TOWARDS A TETHER BASED FREIGHT DELIVERY INFRASTRUCTURE BETWEEN EARTH AND MOON

Final Report

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Abstract

The aim of the project is to construct a mission architecture for the bidirectional transportation of 1000kg of freight between the Earth and the Moon utilising Motorised Momentum Exchange Tethers (MMETs). This report will outline the research conducted thus far; focusing on transportation methods from Earth to Low Earth Orbit (LEO), Moon based tether system to the lunar surface, and the technical aspects of MMETs that will be necessary to calculate the specification required for their design. The future steps of the project will then be discussed which will form the building blocks of the mission architecture as a whole.

Contents

1 Nomenclature

A	cross sectional of tether, m^2
a_n	semi-major axis of orbit n, $a_n = \frac{1}{2}(r_{nP} + r_{nA})$
b	semi-minor axis of orbit
CoM	centre of mass of tether
E	eccentric anomaly
e_n	eccentricity of orbit n
H_{nP}, H_{nA}	pericentre and apocentre altitude of orbit n
I_{Pl} , I_T	mass moment of inertia of payload and tether
i	orbital inclination, degrees
L	tether length from CoM to payload
M_M,	motor and payload mass, kg
$M_{Pl,1}$, $M_{Pl,2}$	mass of upper and lower payload respectively, kg
R_E	radius of the Earth, 6371km
R_M	radius of the Moon, 1737km
$r(\theta)$	radius of an orbit at true anomaly θ
r_c	circular orbit radius at payload release, m
r_M, r_{Pl}	radius of motor and payload, m
r_{nP}, r_{nA}	pericentre and apocentre radius of orbit n
T	rotational or orbital period
v_{nP}, v_{nA}	pericentre and apocentre velocity of orbit n
$v_{Pl,1}$, $v_{Pl,2}$	tangential velocity of upper and lower payload respectively
$v_{tip,E}$, $v_{tip,M}$	tangential tip velocity relative to the Earth and Moon,
$v_{tip,e}$, $v_{tip,l}$	tangential tip velocity relative to the eMMET and lMMET,
$x_{Pl,1}$, $y_{Pl,2}$	position of upper and lower payload on orbit about body
$\dot{x}_{Pl,1}$, $\dot{y}_{Pl,2}$	velocity component of upper and lower payload on orbit
β_i	inertial launch azimuth
$\dot{\theta}$	orbital angular velocity of tethers centre of mass
μ	gravitational parameter of body being orbited
ρ	density of tether, kg/m^3
σ	tensile strength of tether, N/m^2

τ	motor torque, Nm
ϕ	latitude on Earth's surface, degrees
$\ddot{\psi}$	angular acceleration of tether, rad/s^2
$\dot{\psi}$	angular velocity of tether, rad/s
ψ	angular displacement of tether, rad

Subscripts:

S	refers to stator parameters
e	with respect to the eMMET's centre of mass
l	with respect to the lMMET's centre of mass
E	with respect to the Earth's centre of mass
M	with respect to the Moon's centre of mass

2 Introduction

The construction of a mission architecture for the bi-directional transport of freight between the Earth and the Moon utilising Motorised Momentum Exchange Tethers (MMET) is presented in this report.

The mission will consist of 3 main transportation stages;

- Earth's surface to the LEO based tether, eMMET
- The eMMET to the Moon based tether, lMMET
- The lMMET to the lunar surface

The cost of such a mission will be estimated and compared to conventional rocketry to determine the feasibility of such a system. There has been many studies into the detailed design requirements and dynamical problems that would be need to be overcome if such a system were to be implemented. To achieve a full Earth to Moon transfer via MMET's the study focuses on the key design features of the MMET's, such as the velocity components required and the stress within the tether lengths.

The tethers that will be utilised will be symmetrical dumb bell tethers spinning about a central facility that will house the motor and power source and storage. To maintain symmetry the tether will perform catch and release operations at the same time to ensure the dynamics of the tether remained balanced. This means that the eMMET must catch a payload from the Earth at the same time it catches a payload incoming from the lMMET, and release said payloads at the same time.

Finally, conceptual design solutions to logistical and system performance concerns such as, power supply, safety and infrastructure, GNC and telecommunications will be explored. Moreover, a basic cost analysis will be run throughout the architecture design to give an idea of the expense required to implement such a system.

3 Methods of Earth to LEO Delivery

3.1.1 Conventional Rocketry

A variety of technologies have been utilised to inject payloads of numerous sizes into a Low Earth Orbit (LEO). Of course, the most widely known and used was and remains conventional rocketry. Conventional rockets use a fuel mixture of liquidised hydrogen and oxygen to propel the rocket and its payload vertically upwards, making use of Newton's 3rd Law. These rockets are non-reusable and extremely inefficient in that 80-90% of the initial mass of the system is made up of propellant i.e. not payload. For this mission, conventional rocketry is deemed unlikely to suit the demands of the architecture in that the bi-directional nature requires a more continuously usable system to facilitate frequent payload transfers. Additionally, the cost per kilogram of payload to LEO is high. However, when insurance and sub-contractor costs are factored in the charges levied to customers are much larger, though it is important to note the distinction between the costs for a government backed launch compared to the price charged by a profit oriented private venture. It should also be noted that conventional rocketry is likely to be required in part for the construction of the tether system, however this will be discussed in due course.

3.1.2 Rocket Integration Technology

There are two methods of integrating conventional rockets with other common modes of altitude gain. The first method considered is the integration of High Altitude Balloons (HAB) with rocketry. Whilst the balloons would allow for a reduced fuel requirement, since the rocket would not have to boost through the fuel intensive lower atmosphere, this method was discarded for several reasons. Firstly, it would add complexity to an already very complex mission architecture for what was realistically very little saving in terms of cost once the added complexity and infrastructure required was considered. The largest balloons accommodate a payload of around 90kg and are difficult to steer as well as highly dependent on weather conditions. The only positive to be considered was the added flexibility in terms of launch sites, however for a project of this scale, likely involving mass government cooperation, this was deemed non-essential.

The second integration technology considered was the use of a traditional aeroplane to ‘bus’ a rocket through the fuel intensive lower atmosphere to be launched i.e. a two-stage rocket launch. This was similar to HAB in its benefits in that it could use a conventional runway to launch. Further to this, benefits included larger payload carrying capabilities compared to HAB and the fact that conventional aircrafts are not affected by weather delays in the same ways as rocketry. Air launch takes place in the stratosphere and not the troposphere so the launch is not usually affected by conventional weather. Orbital Sciences Pegasus rocket has achieved multiple successful air launches with payloads up to of 473kg [1]. Savings in terms of cost are again minimal due to added complexity as well as aircraft fuel costs. Moreover, the rocket must perform a large direction change from horizontal flight to vertical. This is achieved using a large Delta wing however this adds mass to the rocket and, even with this, a portion of the initial horizontal velocity is lost in the transfer. Ultimately added complexity and no real saving in terms of cost meant this option was not considered viable.

3.1.3 Non-Conventional Reusable Rocketry (i.e. VTOL)

Re-usable rocketry is one of the two most promising advances in space exploration in recent years. A large portion of the cost for conventional rockets comes from the fact that the booster engines of the rocket must be discarded after use, incurring large capital costs. Making the process reusable would greatly reduce the cost of space travel and large steps have already been made towards making this a reality. The SpaceX program has already successfully managed to vertically land its Falcon 9 rocket and whilst the rocket is not reusable as yet it is highly likely it will be by 2027, when this mission would hope to be implemented. The SpaceX rocketry, once reusable, quotes a customer charge of $4640/kg to LEO for the Falcon 9 rocket carrying up to 22800kg and a very speculative cost of $1700/kg for the Falcon Heavy, essentially a Falcon 9 with two additional booster engines, carrying a maximum payload of 54400kg [2]. NASA has already contracted SpaceX to supply the International Space Station therefore this technology is proven to work and, of equal importance, the fact it is reusable ~10 times means it is both flexible and consistent enough to satisfy the requirements of the mission. This is ultimately why this method of transport was chosen for the transfer of payload from Earth to Low Earth Orbit. However, in the future more suitable options may be available.

3.1.4 Single Stage TO Orbit Spaceplane (i.e. HTOL)

Single stage launch directly to orbit using spaceplanes is perhaps the most exciting and innovative potential advance in space technology. Foremost in this field at present is the Skylon spacecraft designed by British company Reaction Engines Limited. Skylon intends to utilise a SABRE engine which combines air breathing technology and traditional rocket propulsion to launch 15 tonnes of payload into LEO at a projected cost of $1500 to $2700 depending on how optimistic the estimate [3]. It is initially designed to operate as a conventional jet engine up to Mach 5.5 and 26km altitude after which point the air inlets would close and the spacecraft would operate as a normal rocket. This combination of technologies would allow a reduction in propellant usage since conventional engines, in conjunction with winged crafts, are much more efficient in counteracting the drag forces experienced in atmosphere than simply expelling propellant. Additionally, this system is intended to be fully reusable up to 200 times once operational which would provide a large advantage for the purposes of the mission proposed. Unfortunately, REL are struggling to procure funding for their spaceplane at present and their project, as well as others like it, are merely in the early stages of design and testing with no significant vehicle tests scheduled in the near future. The future of space travel relies on cost reduction so both the reusable VTOL spacecrafts, such as SpaceX's Falcon range, and HTOL crafts, such as REL's Skylon, are likely to have a place in future orbital missions once fully realised. However, for the purposes of this mission which has a view to implementation by 2027, the SpaceX program, and reusable VTOL in general, are the most viable options for payload transfer from Earth to LEO.

4 eMMET to lMMET Transfer

At present the transport of payload, be it freight or passengers, into the Earth's orbit and beyond requires the use of propellant to provide enough energy for the journey. The use of large Motorised Momentum Exchange Tethers (MMETs) to propel a payload from a specific orbit to a higher orbit, or into an escape planetary transfer, would reduce the dependency on propellant for such missions.

The transfer of payload between the Earth and the Moon using MMETs is dependent on the dynamics of the tether systems, and their interactions with the payloads, on their orbits. The level of detail which could be considered is vast and varied. It was important that the correct level of detail was chosen so that a proper construction and assessment of the mission architecture could be conducted. Space is three dimensional and a real tether system would be fully described within this space. As a mission architecture of this scope has not been conducted previously, a baseline design had to be laid out. Therefore, the tether system was chosen to be modelled ideally in a two-dimensional space, with defined assumptions. The tethers were constrained to rotate with two degrees of freedom in a circular motion. This meant that any destabilisation of the tether due to planetary perturbations or incorrect payload capture could be neglected. The dynamics of a real MMET would be complex with vibrations of the tether length and its elasticity impacting its motion. Ismail and Cartmell examined these phenomena in two and three dimensions and the extension of the tether lengths, at varying angular velocities, would have an impact on the position of the payload. Additionally, the global motion of the tether could become chaotic under certain conditions [4]. The implementation of such a model for this study was out with the scope of the project and would have added an unnecessary level of complexity to the mission architecture. It is equally possible to under define the system which would lead to a lack of accuracy and an incomplete analysis and hence an invalid result of the project objective. The tether was therefore assumed to act as a rigid body with a mass.

Since the dynamics of the MMETs were to be analysed in two dimensions, the orbital and rotational planes of the tethers had to be coplanar. This was necessary because the payload had to meet the upper tip of the tether with zero relative velocity for the tether balance to be maintained. It would not be possible to model the tethers in two dimensions without the payload incoming on the same plane. The tethers were chosen to orbit and rotate on the Moon's orbital plane about the Earth. If an inclination change between the eMMET and the lMMET was to occur, then the payload trajectory would have to be altered so as to align it with the tether plane. This would involve using a sizeable amount of chemical propellant and therefore detract from the purpose of the mission.

