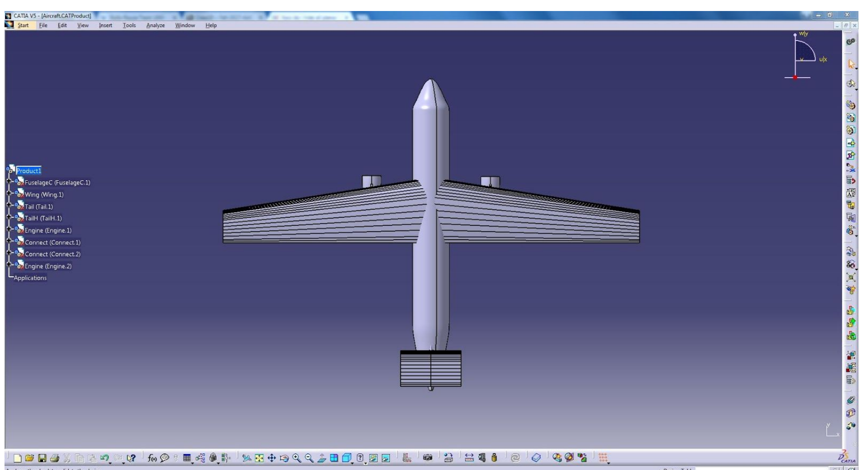




To fit the second stage and fairings, we modified the design of the C-5M and decided to remove the upper level crew quarters and place those right behind the cockpit as this would give us increased height to accommodate the payload and widen the fuselage by a few decimeters. Our updated design shortened the overall length to 42 meters, increased the width of the plane to 6.5 meters and kept the height constant at 7.1 meters. This significantly impacted the cargo bay of the aircraft too which was shortened to 30 meters in length, 5.8 meters in width and 6.5 meters in height. This was large enough to accommodate the payload with enough room to spare as the maximum diameter of the payload was 5.2 meters and the combined length of both the payloads was a 22.3 meters which fits very comfortably in our cargo bay. Approximating the weight we figured our aircraft would weigh significantly less than the C-5M and would be around 200,000 kgs fully loaded along with the payload.



We solved for Wing Area (shows as A) using the equation for lift $L = \frac{1}{2} * C_l * \rho * V^2 * A$ by setting flight conditions at the roughest that they could get which would be with maximum payload at 45,000 feet. This let us set Lift equal to the weight of the aircraft which came out to be 356072.524 lbs with the payload. We also determined density of air at 45,000 feet and set our velocity to a fair range of 476.305 ft/sec and this enabled us to get a wing area of 3880.3 ft^2. With the wing span remaining the same as the C-5M which was 229.659 feet we were able to get an aspect ratio of 13.5926 which is very respectable for a plane of this size. All of the calculation found can be double checked using the Aircraft Estimation Matlab code found in the appendix.



Our aircraft starts its journey fully loaded with 44092.452 lbs of fuel and a total wet weight of 343159.432 lbs. With its total thrust of 117732 lbf coming from a twin Trent 1000 engine setup, it will require 3203.3 feet to takeoff which is well below the 15,000 feet limit. It will then start its climb to 45,000 feet with a Rate of Climb of 151.3873 feet/sec and should reach that altitude in about 14.95 minutes while throttling the engine to maintain a healthy fuel consumption rate. At this rate, the endurance of our aircraft with max payload to max payload with 20% fuel remaining is 252.3308 minutes which is more than enough for the required mission time. At this point our aircraft would loiter until it encounters the payload it is supposed to capture.

We plan on capturing the rocket parts using the same approach as was employed by the Corona Satellite which is slowing down the stages and deploying drogue parachutes well above 45,000 feet so that they are slow enough to be captured by the aircraft. We plan on capturing the moving payloads at an intercept speed of 410 ft/sec which is well above the calculated stall speed of 390.7673 ft/sec. This also accounts for the increase in drag while pulling in the payload. The plane's trim will be changed once the payload is adjusted in the cargo bay as well.

After capturing the payload it will start its unpowered descent back to the runway where it will take 2520.5 feet to land which is well under the 15,000 feet limit that we have.