Now that the plane of the orbits was determined it was necessary to define the mechanical model of the orbits that would be used for the transfer calculations. To achieve this, the Earth and Moon were assumed to be perfectly spherical and perturbation effects were neglected. The Earth based tether was to operate within the lower region of the LEO, where atmospheric drag would affect the tethers motion. This effect was also neglected within the calculations. Although, the orbits of the tethers were idealised when calculating the parameters of the tether system, these effects and required solutions were considered and are discussed in Section 11.2.

The tethers were modelled to be symmetrical about their centre of mass and the tether would catch payloads at each end simultaneously and release the payloads from each end simultaneously. The capture and release operations were assumed to occur perfectly and instantaneously. So, the physical interaction, such as energy dissipation, between the tether tip and the payload could be ignored. Additionally, destabilisation due to failed captures was not considered. These assumptions meant that the tether would maintain its orbit about its respective celestial body. Again in reality the discontinuity of the mass of the tether between payload capture and launch would result in a difference in energy and hence a long-term change of the orbit [5].

5 Lunar Orbit to Lunar Surface Transfer

5.1 Powered Descent

A large number of issues arise when travel beyond LEO or GEO is considered, in this case to the Moon. Obviously, the infrastructure does not exist on the Moon to accommodate fuel production therefore any fuel for a return journey must be transported there. Thus, the fuel requirements increase greatly and so too does the weight allowance for said fuel. Since, a powered descent is currently required to touch down on the lunar surface, fuel demands mean profitability of any such private excursion to the Moon would be minimal. The current rate to transport 1kg of cargo privately from the surface of the Earth to the lunar surface is a minimum of $1.2 million with additional costs incurred depending on specific requirements for the payload such as communication, power and thermal control needs [6]. These prices are quoted from Astrobotics, the leading competitor in the private sector, who work in partnership and as a sub-contractor of NASA. From a Payload User Guide produced by Astrobotics, the company has given data and a mission architecture for its first planned mission to the Moon. In this it gives the dry mass of the lunar lander as 222kg, with 35kg of this being the payload allowance. Also given is the wet mass of the lander i.e. the mass of the lander plus cargo and fuel which totals 700kg. Thus, only 5% of the mass which leaves Earth actually consists of useful payload whilst over 68% consists of fuel required to transport it. Reducing the mass of fuel required for this mission would greatly reduce the cost per kilogram of payload and, in turn, increase the feasibility of such enterprises. It is important to note the figures quoted above are for a one-way transfer, so no return journey of the lander module would be possible. This greatly hampers investment in space travel beyond LEO since any vehicles used are not reusable.

5.2 **Lunar Tether**

Whilst all currently practical methods of transporting cargo from a lunar orbit to the lunar surface are based on a powered descent, there is promising work ongoing in the field of lunar based tethers. In 1978, Moravec [7] put forth the idea of a non-synchronous tether, or ‘Lunar Skyhook’ as he named it, which could provide a minimal propellant transfer of payload from lunar orbit to the lunar surface. He proposed a large central facility with two tether arms both having length equal to the orbit distance of the aforementioned facility. It was to rotate in prograde with its orbital rotation with the velocity magnitude at the tip of each tether equal to the orbital velocity of the central facility, thereby allowing the velocity relative to the surface of the Moon to be zero at the point where the tether tip ‘grazes’ the lunar surface. The best way to visualise this is to imagine the tether as the spokes of a bicycle wheel rolling around the surface of the Moon. Moravec also concluded that if the arms had equal lengths then an arm length of 1/6 the Moon’s diameter would allow the mass of the tether system to be minimised. This would allow each of the arms to contact the surface at a rate of 3 per orbit which could, in theory, allow 6 payload transfers, at different locations, per orbit. An illustration of this process can be seen in Figure 1.

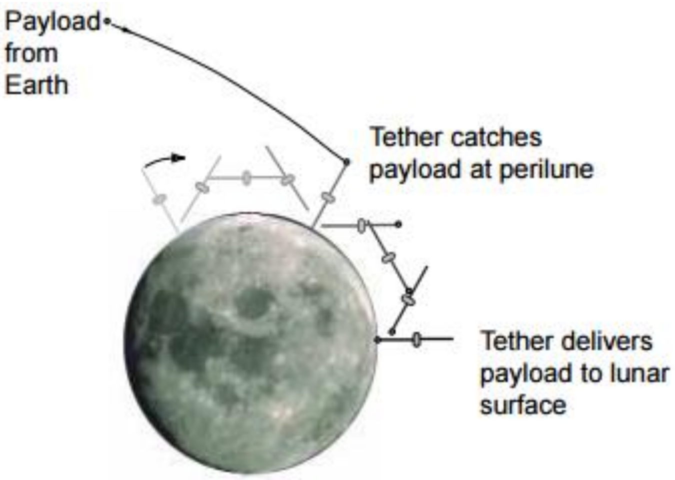

Figure 1: Illustration of Moravec's Lunar Tether [8]

In reality a tether which literally touches the lunar surface would likely be very impractical due to the uneven nature of the Moon's surface. Additionally, this would not allow a very large margin for error, which, given the scale of the tether system and the accuracy of timing required could result in operational failure of the tether. In reality the tether would possibly extend to a distance of perhaps 100m-1km from the surface and then release the payload which could either be caught by an appropriate system such as a safety net, or the payload could even be equipped with a small rocket booster and orientation system to lower itself to the surface.

This release of a payload, if not symmetrical, would produce instability in the orbit of the tether system which, while it is correctable, is not an entirely desirable impact on such a system. Therefore, it would be preferable if each arm of the tether was loaded and unloaded simultaneously. As an example, when the payload is released to the lunar surface a payload of equal mass would be released on an Earth transfer trajectory and be caught at the eMMET. These catch and release operations would continue in tandem and, ideally, indefinitely as long as required.

5.3 **Comparison of Methods**

It is difficult to practically compare a standard powered descent with any Lunavator method since the latter only exists in theory. It is, however, possible to carry out a conceptual comparison based on sensible assumptions. Of course, the best method of comparison would likely be the economic cost and a breakdown of some of the costs involved in a project of this nature is given later in this report. In this section a comparison of basic economic factors will be carried out.

The main difference between the methods is that a powered descent will require a much larger delta v to successfully land on the lunar surface and will obviously require a larger amount of propellant. The actual costs for the propellant are minimal compared to overall project costs however in conventional rocketry the propellant can account for 70-80% of the overall wet mass meaning space and energy is being expended to launch propellant for use throughout the descent. If the Lunavator was used for descent some propellant would still be required for course corrections and possibly emergency manoeuvres, however, the delta v would be very small by comparison. Of course, there would be large capital costs to set up and develop the Lunavator system, for which conventional rocketry would be required. However, this initial outlay would be the main component of the overall costs and therefore it would be expected that for frequent, high volume traffic between the Earth and the Moon this method would lead to large transport cost reductions since the Lunavator process would be continuous and non-propellant intensive. It could transport much larger quantities of smaller payloads as opposed to the larger more infrequent payloads which would be transported by conventional rocketry. In this way, more mass could be transported for a lower cost since operational costs would be minimal by eliminating the requirements to send large quantities of propellant along with payloads.

Of course, this method would not entirely replace conventional rocketry since any further excursions would requires rocketry to set up an appropriate system so advancing technology in both fields would be equally pertinent.

Ultimately the deciding factor for the purpose of this mission architecture was the inherent bi-directional capability of the Lunavator system compared with conventional rocketry. Transporting sizeable payloads from the Lunar surface back into an Earth transfer orbit would require a powered ascent utilising fuel carried from the start of the mission. Given that the mission specifies the use of an Earth based tether payloads larger than 1000kg are difficult to transport due to material limitations. Requiring said payload to both descend to, and ascend from, the lunar surface would require an unacceptable fuel fraction and thus make this mission unfeasible before its inception.

Subsequently the Lunavator design was chosen to transport payload from lunar orbit to the lunar surface.

5.4 **Lunar Landing Site**

Since the transfer of payload was required to be carried out on the same plane, the positions for acceptable bases were limited to an orbital inclination of around 1.8º relative to the Moon's equator. Transport to a base not on this inclination would require the use of chemical propellant. Ideally the most viable positon for a lunar base would be located around the poles due to the likely presence of lunar ice in these locations. Were this base to be set up this ice could be utilised to produce consumables for human habitation as well as produce rocket propellant. This has the potential to greatly reduce the fuel mass fraction of spacecraft travelling from the Earth to the Moon and beyond since there is a reduced requirement to transport an entire journeys fuel from the deep gravity well of Earth into space. This base could serve as a refuelling station for passing spacecraft in a not too distant future.

Unfortunately, low orbits around the moon are highly unstable and erratic due to the presence of large concentrations of mass or 'mascons' [9] which cause anomalies in the moon's gravitational field. Imaging of these 'mascons' on the Moon are shown in Figure 2.

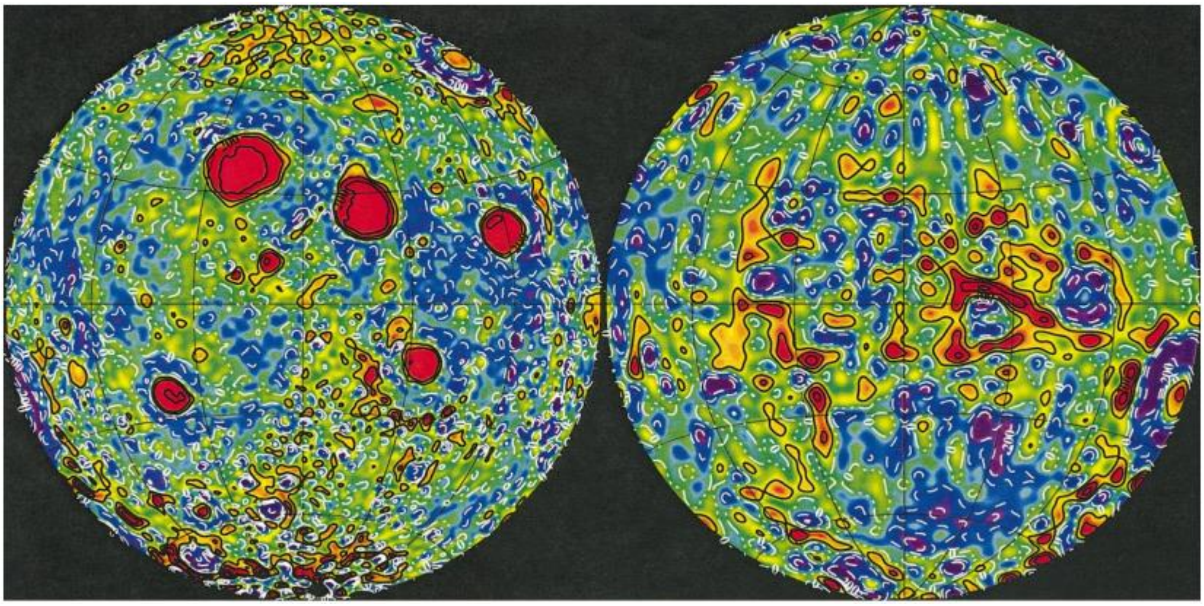

Figure 2: Imaging of mass concentration variation on the near (left) and far (right) sides of the Moon. The most significant 'mascons' are present on the near side and are clearly visible as 5 large craters shown in red above.

As a satellite passes over these, it will be pulled in an unpredictable manner thereby affecting its trajectory. There do exist 'frozen' orbits at 27º, 50º, 76º, and 86º inclination relative to the moon's equatorial plane upon which a satellite could remain in orbit indefinitely. Unfortunately, since the lMMET was decided to orbit on a plane of approximately 1.8º, it cannot orbit at an altitude which is affected by these 'mascons' i.e. below ~100km without significant use of propellant for station keeping. This was not desirable so an orbit altitude greater than 100km was selected although not solely for this reason.

In the end, the lunar base selection was highly constrained by the required orbital plane and so a base on the transfer orbit inclination had to be selected. Ideally a base around the poles would be chosen since there is a nearby 'frozen' orbit of 86º meaning the altitude of the lMMET could vary below 100km if required. This possibility could perhaps be explored in future works on the topic.

6 Tether Mechanics

6.1 Components

The key components of the MMET are the launcher motor, contained in the central facility (consisting of a rotor and a stator), the payloads and the tether lengths connecting them to the central facility, and the out-rigger masses and the tether lengths connecting then to the central facility. Although there would be many additional and vital components within the MMET, required for its operation, only the components listed above will be considered for this analysis. These are the components that will influence the MMET's capability to transfer payload, within a mission architecture of this kind. The layout of the MMET can be seen in Figure 3 [2].

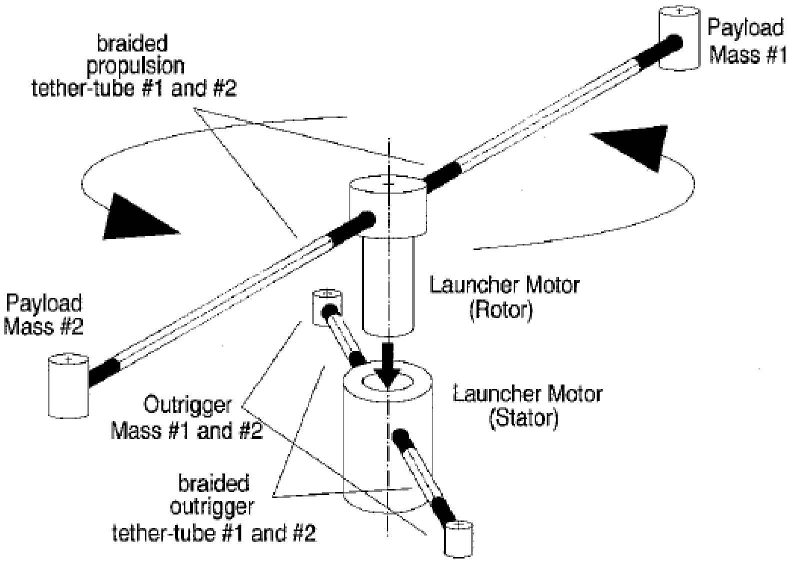

Figure 3: Conceptual MMET as suggested by Cartmell and Ziegler [2]

The mechanics and more specifically the dynamics of the tethers, both the eMMET and lMMET, will be analysed in 2D. The tethers will operate on the orbital plane of the Moon about the Earth. This allows the 2D analysis to be extended to the orbital transfer of the payload from the eMMET to the lMMET. The Moon's orbital plane varies from 18.28 - 28.58 degrees relative to the Earth's equatorial plane [10]. However, for the purposes of this study a value of 18.28 degrees will be assumed and will be explained in Section 0. The tethers will spin in prograde about their centre of mass with respect to their orbital motion. This motion can be seen in Figure 4.

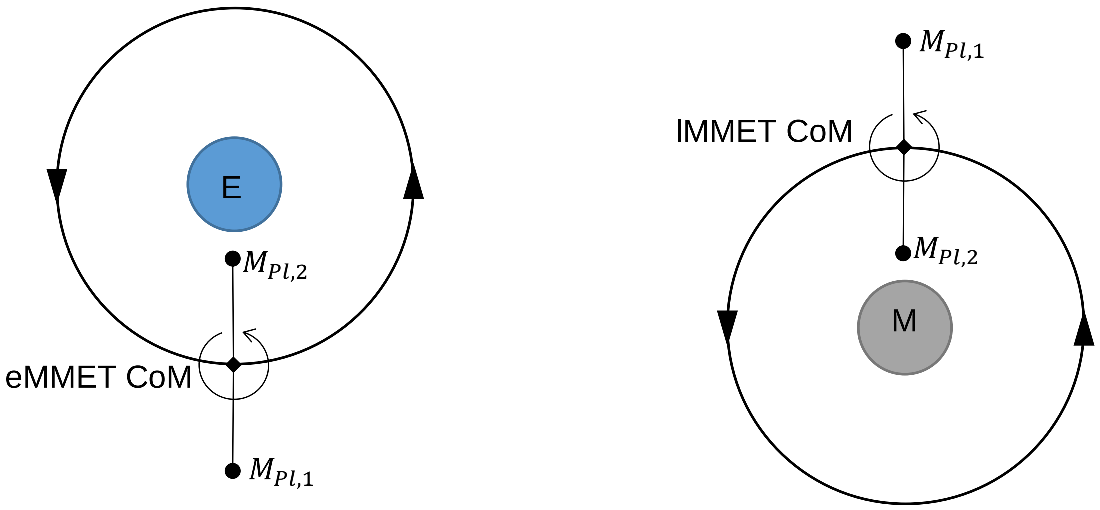

Figure 4: Rotational motion of eMMET (left) and lMMET (right)

6.2 MMET Velocity Vectors

The purpose of the MMET is to increase the momentum of a payload, by increasing its velocity, so that it can be transferred into a higher orbit. The orbit in question will be a lunar transfer orbit with its apogee at the lMMET's upper tip within the lunar orbit. The tangential velocity at a specific orbital radius, r, can be determined from Keplerian Motion [11]. Equation 1 gives the tangential velocity of an orbiting body;

$$v_{CoM} = \sqrt{\frac{2\mu}{r} - \frac{\mu}{a}} \tag{1}$$

This is the velocity that the centre of mass of the MMET will possess relative to the planetary body it is orbiting. The upper and lower payloads will have a different velocity than that calculated from Equation 1. If the tether is considered to be hanging, aligned with the gravity gradient of the orbited body, the upper and lower payloads will have the same orbital angular velocity as the CoM. This angular velocity can be calculated from Equation 2 where the velocity is that calculated from Equation 1 at the same orbital radius, r.

$$\dot{\theta} = \frac{v_{CoM}}{r} \tag{2}$$

Hence, the variation of velocity of the payloads due to their distance L from the tether CoM would be $L\dot{\theta}$. The velocity of the upper and lower payloads on a hanging tether are shown in Equation 3 [12].

$$\begin{aligned} v_{Pl,1} &= v_{CoM} + L\dot{\theta} \\ v_{Pl,2} &= v_{CoM} - L\dot{\theta} \end{aligned} \tag{3}$$

For the upper payload to have sufficient velocity to get to the Moon, additional velocity must be added. This is done by spinning the tether in prograde on the orbital plane. The tether will spin about its CoM at angular velocity, $\dot{\psi}$, so the tangential velocity of the upper and lower payloads, when connected to the MMET, will be calculated as shown in Equation 4 [12]:

$$\begin{aligned} v_{Pl,1} &= v_{CoM} + L\dot{\theta} + L\dot{\psi} \\ v_{Pl,2} &= v_{CoM} - L\dot{\theta} - L\dot{\psi} \end{aligned} \tag{4}$$

This additional velocity is only achievable when the tether is aligned along the gravity gradient of the celestial body which it orbits and the greatest velocity is achieved at the perigee of the tethers orbit.

6.3 **Tether Length Strength**

The stress in the tether is a function of the tether material properties and dimensions, the mass of the payload and the angular velocity of the tether about its CoM. The stress must not exceed the maximum tensile strength of the tether material and for the chosen material, Spectra 2000^{TM}, is 3.25GPa [5]. This relationship can be seen in Equation 5 [5].

$$\sigma = \frac{\dot{\psi}^2 L \left(M_{Pl} + \frac{\rho A L}{2} \right)}{A} \tag{5}$$

A safety factor (SF) of 1.5 was deemed adequate for the scope of this study, this ensured that a reasonable degree of error was considered. It is important, however, to note that Spectra 2000^{TM} is the strongest material currently available but this is unlikely to be the case when this system is implemented. The mission architecture is intended to be implemented in 2027 so material science is likely to have advanced greatly within this time period.

6.4 **Motor Torque**

The motor in the MMET is located in the central facility and is responsible for maintaining the angular velocity required to transfer the payload to its destination. It must also be able to achieve this angular velocity from rest. The torque required to 'spin up' the MMET over a time, t, was calculated. This was done in Mathematica 11.0, by determining the kinetic and potential energy of the tether, Equations 6 and 7 respectively, at x and y positions on a circular orbit. The kinetic and potential energy for a given torque were substituted into Lagrange's equation and the angular position and velocity of the tether about its CoM were plotted against time. The inertia of the tethers and the payloads are shown in Equations 8.

$$T_k = (0.5\times M_1(((\dot{x}_{Pl,1}[t])^2) + ((\dot{y}_{Pl,1}[t])^2))) + (0.5\times M_2(((\dot{x}_{Pl,2}[t])^2) + ((\dot{y}_{Pl,2}[t])^2))) + (0.5\times \rho AL_1(((\dot{x}_{T,1}[t])^2) + ((\dot{y}_{T,1}[t])^2))) + (0.5\times \rho AL_2(((\dot{x}_{T,2}[t])^2) + ((\dot{y}_{T,2}[t])^2))) + (0.5\times(I_{Pl,1} + I_{Pl,2} + I_{T,1} + I_{T,2})\times((\dot{\psi}[t] + \dot{\theta}[t])^2)) \tag{6}$$

$$U_p = -\frac{\mu M_1}{\sqrt{r_c{}^2 + L_1{}^2 + (2r_c L_1 \mathrm{Cos}[\psi[t]])}} - \frac{\mu M_2}{\sqrt{r_c{}^2 + L_2{}^2 + (2r_c L_2 \mathrm{Cos}[\psi[t]])}} - \sum_{i=1}^{n} \frac{\mu\rho AL_1}{n\times\sqrt{r_c{}^2 + \left(\frac{((2i)-1)L_1}{2n}\right)^2 + \left(\left(\frac{2r_c L_1((2i)-1)}{2n}\right)\mathrm{Cos}[\psi[t]]\right)}} - \sum_{i=1}^{n} \frac{\mu\rho AL_2}{n\times\sqrt{r_c{}^2 + \left(\frac{((2i)-1)L_2}{2n}\right)^2 - \left(\left(\frac{2r_c L_2((2i)-1)}{2n}\right)\mathrm{Cos}[\psi[t]]\right)}} \tag{7}$$

$$I_{Pl} = M_{Pl}(0.5r_{Pl}{}^2 + L^2)$$
$$I_T = \rho A\mathrm{L}\left(\frac{1}{12}(3r_T{}^2 + L^2) + \frac{L^2}{4}\right) \tag{8}$$

If the motor provided insufficient torque, the tether would not be able to ‘spin up’ to its required velocity. The torque was varied to determine the required amount for the tether to rotate fully. Figure shows plots of angular displacement, left, and angular velocity, right, against time of the IMMET for different values of torque. The value of torque in Figure (a) is 310MNm, (b) is 315Mnm and (c) is 625MNm. It can be seen that 315MNm of torque is enough to overcome the gravity gradient of the Moon, however, the angular velocity does not increase constantly and instead wavers. This motion of the tether could be potentially damaging to the tether structure, as well as the motor and its components. The torque was increased further until the variation of angular acceleration was minimised. The value of torque to achieve this was 625MNm of torque, and still a slight variation still exists.

This demonstrates the large values of torque that have to be applied by the motor to achieve the angular velocities necessary for payload capture and launch. The large tether lengths that would be needed for the lMMET would prove extremely difficult to spin up utilising a motor alone as even with a torque as large as 300MNm the tether swings and is unable to overcome the gravity gradient. The values of motor torque for the final lMMET and eMMET designs are displayed in section 10.