C. Aerodynamics

We decided that for our airplane to have the best chance to capture the payload at that high an altitude successfully, we would need very high stability from the wings as it would be hard enough to capture a moving satellite in those low density, low lift conditions. We also valued the total lift it would provide at various angles of attack and the high L/D ratio it would provide to increase lift. So for this we compared various airfoils that we could find on the back of the book and in high altitude research papers and found that the NACA 4412 airfoil best suited our purpose. The Selig S1223 High altitude low reynolds number airfoil was a close second but due to the high instability of the airfoil at high angle of attack conditions it was not chosen as this would prove to be very dangerous at low lift conditions while performing maneuvers to capture the payload. The scoring system for this matrix is also provided below for reference.

Airfoil Selection	Weight	Selig S1223	NACA 4412	NACA 1412	NACA 4415	NACA 63-210
Cm vs AoA	33	4	8	2	0	6
L/D Ratio	27	8	6	2	4	0
Cl vs AoA	22	8	6	2	4	0
AoA	18	2	4	8	6	0
Total Points	100	560	630	308	304	198

Scoring System	
Rating	System
Best	8
Better	6
Average	4
Poor	2
Worst	0

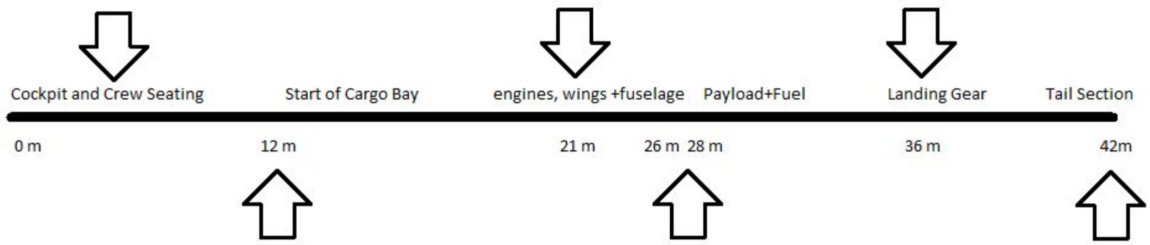
$$C_{D_p} = \sum_{i=1}^{\#\text{comp}} \frac{K_i Q_i C_{f_i} S_{wet_i}}{S_{ref}}$$

We calculated the total drag buildup using the equation provided to us in class by Dr. Chen and all of this was found using equations given in the slides which are discussed in more detail in the Matlab section of the Appendix. We found total parasite drag to be 0.00061465 slugs/ft^3, induced drag to be 0.0710 slugs/ft^3 and the drag coefficient at Cl_{max} to be 0.0224. This resulted in a total drag of 1913.2 slugs/ft^3. We also kept the high-lift devices from the C-5M as they were as they offered a 20% increase in C_L at low altitudes and also proved to be very helpful in providing drag to slow down the plane and come to a stop much before the maximum runway length.

D. Stability and Control

Using the C_M vs Angle of Attack graph of the NACA 4412 airfoil, we estimated the C_M of the aircraft to be -0.11 as it was the one value where the graph for C_M vs Angle of Attack was constant.
We found the center of gravity of the aircraft using the nose of the aircraft as origin and calculating the distance times the weight of individual aircraft components and summing them up and finally dividing them by the total

weight of the aircraft to get to know where exactly the center of gravity of the aircraft was located.



Component	Distance from Start of plane (in Meters)
Trent 1000 engine x 2	21
Weight of Fuselage and Wings	21
Weight of Payload	28
Weight of Landing Gear	36
Weight of Fuel	26
Center of Gravity =	22.55171939

Hence center of gravity was found to be located at 22.55 meters from the start of the aircraft. Calculations for it are also shown in the appendix.

E. Aircraft Propulsion

For the propulsion system, we have done a thorough research on modern turbofan engines from various manufacturers such as GE Aviation, Rolls Royce, and Pratt & Whitney. Our goal was to find an engine which provides high thrust with high efficiency, and low weight to maximize the aircraft's performance and to be economically feasible.