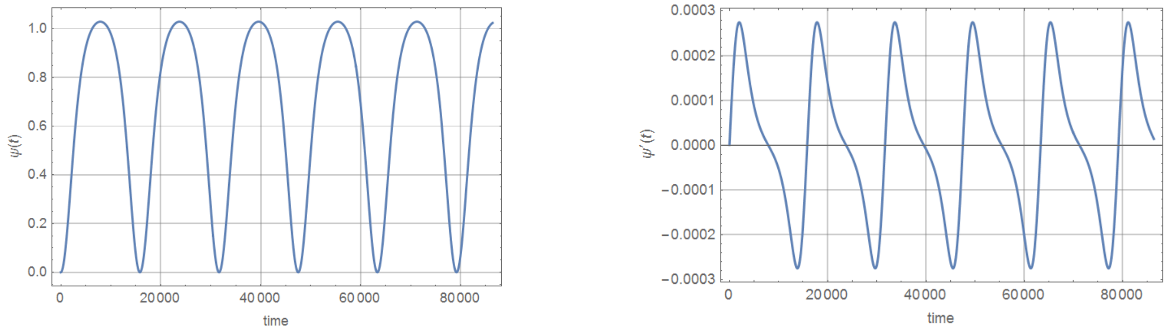

Figure 5 (a): Torque=310MNm. Angular displacement of the lMMET about its CoM (left) and angular velocity of the lMMET about its CoM (right). Time (s), Displacement (radians) and Velocity (rad/s)

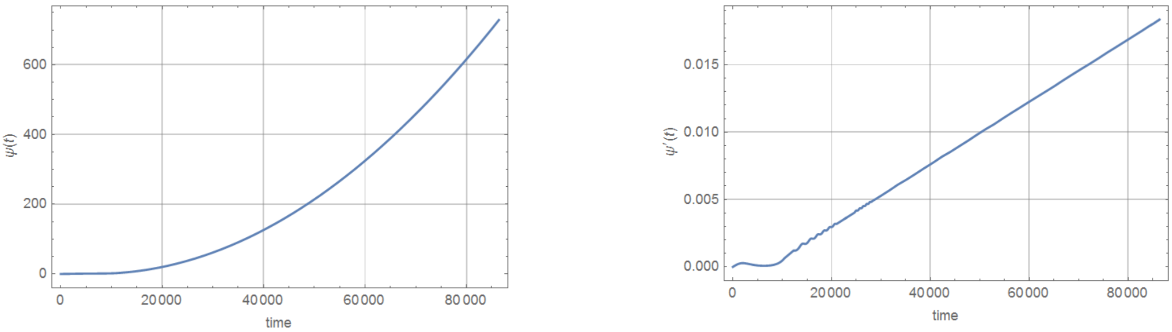

Figure 5 (b): Torque=315MNm. Angular displacement of the lMMET about its CoM (left) and angular velocity of the lMMET about its CoM (right). Time (s), Displacement (radians) and Velocity (rad/s)

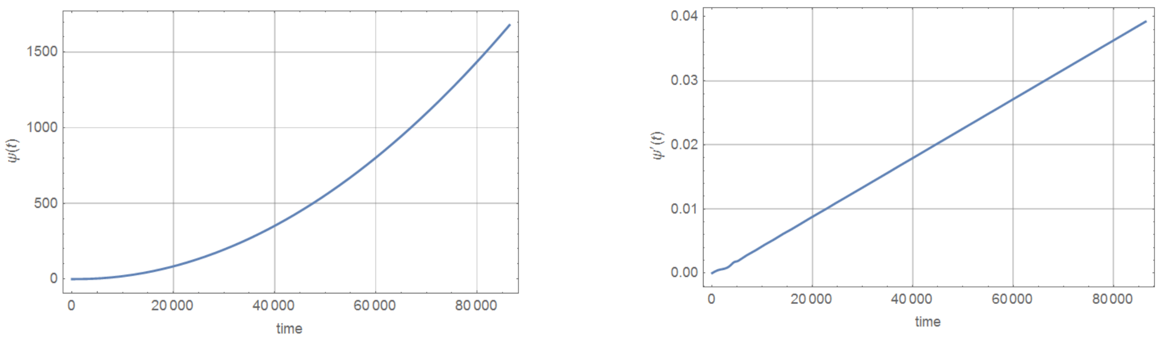

Figure 5 (c): Torque=625MNm. Angular displacement of the lMMET about its CoM (left) and angular velocity of the lMMET about its CoM (right). Time (s), Displacement (radians) and Velocity (rad/s)

6.5 Stator Reaction Torque and Dimensions

For a motorized ‘spin up’ of the MMETs a reaction torque would need to exist. This reaction torque is provided by a counter rotating stator, previously shown in Figure 3. The stator tether lengths were taken to be 10km less that their equivalent payload tether lengths. This would allow a healthy gap between the ballast masses and the incoming and outgoing payloads. The stator dimensions were assessed using the same method set out in Section 6.4 and the stator parameters are shown in Section 10.

7 Orbital Mechanics/ Transfer Calculations

The required velocity to transfer the payload into a Lunar Transfer Orbit was calculated using a low energy Hohmann transfer [11]. The orbits involved in the payload transfer from the eMMET to the lMMET and then from the lMMET to the eMMET are as follows

1. eMMET’s CoM orbit about the Earth
2. eMMET to lMMET transfer orbit
3. Moon’s orbit about the Earth
4. lMMET’s CoM orbit about the Moon
5. Circular Lunar orbit at radius $r_5 = r_4 + L_l$
6. lMMET to eMMET transfer orbit

The orbital parameters of each orbit will be referred to with a subscript number specifying the orbit as listed above and with a subscript identifying the position on the orbit where appropriate. These orbits are shown in Figure .

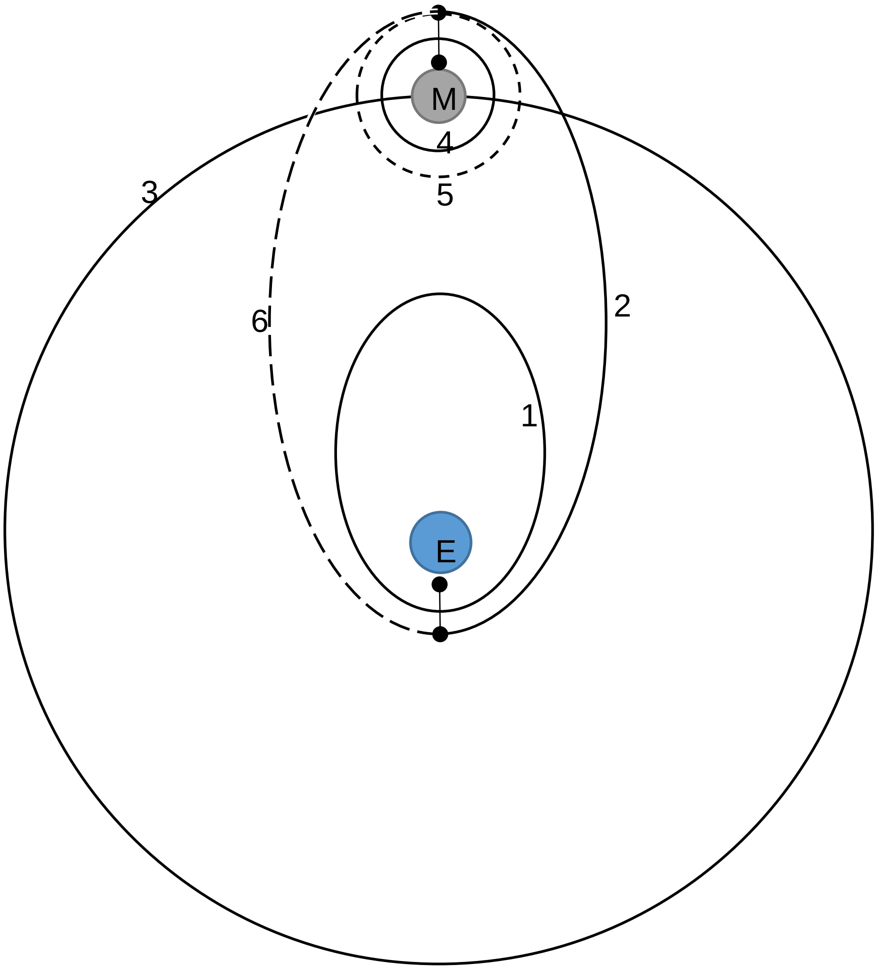

Figure 6: Tether based freight delivery infrastructure orbits (not to scale, 2D)

The velocity change required to transfer the upper payload of the eMMET to the upper tip of the lMMET was calculated for using a Hohmann transfer.

$$\Delta v1 = \sqrt{\frac{2\mu_E}{r_{2P}} - \frac{\mu_E}{a_2}} - \left[\sqrt{\frac{2\mu_E}{r_{1P}} - \frac{\mu_E}{a_1}} + L_e\dot{\theta}_{1P}\right] \tag{9}$$

Where, r_{2A} is the distance from the Earth's centre to the lMMET's upper tip.
So, the tangential velocity of the upper tip due to the rotation of the eMMET must equal $\Delta v1$. This allows the angular velocity of the eMMET about its CoM to be calculated from Equation 10.

$$\dot{\psi}_e = \frac{\Delta v1}{L_e} = \frac{v_{tip,e}}{L_e} \tag{10}$$

At the apogee of orbit 2, the velocity of the payload, arriving at the lMMET, must have zero velocity relative to the lMMET's upper tip. The payload at this point is assumed to be under the influence of the Moon's gravity and its velocity relative to the Moon is taken to be the orbital velocity at its radius from the Moon's centre, plus the velocity of the Earth relative to the Moon (equal to the velocity of the Moon about the Earth) and was calculated from Equation 11.

$$v_{Pl,M} = v_5 + v_{E,M} = \sqrt{\frac{\mu_M}{r_5}} + \sqrt{\frac{\mu_E}{r_3}} = v_{tip,M} \tag{11}$$

Since this study focuses on the feasibility of a mission architecture for the transport of freight from the Earth to the Moon this level of detail is acceptable.

Now that the velocity of the lMMET's upper tip has been defined the angular velocity of the lMMET about is CoM can be calculated by rearranging Equation 4 for the upper tip as seen below;

$$v_{tip,M} = v_4 + L_l\dot{\theta}_l + L_l\dot{\psi}_l$$

$$\Rightarrow\ L_l\dot{\psi}_l = v_{tip,M} - \left(v_4 + L_l\dot{\theta}_l\right)$$

$$\Rightarrow\ \dot{\psi}_l = \frac{v_{tip,M} - \left(v_4 + L_l\dot{\theta}_l\right)}{L_l}$$

When the payload is released it will have the same velocity as when it arrived allowing for the same return trajectory to be assumed. Hence, the payload will arrive at the eMMET, from the lMMET, with the same velocity as it departed.

8 Synchronising Tether Systems

For a synchronised tether system the orbital periods of the tethers, the rotational period of the tether tips and the timing to payload capture and release must meet the following requirements set out by Cartmell [13].

1. The orbital period of the eMMET must be even-integer harmonic with the orbital period of the Moon about the Earth. This ensures that the eMMET arrives at the

perigee of its orbit about the Earth when the Moon arrives at its predetermined launch position.

$$T_{eMMET} = \frac{T_{Moon}}{n_1} \qquad n_1 = even\ integer$$

2. The orbital period of the eMMET must be even-integer harmonic with the orbital period of the Moon about the Earth. This ensures that the eMMET arrives at the perigee of its orbit about the Earth when the Moon arrives at its predetermined launch position.

$$T_{lMMET} = \frac{T_{eMMET}}{n_2} \qquad n_2 = even\ integer$$

3. The orbital period of the eMMET must be even-integer harmonic with the orbital period of the Moon about the Earth. This ensures that the eMMET arrives at the perigee of its orbit about the Earth when the Moon arrives at its predetermined launch position.

$$T_{eMMET(tip)} = \frac{3T_{eMMET}}{2n_3} \qquad n_3 = even\ integer$$

4. The rotational period of the lMMET's tips must be odd-integer harmonic with the eMMET's orbital period, plus an added ¾ rotation. This allows the lMMET's tips to perform the same catch and throw operations as the eMMET's tips.

$$T_{lMMET(tip)} = \frac{7T_{eMMET}}{4n_4} \qquad n_4 = odd\ integer$$

5. The time between capture and launch operations of the lMMET

$$t_{lMMET(C\&L)} = \frac{T_{eMMET}}{n_5} \qquad n_5 = odd\ integer$$

6. Total Earth to Moon to Earth transfer time

$$t_{E\to M\to E} = n_6 T_{eMMET} \qquad n_6 = odd\ integer$$

It can be seen that, requirements 1 to 4 determine the periods of the MMET's both about their orbiting bodies and their CoM's. A constant value of the Moons' orbital period about the Earth was taken, as 27.45 days [10]. It can be seen that any alteration of the eMMET's period about the Earth affects requirements 2 to 6 and must be accommodated for by its corresponding integer.

Assessing the requirements set out above with the model represented in Section 10 shows that a compromise must be made to the requirements to allow for a feasible mission. For example the time between capture and launch operations of the lMMET will be taken to equal half of its rotational period about its CoM, so that payloads can arrive and leave the Moon within the same month. This will involve additional propulsion being carried by the payload capsule to correct its trajectory and velocity.

9 Earth to LEO Transfer

9.1 Rendezvous Trajectory Design

The upper tip of the eMMET captures the freight, propelled by lMMET, at the perigee of its orbit. For the requirement of simultaneous loading, the lower tip of the eMMET has to capture a payload, of equal mass, at the same time. In order to achieve this, the relative tangential velocity between the lower tip of the eMMET ($v_{tip,2,E}$) and the incoming payload (v_{pl}) has to be zero. This is represented in Figure 7.

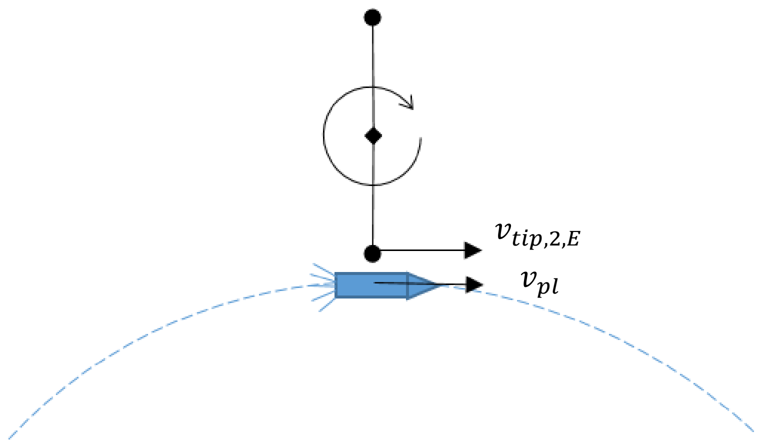

Figure 7: Relative velocity at capture

A reusable VTOL launch vehicle, such as the Falcon 9 rocket, would be used to transport the 1000kg freight, from the surface of the Earth, to the eMMET's lower tip. Therefore, the primary design requirement for the orbital trajectory, about the Earth, of the launch vehicle was to attain a tangential (orbital) velocity and altitude, equal to that of the eMMET's lower tip ($v_{pl} = v_{tip,2,E}$), at the point of the eMMET's perigee. Therefore, the values of these design parameters, as dictated by the orbit and length of the eMMET, were to be used in determining the size and shape of the orbit: length of semi-major axis, a, semi-minor axis, b, and the eccentricity, e:

$$a = \left(\frac{2}{r} - \frac{v^2}{\mu_E}\right)^{-1} \tag{12}$$

$$b = \sqrt{r_P * r_A} \tag{13}$$

$$e = \sqrt{\frac{r_A - r_P}{r_A + r_P}} \tag{14}$$

As the aim of this section of the mission architecture was only to deliver the freight at a specific velocity and altitude, there was no requirement for the launch vehicle to be in an orbit but rather: launch (L), rendezvous with eMMET (A) and return back to Earth without completing a full orbit. This type of orbit is known as a Sub Earth Orbit (SEO), where the orbital trajectory intersects the surface of the gravitating body that it orbits. As illustrated by Figure 8, this would be achieved by setting the rendezvous altitude, point A, as the apogee, with a perigee which lies below the surface of the Earth at point P to get the required velocity at point A. Note orbit 7 is the sub-earth orbit:

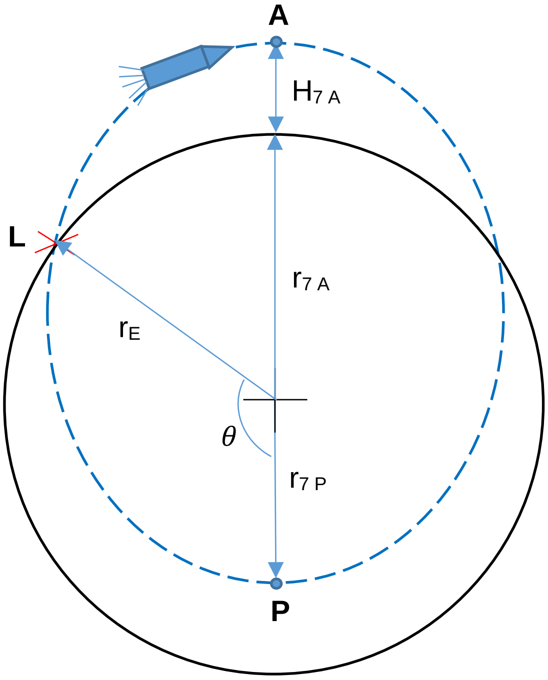

Figure 8: Launch vehicle's Sub-Earth Orbital trajectory

9.2 Arrival Synchronization at Capture

With the trajectory design in place, it was ensured that the launch vehicle would reach the point of rendezvous at a desired velocity and altitude. However, another critical requirement, for the rendezvous to take place, was the arrival timings. This would be achieved by calculating the time taken by a launch vehicle, from launch (L), to reach the perigee of the eMMET's orbit, or the apogee of the SEO (A), on the SEO trajectory. Since the position of the eMMET, in its Earth orbit, could be determined as a function of time, the launch can, therefore, be planned for initiation whenever the eMMET acquires the position which requires the same amount of time as the launch vehicle to reach the predetermined point of capture (A). Thus, synchronising the arrival timings at capture. As the time taken from L to A (t_{LA}) could not be determined directly, it was to be calculated by finding the difference between the time interval from P to A (t_{PA}) and time interval from P to L (t_{PL}).

The procedure in determining the position of a point in an elliptical orbit, as a function of time, is to: compute the mean anomaly, M; then using this value to iterate for the eccentric anomaly, E; which then enables the computation of the true anomaly, θ and hence, the orbital radius/position, r [14]. For the case of finding t_{PL}, however, this procedure was applied in reverse, as the positions (P, L and A) were known. The true anomaly (θ), measured from P as shown by Figure , was to be calculated first, by solving the polar equation of an ellipse (equation 15) for θ when r equals the radius of the Earth at L. Then, the eccentric anomaly (E) was to be found using the trigonometric identity, as presented by equation 16 [14]. This value of E was then to be used in the Kepler's equation, equation 17, to find the mean anomaly (M). Finally, the mean anomaly, together with the mean motion (n), defined by equation 18, were used to compute t_{PL} (equation 19).

$$r(\theta) = \frac{a(1-e^2)}{1+ecos(\theta)} \tag{15}$$

$$tan\left(\frac{\theta}{2}\right) = \sqrt{\frac{1+e}{1-e}}tan\left(\frac{E}{2}\right) \tag{16}$$

$$M = E - esinE \tag{17}$$

$$n = \sqrt{\frac{\mu}{a^3}} \tag{18}$$

$$M = nt_{PL} \tag{19}$$

$$T = 2\pi\sqrt{\frac{a^3}{\mu}} \tag{20}$$

Unlike the calculation for t_{PL}, t_{PA} was to be simply evaluated as half of the launch vehicle's orbital period (equation 20). Once the values of t_{PL} and t_{PA} are computed, then t_{LA} would be computed as the difference of these time intervals. The outcome of this computation would, then, suggest to initiate the launch when the eMMET acquires a position, in its orbit, which would take a period of time equivalent to t_{LA} to reach its point of perigee (or rendezvous).

9.3 **Optimal Launch Location Calculation**

Along with the design of the launch vehicle's trajectory, the orbital inclination is one of the fundamental elements that defines an orbit. The knowledge of the desired orbital inclination allows the determination of the optimal launch site, for the launch vehicle. The orbital inclination, i, dictates the north and south latitude bounds for all possible launch sites, from which the desired inclination can be achieved [15], as shown by Figure . In other words, if the latitude of the launch site (ϕ) is higher than the desired orbital inclination, then the orbit cannot be reached directly.

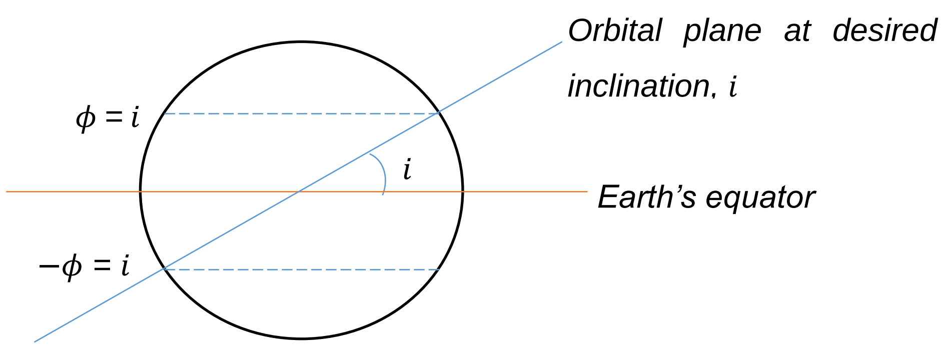

Figure 9: Possible launch sites latitudes, ϕ, for a given orbital inclination, i

Since the eMMET's orbit was chosen to be coplanar with that of the Moon's, the launch vehicle has to orbit at this same inclination for the primary purpose of rendezvous with the eMMET. However, the orbital inclination of the Moon, relative to the Earth's equator, varies between 18.28 and 28.58 degrees [16]. As the minimum orbital inclination equals the latitude of the launch site (or mathematically $i \geq \phi$), the inclination was chosen as 18.28 degrees. This would allow the determination of a launch site latitude lower than the minimum inclination of the Moon and as a result, all possible inclinations of the Moon (18.28-28.58 degrees) would be achieved from that launch site.

The orbital inclination and the latitude of the launch site are related by the inertial launch azimuth, β_i, as illustrated by equation 21 [17].

$$\cos(i) = \cos(\phi)\sin(\beta_i) \tag{21}$$

β_i defines the angle, measured eastwards, from due north to the projection of the desired orbital plane onto the launch site. This provides the direction of flight, at launch, from an inertial reference frame. Since, the Earth constantly spins about its axis of rotation, every point on the Earth's surface has an eastward tangential velocity (v_{earth}). This velocity varies with the latitude by $v_{earth\,equator}\cos\phi$ (where, $v_{earth\,equator}$ is the tangential velocity at the equator), with the maximum achieved at the equator due to the greatest circumference covered for a given time of rotation. Hence, this needs to be accounted for in determining the total velocity required by a launch vehicle (v_{rot}), at the expense of the Earth's rotation (i.e. rotational reference frame). This statement is schematically represented by Figure 10.