Rank	Engine	Weight	GE90-115B	CF6-80E1	GENx-1B76	Trent XWB-97	Trent 1000	PW 4098
1	Thrust to Weight ratio	33	6	4	0	2	10	8
2	Max Thrust	27	10	0	2	8	4	6
3	Efficiency (Takeoff Pressure Ratio)	22	4	0	6	10	10	2
4	Dry Weight	18	0	10	6	2	8	4
	Total Points	100	556	312	294	538	802	542

Scoring System	
Rating	**System**
Best	10
	8
	6
	4
	2
Worst	0

As shown in the chart above, the Trent 1000 engine from Rolls Royce is chosen to be the best fit for our recovery aircraft. We thought that the thrust to weight ratio is the most meaningful and important to consider. Even though the Trent 1000 engine does not provide the most thrust out of all of the candidates, but it generates enough thrust for our aircraft to reach the recovery altitude and stay in the air until the upper stage is captured. We also have looked at the pressure ratios to evaluate engine efficiency since one of our major concerns is the amount of money required for the mission. The Trent 1000 was the winner in this category as well.

The Trent 1000 engine has a thrust to weight ratio of 6.01, which is 11% greater than the TF39-1C, the engine attached on the C-5's[15]. Also, the Trent engine produces max thrust of 78,000 lbf (34,700 N). This number is almost twice as large as the TF39-1C.[16] Its high thrust to weight ratio while having a high efficiency made it the best fit for our aircraft, and therefore we decided to implement two of them onto our C-5 based recovery aircraft.

F. Weights and Cost

After the designing process, we were able to calculate the total weight and the center of gravity as well. The total dry weight of the aircraft turned out to be 135,654.5 kg, and it would increase to 161,511.78 kg once it is loaded with payloads and enough fuel to return to Earth. We were able to estimate and calculate the mass of the recovery aircraft by determining the fuselage material and estimating the total volume of the aircraft. The product of density and the volume gave us the estimated mass of the aircraft.

Component	Mass (kg)
Trent 1000 engine x 2	11872
Density of al 7075 kg/m^3	2810
Estimated volume of Fuselage m^3	41.45
Weight of Fuselage and Wings	116474.5
Weight of Payload	5857.28
Weight of Landing Gear	7308
Weight of Fuel	20000
Total Weight of Aircraft with Payload	161511.78
Total Weight of Aircraft without Payload	155654.5
Dry Weight of Aircraft	135654.5
Inert Mass of Aircraft	135654.5
Cost of Fuel ($)	2000000
Cost of Inert Mass ($)	135654500
Total Cost of Aircraft ($)	137654500

During the designing process, we put lots of effort on decreasing weight because we were aware that massive aircraft will result in high operational cost, which is not very desirable. In the end, we estimated fuel cost of $2,000,000 and the inert mass cost of $135,654,500. These costs are summed up to $137,654,500. This cost is almost half of the unit cost of the C-5, so we believe that the aircraft is economically feasible while bringing the best performance it can handle.