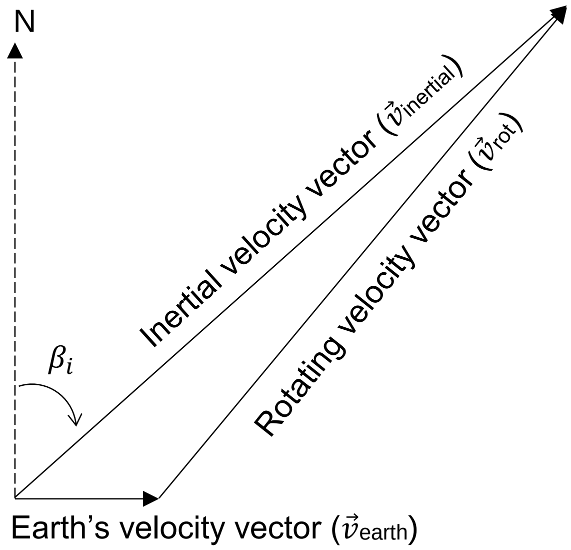

Figure 10: Rotating Launch azimuth

As a result, two possibilities were to be investigated: either launch near the Earth's equator for a greater eastward velocity of the Earth, at a decreased launch azimuth (less easterly launch) or launch near, but lower than, the latitude equal to the target orbital inclination (18.28°), for increased launch azimuth but lower eastward velocity of the Earth. Hence, the total velocity required by a launch vehicle from a rotating frame of reference (v_{rot}), for a target payload tangential velocity at rendezvous of 7.5695km/s (see Table 4), was calculated for various latitudes and the results are presented in table 1. Full tabulated values in Appendix. The values of the parameters were computed from the equations derived in [17].

Latitude (deg)	Inertial Azimuth (deg)	vrot (m/s)
0	71.72	7129.3629
5	72.39458627	7129.24766
10	74.61893933	7128.90543
15	79.42983182	7128.34657
18.28	90	7127.87018

Table 1 : Dependence of total velocity required with latitude

v_{rot} was found to decrease further from the equator and a maximum when $\phi = i$. This finding concluded that a more easterly launch, i.e. launch azimuth close to 90 degrees, was more favourable than a higher launch site inertial velocity. Therefore, the optimal launch site had to be at or near the latitude equal to the orbital inclination of 18.28

degrees. This was found to be the Satish Dhawan Space Centre, India [18], at a latitude of 13.7374° N.

10 Results

10.1 Tether System with Synchronisation

The Perigee altitude of the eMMET was chosen to be 300km so that the lower tip of the tether would extend down to an altitude of 250km. The orbit of the eMMET was originally a circular orbit about the Earth, however, the tangential velocity component of the eMMET was insufficient and required the tether to spin at an angular velocity which resulted in a stress greater than the material stregth. The solution to this was to change the orbit to an elipticle orbit by increasing the appogee radius. The appogee radius was incresed to 16000km, corresponding to an altitude of 9629km. This allowed the required angular velocity of the eMMET, and hence the stress, to be reduced to an acceptable level. Once the tether system was constructed the cross sectional area of the tethers was reduced from the original 6.283x10^{-5}m^2 where possible to reduce the mass of the tethers. The torque required to spin up each tether was determined and the time this would take. The key orbital and tether parameters of the eMMET and the lMMET are shown in table 2.

eMMET				lMMET			
Orbital		Tether		Orbital		Tether	
$\boldsymbol{H_{1P}}$	$300\ km$	M_{Pl}	$1000\ kg$	H_{4P}	$580\ km$	M_{Pl}	$1000\ kg$
$\boldsymbol{H_{1A}}$	$9629\ km$	$\dot{\psi}_e$	$0.030905\ rad/s$	H_{4A}	$580\ km$	$\dot{\psi}_l$	$0.000889\ rad/s$
$\boldsymbol{r_{1P}}$	$6671\ km$	L_e	$50\ km$	r_{4P}	$2317\ km$	L_l	$570\ km$
$\boldsymbol{r_{1A}}$	$16000\ km$	$v_{tip,e}$	$1.545\ km/s$	r_{4A}	$2317\ km$	$v_{tip,l}$	$0.5066\ km/s$
$\boldsymbol{v_{1P}}$	$9.184\ km/s$	$T_{tip,e}$	$203.306\ s$	v_{4P}	$1.455\ km/s$	$T_{tip,l}$	$7070\ s$
$\boldsymbol{v_{1A}}$	$3.829\ km/s$	A_e	$6.283\times10^{-5}\ m^2$	v_{4A}	$1.455\ km/s$	A_l	$3.00\times10^{-7}\ m^2$
$\boldsymbol{a_1}$	$11336\ km$	ρ_e	$970\ kg/m^3$	a_4	$2317\ km$	ρ_l	$970\ kg/m^3$
$\boldsymbol{e_1}$	0.4114	σ_e	$1.918\ GPa$	e_4	0	σ_l	$1.625\ GPa$
$\boldsymbol{T_{CoM,E}}$	$12011\ s$	SF	1.5	$T_{CoM,E}$	$10005.88\ s$	SF	1.5
		$SF.\sigma_e$	$2.877\ GPa$			$SF.\sigma_l$	$2.438\ GPa$
		τ	$25000000\ Nm$			τ	$625000000\ Nm$
		$t_{spin\ up}$	$24160\ s$			$t_{spin\ up}$	$3066\ s$

Table 2: Orbital and tether parameters of eMMET and lMMET

The stator had to provide an equal reaction torque to allow the tether to 'spin up'. The length of the stator tethers was made to be 1km less than the payload tethers so that interaction between the payload and ballast masses could be avoided. The angular velocity of the stator when the tether achieves its required angular velocity is shown in table 3 along with the component parameters.

Stator Parameters

eMMET		lMMET	
$\boldsymbol{\dot{\psi}_{S,e}}$	$0.032597\ rad/s$	$\dot{\psi}_{S,l}$	$0.000844\ rad/s$
$\boldsymbol{\rho_e}$	$970\ kg/m^3$	ρ_l	$970\ kg/m^3$
$\boldsymbol{M_B}$	$1000\ kg$	$M_{\boldsymbol{B}}$	$1000\ kg$
$\boldsymbol{A_{S,e}}$	$6.283\times10^{-5}\ m^2$	$A_{\boldsymbol{S},l}$	$3.00\times10^{-7}\ m^2$
$\boldsymbol{L_{S,e}}$	$49000\ m$	$L_{\boldsymbol{S},l}$	$569000\ m$
$\boldsymbol{\sigma_{S,e}}$	$2.07\ GPa$	$\sigma_{\boldsymbol{S},l}$	$1.46\ GPa$
$\boldsymbol{\sigma_{S,e}.1.5}$	$3.1\ GPa$	$\sigma_{\boldsymbol{S},l}.\mathbf{1.5}$	$2.19\ GPa$

Table 3: Stator rotational parameters

The upper and lower payload velocities at the eMMET and the lMMET, relative to the Earth and Moon respectively are shown in table 4.

Payload Velocity

eMMET		lMMET	
$\boldsymbol{v_{Pl,1,E}}$	$10.798\ km/s$	$\boldsymbol{v_{Pl,1,M}}$	$2.3195\ km/s$
$\boldsymbol{v_{Pl,2,E}}$	$7.5695\ km/s$	$\boldsymbol{v_{Pl,2,M}}$	$0.5904\ km/s$

Table 4: Payload tangential velocities at eMMET and lMMET

10.2 Tether System Without Synchronisation

The tether set up in Section 10.1 was altered so that it met the synchronisation requirements set out in Section 8. This meant that the apogee radius of the eMEMT had to increase and the tether length decrease. The biggest change occurred at the lMMET where the length of the tether was decrease by just less than 200km. This allowed the motor torque to be reduced, however, the altitude of the lower payload was raised by this. The payload would therefore have to reach the surface of the Moon from a greater distance and could reduce the accuracy of payload landing.

eMMET				lMMET			
Orbital		Tether		Orbital		Tether	
$\boldsymbol{H_{1P}}$	$300\ km$	M_{Pl}	$1000\ kg$	H_{4P}	$586.835\ km$	M_{Pl}	$1000\ kg$
$\boldsymbol{H_{1A}}$	$18914\ km$	$\dot{\psi}_e$	$0.022090\ rad/s$	H_{4A}	$586.835\ km$	$\dot{\psi}_l$	$0.001608\ rad/s$
$\boldsymbol{r_{1P}}$	$6671\ km$	L_e	$45.74\ km$	r_{4P}	$2323.835\ km$	L_l	$386.781\ km$
$\boldsymbol{r_{1A}}$	$25285\ km$	$v_{tip,e}$	$1.010 km/s$	r_{4A}	$2323.835\ km$	$v_{tip,l}$	$0.6218 km/s$
$\boldsymbol{v_{1P}}$	$9.724\ km/s$	$T_{tip,e}$	$284.439\ s$	v_{4P}	$1.453\ km/s$	$T_{tip,l}$	$3908\ s$
$\boldsymbol{v_{1A}}$	$2.565\ km/s$	A_e	$1.5\times10^{-5}\ m^2$	v_{4A}	$1.453\ km/s$	A_l	$5.00\times10^{-7}\ m^2$
$\boldsymbol{a_1}$	$15978\ km$	ρ_e	$970\ kg/m^3$	a_4	$2323.835\ km$	ρ_l	$970\ kg/m^3$
$\boldsymbol{e_1}$	0.5825	σ_e	$1.983\ GPa$	e_4	0	σ_l	$2.005\ GPa$
$\boldsymbol{T_{CoM,E}}$	$20100\ s$	SF	1.5	$T_{CoM,E}$	$10050.18\ s$	SF	1.5
		$SF.\sigma_e$	$2.975\ GPa$			$SF.\sigma_l$	$3.007\ GPa$
		τ	$25000000\ Nm$			τ	$350000000\ Nm$
		$t_{spin\ up}$	$9150\ s$			$t_{spin\ up}$	$3320\ s$

Table 5: Orbital and tether parameters of eMMET and lMMET (Synchronised)

The integers used to produce the results in Table 5 are shown in table 6.

$\boldsymbol{n_1}$	**118**
$\boldsymbol{n_2}$	2
$\boldsymbol{n_3}$	106
$\boldsymbol{n_4}$	9
$\boldsymbol{n_5}$	7
$\boldsymbol{n_6}$	57

Table 6: Integer values

The angular velocity of the stator when the tether achieves its required angular velocity is shown in table 7 along with the component parameters.

Stator Parameters

eMMET		lMMET	
$\dot{\boldsymbol{\psi}}_{\boldsymbol{S,e}}$	$0.023154\ rad/s$	$\dot{\psi}_{S,l}$	$0.001543\ rad/s$
$\boldsymbol{\rho_e}$	$970\ kg/m^3$	ρ_l	$970\ kg/m^3$
$\boldsymbol{M_B}$	$1000\ kg$	$M_{\boldsymbol{B}}$	$1000\ kg$
$\boldsymbol{A_{S,e}}$	$1.50\times10^{-5}\ m^2$	$A_{S,l}$	$5.50\times10^{-7}\ m^2$
$\boldsymbol{L_{S,e}}$	$44742\ m$	$L_{S,l}$	$385781\ m$
$\boldsymbol{\sigma_{S,e}}$	$2.11\ GPa$	$\sigma_{S,l}$	$1.84\ GPa$
$\boldsymbol{\sigma_{S,e}.1.5}$	$3.17\ GPa$	$\sigma_{S,l}.\mathbf{1.5}$	$2.76\ GPa$

Table 7: Stator rotational parameters (Synchronised)

The upper and lower payload velocities at the eMMET and the lMMET, for the synchronised system, relative to the Earth and Moon respectively are shown in table 8.

Payload Velocity

eMMET		lMMET	
$\boldsymbol{v_{Pl,1,E}}$	$10.734\ km/s$	$\boldsymbol{v_{Pl,1,M}}$	$2.316\ km/s$
$\boldsymbol{v_{Pl,2,E}}$	$8.714\ km/s$	$\boldsymbol{v_{Pl,2,M}}$	$0.589\ km/s$

Table 8: Payload tangential velocities at eMMET and lMMET (Synchronised)

11 LOGISTICS AND SYSTEM PERFORMANCE

When designing a system of such complexity it is inevitable that the scope of the work will involve issues much greater than the design of the tether transport system. The feasibility and sustainability of such a system must also be considered with attention paid, first and foremost, to safety and maintenance, in addition to the economic and environmental implications. In this section, several logistical difficulties and their potential conceptual solutions will be detailed, along with a costing exercise run in conjunction with the mission design.

11.1 Power Supply

A key difficulty facing any space based structure is how it will be powered, whether it be in terms of providing propulsion or simply powering the electrical systems required for everyday functions. Obviously, a constant supply of electricity is much harder to come by in space than on Earth, however space does have one key advantage, a wealth of solar energy. Solar energy is a very convenient, and in fact the only currently viable method of electricity generation in space. When talking in terms of powering a large gear motor to provide the torque required there is no other source of energy that suits the purpose. Similar to the ISS [19], it is anticipated that both the eMMET and lMMET will be powered by a large array of photovoltaic (PV) panels to power both the gear motor and the electrical systems of the Control Module. It is, however, important to remember that a satellite orbiting the Earth in LEO and the Moon will at times be obstructed from sunlight by the celestial body it orbits. Therefore, it would be pertinent to have a means by which energy produced, whilst in sunlight, could be stored and utilised in order to maintain a constant supply of power when out of sunlight. To achieve this, large Nickel-Hydrogen batteries could be used, similar to those in place on the ISS [20] thus supplying constant electrical power to the gear motor and electrical systems. Of course, these panels degrade over time particularly in the harsh conditions of space so they would need to be maintained and replaced periodically.

11.2 Safety and Infrastructure

Clearly a large structure over 100km in diameter orbiting the Earth presents a great deal of infrastructural difficulties, especially given the number of objects already in orbit around the Earth. There are millions of pieces of debris within the Earth’s orbit that range from large trackable satellites and objects to tiny particles such as flecks of paint. Due to the high velocities of these particle, even a small fleck of paint can damage a spacecraft [21]. The strength of the tether is, thus, of critical importance as the system is desired to be reusable and be able to operate continuously. Not only is the tensile strength of the tether important but also its durability. For the Earth-Moon tether system to be cost effective it must operate without crewed maintenance for a reasonable time period. This means that the tether must withstand orbital strikes from debris in the Earth's orbit. While the tethers are very thin they are still extremely long which creates a large area for a strike to take place. The solution to this is to build in redundancies in the form of multiple strands within the tether. A tether such as this has been developed by Tethers Unlimited™ called The Hoytether™ [8]. The tether consists of multiple lines in a tubular structure. There are two types of line within the structure; primary lines and secondary lines. The primary lines of the structure carry the tensile stress when the tether is first deployed and undamaged. However, if the tether is struck by a piece of debris which severs the primary lines then the secondary lines support the forces within the tether. This can be seen in Figure 11 [8].

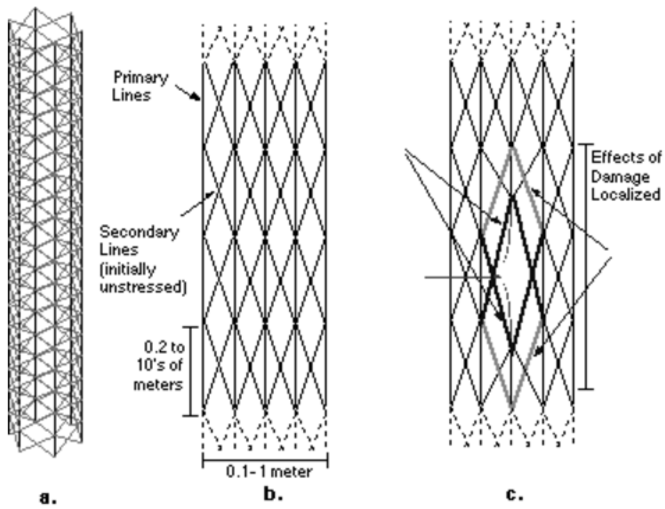

Figure 11: The Hoytether™ structure [8]

The Hoytether™ is useful for counteracting strikes by small objects, the exact size will vary depending on the chosen size specifications. However, for sufficiently large objects, both primary and secondary lines of the tether would be severed simultaneously causing catastrophic failure of the tether. Fortunately, objects of this size are tracked and catalogued by the United States Space Surveillance Network [21], additionally NASA has a set of guidelines in place for the avoidance of such collisions. They imagine a protected space around their vehicles into which large space debris should ideally not enter. They then estimate the probability of a collision and if it exceeds 1 in 100000 they are likely to take evasive action [21] by means of a small “debris avoidance manoeuvre”. It would be pertinent that anyone operating a tether system such as that proposed in this mission architecture would have access to this catalogue of space debris. It is, however, unclear how the tether system would react to any such invasion of its protected space since any manoeuvre would affect the rotational dynamics of the system. This is an area which would require significant research to establish the best means by which to ‘move’ the tether system, something which would also be required for station-keeping.

In order to remain in LEO, the eMMET would have to maintain its orbit via station-keeping. This is because the drag acting upon it due to air molecules will cause orbital decay, for example the ISS loses around 90 metres of altitude each day [22]. Periodic boosts would be required to keep the eMMET at an acceptable altitude which, while relatively easy for traditional satellites, would be very difficult to achieve successfully on a rotating tether system. Station-keeping manoeuvres typically involve chemically boosting the station to a higher altitude however other methods are being explored in depth.

At present the lMMET is unlikely to suffer the same space debris issues as the eMMET due to the moons lack of natural satellites making collisions distinctly unlikely. If a lunar base were to be established then an increased volume of man-made objects orbiting the moon may lead to increased space debris in lunar orbit in the future.

11.3 Guidance, Navigation and Control (GNC) and Telecommunications

A space system as complex as the architecture proposed for this mission would require the most advanced navigation and control systems available. In terms of capability it must be capable of coordinating the transfer of a large payload between two celestial bodies with extreme accuracy and very little margin for error. Additionally, in order to meet the bi-directional requirements of the mission architecture the system must be able to coordinate a transfer at both ends of the tether at precisely the same time, so that the tether maintains orbital stability. The number of moving parts and potential errors is great so constant communication between each station would be required. The stations in this architecture would be defined as:

- An Earth base with overriding control of operations capable of handling and carrying out complex trajectory alterations.
- The eMMET which ideally should not move position out with station-keeping but should serve as a point from which to relay and boost signals onwards. It must however be able to accurately transmit its position and determine the trajectory of incoming payloads.
- The payloads which should each be capable of receiving and carrying out course corrections based on data received from Earth base, in order to ensure successful rendezvous.
- The lMMET which should perform the same functions as the eMMET.
- An assumed basic lunar base with a capability to rendezvous with the lMMET to coordinate return of payload to Earth.

The precision required to coordinate the payload handovers, particularly at the lMMET end, would require very fast and efficient data transfer due to the 1.3 second delay brought about by the time it takes for the signal to travel to the Moon. The targeting accuracy ultimately required is beyond anything which has been attempted previous therefore significant investment in R&D and testing would likely be required. A solution to this is not currently available to hand, however, it is clear that only the most advanced GNC systems would be capable of accomplishing the payload handovers.

12 Cost Analysis

Initially, it was intended to run a full costing exercise in conjunction with the construction of the mission architecture to allow for comparison with the cost for conventional rocketry to carry out the same task i.e. a bi-directional transfer of 1000kg payloads. However, as the work progressed it became apparent that within the timescale and given the extremely speculative numbers involved when discussing the economics of space travel, this would not be possible. Additionally, the work involved could plausibly constitute an area of research in itself and in the end may be useless regardless since a great deal of the costs are impossible to estimate as they have never been done before, for example, carrying out construction in orbit around the Moon.

Instead, it was decided to give a breakdown of some of the estimated capital costs involved in the setup of the tether system.

12.1 Launch of eMMET and lMMET

This section covers only the costs of lifting the required parts into a Low Earth Orbit using the SpaceX Falcon 9 rocket. SpaceX advertise as charging 4640$/kg to LEO assuming their rockets are fully reusable [2]. This is obviously very speculative, however, it is not unreasonable to assume that by the time of implementation this technology would exist. Therefore, the total mass of the eMMET system must be estimated. The control module which would house the gear motor and power supply was estimated to be around 5mT, which would fall into the category of a large satellite as defined by the FAA [22]. The tether mass was estimated simply by considering the length, cross-sectional area and density of one tether length which could then be multiplied by a factor of 2 for a symmetric system. Therefore, the tether mass was found to be:

$$mass\ of\ one\ length = \rho * A_c * L$$

Where the length, L, of one tether arm for the eMMET was 50km, the cross-sectional area, A_c, was $6.283e^{-5}m^2$ and the density was $970kg/m^3$ for the selected material Spectra 2000^{TM} [5]. Therefore:

$$mass\ of\ one\ length = 970 * 6.283e^{5} * 50000$$

$$mass\ of\ one\ length = 3046.77kg$$

So total eMMET tether mass was seen to be **6093.54kg.**

Next, the ballast masses required to provide reaction torque, and their tethers, were factored in which were seen to be 3987kg as calculated above for each tether giving a total ballast mass of 7973kg. Finally the mass of the grapple mechanism was estimated as roughly 10% of the payload mass which it was required to handle i.e. 1000kg so therefore the grapple mass required to be lifted was 200kg in total.

These masses together resulted in a total eMMET mass of 19166.54kg which when multiplied by the cost per kg to LEO gave us an estimate of launch cost of the eMMET which was $89 million.

The cost to launch the lMMET to lunar orbit was estimated similarly by calculating the total expected mass of the system. The cost to launch 1kg of mass to the Moon is difficult to estimate in real terms however the SpaceX Falcon 9 rocket used to transport the eMMET should soon be capable of reaching the Moon and in fact claims that it could transport 4020kg to Mars [2]. Given SpaceX charges around $62 million for a full launch to GTO, or an altitude of around 42000km, a reasonable estimate to transport the given 4020kg into a LTO was around 5 times this price i.e. $310 million, when risk and additional fuel requirements are factored in. The lMMET would require two such transports leading to a launch cost of around $8 billion Estimates of the mass of the lMMET system are given below in Table 9 alongside estimates for the eMMET for comparison:

Component	Mass (eMMET)	Mass (lMMET)
Control Module	5000kg	5000kg
Tether Arms	6093.54kg	165.87kg
Ballasts	7973kg	2331.2kg
Grapple Mechanism	200kg	200kg
Total System Mass	**19266.54kg**	**7697.07kg**

Table 9 - Component Weights

The lMMET tether mass was calculated using the same method as the eMMET with density $970kg/m^3$, length 570km and cross sectional area $3e^{-7}m^2$.

12.2 Material Costs

The cost of the components which make up the MMET system are difficult to estimate in part since it is hard to say exactly what systems will be required to be in place without in depth knowledge of a wide variety of areas including space communication and construction of space systems. The cost of the physical tethers can be estimated by multiplying the total mass of both tethers and ballast tethers by the cost per kg of Spectra 2000™ which CES Edupack gives as approximately $120/kg giving an approximate material cost of $2 million. However this is simply the cost of the raw material and does not account costs resulting from the construction of the Hoytether™ structure which is simply impossible to realistically estimate.