VI. References

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%Constants determined through sources matching C-5M Super galaxy and
rest taken from behind the textbook Introduction to Flight by John D
Anderson Jr.:
Pres45k = 309.45; %lb/ft^2
DensitySL = 2.3769*10^-3; %slugs/ft^3
Density45k = 4.6227 * 10 ^ -4; %slugs/ft^3
Weight1 = 356072.523883; %lbs
g = 32.2; %ft/s^2
TSFC = 0.576; %(lbm/hr)/lb
Weight0 = 299066.97946; %lbs
WingSpan = 229.659; %ft
ClMax = 2.6;
Cl = 1.75;
Thrust = 58866 * 2; %lbf
LoverD = 78.1;
vel45k = 476.3057743; %ft/s
M = 0.4232594752; %mach
Ct = 0.054; %(lbm/min)/lb
e = 1.01; %Oswald Efficiency Factor
%Calculations:
q = 0.5 * Density45k * (vel45k ^ 2) %Dynamic Pressure at 45,000
feet
S_Wing = Weight1 / (q * Cl) %Wing area required to support
flight at 45,000 feet
AR = WingSpan^2 / S_Wing %Aspect Ratio of Aircraft
Len_Fuse = 137.795 %Length of Fuselage in ft
Dia_Fuse = 21.3255 %Diameter of Fuselage in ft
FineRat_Fuse = Len_Fuse / Dia_Fuse %Fineness Ratio
Swet_Fuse = pi * Dia_Fuse * Len_Fuse * ((1 - (2 /
FineRat_Fuse))^(2/3)) * (1 + (1 / (FineRat_Fuse ^ 2))) %Wetted area
of Fuselage m^2
Len_Nac = 15.652887; %Length of Nacelle in ft
Dia_Nac = 6.230315; %Diamater of Nacelle in ft
Swet_Nac = pi * Dia_Nac * Len_Nac %Wetted Area of Nacelle in ft^2
Swet_Wing = S_Wing * 2 * 1.02 %Wetted Area of wing in ft^2
Swet_Total = Swet_Fuse + Swet_Nac + Swet_Wing %ft^2
wingSweep = 0; %Straight Wing
sweepCorFac = (((2 - M^2) * cos(wingSweep)) / (sqrt(1 - (M^2 *
(cos(wingSweep)^2))))) %Sweep Correction Factor
K_Fuse = 1 + (60 / (FineRat_Fuse) ^ 3) + (FineRat_Fuse / 400) %Form
factor of Fuselage
1
K_Nac = 1 + (0.35 / (Len_Nac / Dia_Nac)) %Form factor of Nacelle
K_Total = K_Fuse + K_Nac %Total Form Factor
C_Total = 0.455 / ((log10(100000) ^ (2.58))) %Total skin Friction
V_stall = sqrt((2 * Weight1) / (Density45k * S_Wing * ClMax)) %Stall
Speed in ft/s
V_takeoff = 0.7 * 1.2 * V_stall %Takeoff Speed in ft/s
Drag_para = (K_Total + C_Total) / S_Wing %Parasite Drag
Drag_induced = Cl^2 / (pi * e * AR) %Induced Drag
Cd = Cl / LoverD %Coefficient of Drag
Cd_Total = Cd + Drag_induced + Drag_para %Total drag coefficient
Drag_Total = (q * Cd_Total * S_Wing) %Total Drag in lbf
Lift_Total = (q * Cl * S_Wing) %Total lift in lbf
```

Range = 2 * (sqrt(2 / (Density45k * S_Wing))) * (1 / TSFC) * (sqrt(Cl)/Cd_Total) * (sqrt(Weight1) - sqrt(Weight0)) %Range of aircraft in nautical miles
Endurance = (1 / Ct) * LoverD * log(Weight1 / Weight0) %Endurance of aircraft in Minutes
TakeOff_Dist = (1.44 * (Weight1 ^ 2)) / (g * DensitySL * S_Wing * ClMax * ((Thrust) - (Drag_Total + (0.07 * (Weight1))))) %Takeoff Distance in ft
Landing_Dist = (1.69 * (Weight0 ^ 2)) / (g * DensitySL * S_Wing * ClMax * ((Thrust) - (Drag_Total + (0.07 * (Weight0))))) %Landing Distance in ft
ROC = vel45k * ((Thrust / Weight1) - (1 / LoverD)) %Rate of Climb in ft/s
ChordLen = 35; %Chord Length in ft
ac = ChordLen / 4 %Aerodynamic Center of Chord
Cm = -0.11 %Coefficent of Moment for airfoil
q =
52.4369
S_Wing =
2
3.8803e+03
AR =
13.5926
Len_Fuse =
137.7950
Dia_Fuse =
21.3255
FineRat_Fuse =
6.4615
Swet_Fuse =
7.3846e+03
Swet_Nac =
306.3757
Swet_Wing =
7.9158e+03
Swet_Total =
1.5607e+04
sweepCorFac =
2.0098
K_Fuse =
1.2386
3
K_Nac =
1.1393
K_Total =
2.3779
C_Total =
0.0072
V_stall =
390.7673
V_takeoff =
328.2445
Drag_para =
6.1465e-04

Drag_induced =
0.0710
Cd =
0.0224
Cd_Total =
0.0940
Drag_Total =
1.9132e+04
Lift_Total =
3.5607e+05
4
Range =
2.5713e+03
Endurance =
252.3308
TakeOff_Dist =
3.2093e+03
Landing_Dist =
2.5205e+03
ROC =
151.3873
ac =
8.7500
Cm =
-0.1100
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