Next we must consider the cost of each control module in turn. The eMMET would be the cheapest to construct since it could be launched as a single unit into LEO. The cost to construct and manufacture this eMMET is again speculatively estimated based upon satellite costs. GlobalCom gave the price of one of their satellites as at least $300 million for a simple weather monitoring satellite [24]. Given the complex and new technology required such as the telecommunication systems, not to mention the large gear motor required, the simple material cost of the eMMET could realistically range from $500 million to $1 billion. The lMMET would likely be a similar cost however extra systems would have to be in place to allow for construction of the tether since it could not be sent into LTO as a single unit.

12.3 Additional Costs for Consideration

Outlined in the section above are some basic costs which would be incurred if the mission architecture described here were to be implemented. These are highly speculative and subject to a great deal of fluctuation however they do give an idea of the economic price of putting an MMET structure in place for bi-directional payload transport. Truly estimating the cost of such a system is not possible to a realistic degree of accuracy and therefore is not attempted. Additional costs which would have to be included are:

- Research and Development
- Man hour costs to build and implement
- Maintenance and Resupply
- Fuel requirements for station-keeping
- Cost of bandwidth to maintain telecommunications
- Cost of establishing a Lunar base capable of sending return payloads
- Training costs for astronauts required to maintain the system
- Everyday costs of running an interplanetary payload transfer system

Most of these costs are well out with the scope of this mission architecture, however a reasonable estimate of capital setup costs would likely run anywhere from $15-20

Billion before running costs are considered. The price of pushing the boundaries of space travel has always been high and the mission proposed here is no exception. The high capital investment would be offset by greatly reduced running costs compared to conventional rocketry, however any attempt to directly compare the two methods would be far too unrealistic. Ultimately economic feasibility projections would be down to the company or government which wished to implement the system. Certainly it is technically feasible and with further work building on the architecture proposed here it could be made to work more efficiently.

Finally for reference Astrobotics, a NASA sub-contractor, estimates that it would levy a charge of $1.2 million per kilogram to transport payload to the Moon utilising solely conventional rocketry [6]. This price does not include a return journey.

Were this system to be implemented effectively, it would be capable of transporting a 1000kg payload to the moon and one back every month. With proper maintenance, a system lifetime of 25 years is very reasonable. Therefore, assuming each 1000kg payload contained even 265kg, the maximum payload Astrobotics is capable of carrying per journey [6], of useful mass we could estimate the return on investment as follows:

$$charge\ per\ payload = \max payload \times Astrobotic\ charge\ levied$$

$$charge\ per\ payload = 265 \times 1.2e^{6} = \$318\ million$$

$$Sales\ per\ year = charge\ per\ payload \times payloads\ per\ year$$

$$Sales\ per\ year = 318e^{6} \times 12 = \$3.816\ Billion$$

From this sales over the lifetime of the system could be estimated:

$$System\ Lifetime\ Sales = Sales\ per\ year \times reasonable\ estimated\ lifetime$$

$$System\ Lifetime\ Sales = 3.816e^{9} \times 25 = \$95.4\ Billion$$

This is clearly a very speculative and basic example of how such a system would be feasible, and of course operating costs/cost of sales as well as insurance would have to be deducted from this figure. However, it does serve to highlight that if the demand were there for such a system it could prove, if not lucrative, then at the very least to a viable alternative to conventional rocketry and certainly much less wasteful.

13 Conclusions

To conclude, a complete mission architecture was successfully constructed for the bi-directional transfer of 1000kg of payload from Earth to the Moon. A reusable VTOL rocketry system, such as SpaceX's Falcon 9 rocket, was selected to transport the payload to and from the Earth's surface to an MMET in Low Earth Orbit. The trajectory and timings of this transfer were specified along with an Earth launch site at Satish Dhawan Space Centre in India. The payload was then accelerated by the eMMET so that it could be projected into a Lunar Transfer Orbit with its apogee within the lunar sphere of influence. At which point a Lunar based tether, or Lunavator, would match the velocity of the payload in order to catch it and transport it to the lunar surface. It should be noted that the Lunavator concept was chosen over a powered descent due to the inherent bi-directional capabilities of the system.

The Earth and Moon tether system had to be synchronised so that the bi-directional element of the mission architecture could be satisfied. Unfortunately the work, at this point, met with a fundamental problem with the Lunavator concept in that the velocity disparity as the tether approached the lunar surface was 0.5904km/s when it would ideally be ~0m/s. This could not be achieved realistically with the system proposed without transferring the payload on a hyperbolic orbit path, or making the tether system unfeasibly long, neither of which were desirable.

Ultimately the velocity disparity of 0.5904km/s would have to be managed by a chemical propellant burn to achieve a delta V of around 0.5904km/s after which the payload could be caught by an appropriate grappling system. This mission could, as stated, be run in reverse since the Lunavator is capable of releasing payloads on an Earth transfer orbit. Again however, a chemical boost would be required to match the payload velocity with that of the tether arm when transferring from Lunar surface to a Lunar orbit.

Finally, conceptual design solutions were provided for logistical and system performance concerns such as, power supply, safety and infrastructure, GNC and telecommunications. Moreover, a basic cost analysis was run throughout the architecture design to give an idea of the expense required to implement such a system. Unfortunately a full comparison with the same payload transfer utilising conventional rocketry could not be performed to an adequate level within the time constraints of the project due to its complex and speculative nature.

14 Future Work

There are a number of areas within this project which could constitute an area of work alone, for example, designing the gear motor which would provide the system with the required torque. This section, however, will cover the areas of this work which it was felt required a greater deal of focus or were perhaps, not fulfilled to the standard expected at the conception of the project.

One of the greatest benefits of the proposed tether system would be its ability to operate with little to no chemical propellant use. Unfortunately the 0.5904km/s velocity disparity at the Lunar surface required, a not insignificant, use of propellant for ascent and descent. It should be expected that, were further work to be put into addressing this problem, that the velocity disparity could be reduced to a much more acceptable value. One solution could be a Lunavator design put forth by Hoyt [8] who theorised a single arm tether capable of accelerating payload. Hoyt proposed to make the tether do the work, so to speak, using only electricity generated from solar energy. The tether system would be composed of one single long tether with a counter balance mass (CBM) at one end with a central facility located between the CBM and the tether tip which was capable of ‘climbing’ the tether in either direction. Initially the facility would be located close to the centre of the tether, hence the centre of mass would be located between it and the CBM around which the system would rotate until the payload was captured at perilune. At this point the facility would utilise captured solar energy, or an alternative power source, to move ‘up’ the tether towards the CBM allowing the centre of mass to remain at the same altitude and thereby prevent any destabilisation of the system’s orbit. Additionally, since the distance from the payload to the centre of mass increases and the facility mass moves closer to the CBM then by the conservation of angular momentum the angular velocity of the payload will increase to match the required speed for zero relative velocity with the lunar surface [15]. This process is outlined below in Figure 12:

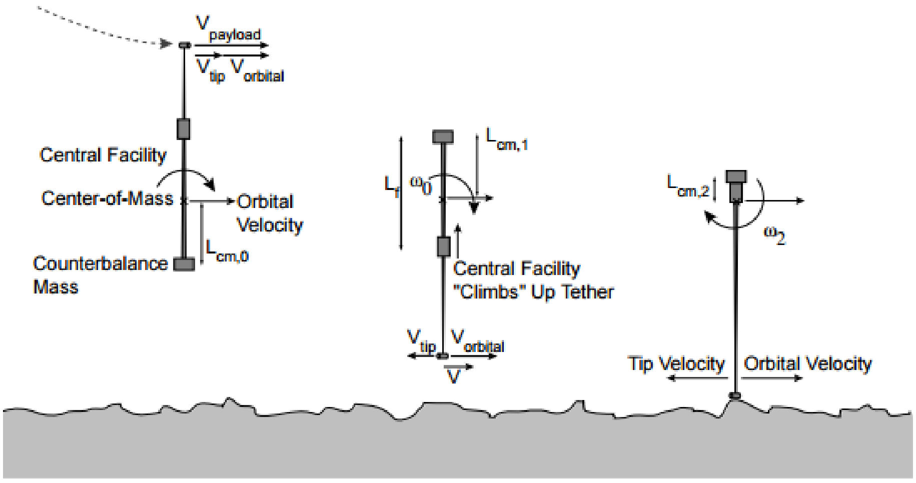

Figure 12- Hoyt's Lunavator tether process [8]

This solution could affect the bi-directional capabilities of the tether, since the catch and release operations are non-simultaneous, however it does show promise and a more in depth analysis could prove fruitful.

Throughout the project it became apparent that a cost analysis and comparison to the extent outlined in the project brief would not be possible. Establishing the economic cost of a mission architecture such as this is a huge undertaking and requires access to data far beyond the reach or scope of the project team. In addition a comparison with current conventional rocketry is difficult not only because the MMET transfer system is extremely hypothetical but because there has been no large scale excursions to the Moon since the Apollo program ended in 1972. This presents a problem in its own right since the advancement of technology means a direct comparison cannot be made with these missions. Additionally the aims of the mission proposed within this report vary greatly from that of the Apollo and so have different constraints.

Obviously any enterprise considering implementing such a system would have to carry out its own economic assessment to determine feasibility. Within the time constraints of this work, however, it was decided that only a basic analysis of some of the costs involved would be possible. In future this area could represent an area of study in itself utilising the data outlined herein.

Project Management Report

15 Project Management

15.1 Alterations to Original Project Objectives –*Project Definition*

In order for the project to be considered successful it must, first and foremost, meet the requirements outlined in the initial statement of purpose agreed with the client, in this case the project supervisor Professor Matthew Cartmell. This agreement states the expected quality, technical achievement and complexity of the work along with the timescale in which the work is expected to be achieved. The initial project guidelines specified the work for 4 students not 3 and given the scale of the architecture to be attempted it was clear that certain parts could reasonably constitute their own area of study for a similar report. Thus, the deliverables outlined were prioritised in terms of importance following a discussion with the client. Listed below are the initial project deliverables as specified in the statement of purpose:

1. To develop a comprehensive understanding of existing information on the subject. This will be presented in the form of a literature review within the interim report.
2. Development of a practical and sustainable method for Earth to LEO freight transport. The final method will be agreed upon between group and supervisor.
3. Development of a practical and sustainable method for lunar orbit to lunar surface freight transport. The final method will be agreed upon between group and supervisor.
4. Technical details of the orbital exchange path of the freight between eMMET and lMMET. This will be presented to supervisor at an intermediate phase of the project.
5. Discussion of the benefits of the existence of a tether system.
6. Analysis of the logistics that would need to be in place for the system to function once in operation.
7. Conceptual design solutions for power supply, Guidance, Navigation and Control (GNC), telecommunications and infrastructural and safety concerns.
8. An estimation of the cost and a comparison to alternative methods to measure the benefits of the tether system.

9. An estimation of the potential environmental impact the tether system would have.
10. Final product - A report presenting the final conclusions on the feasibility and effectiveness of an Earth to Moon tether transport system.

Each deliverable could be further broken down however this will be covered later in the report. In terms of prioritisation the complete mission architecture was deemed the most important facet of the project with particular attention paid to the orbital transfer of payload from the eMMET to the lMMET. Additionally, the Earth to LEO transfer was required to be completed at a level of accuracy not attempted previously. The benefits of this system compared to the same mission carried out by conventional rocketry were to be thoroughly examined however a full comparison may prove difficult to achieve. The cost analysis, intended to be run in conjunction with the architecture, was

given lesser importance since the work involved would require a project in its own right to be successful. Additionally, the conceptual design solutions presented in point 7 above were specified to be of lesser importance than achieving a full mission architecture.

15.2 Roles and Responsibilities

15.2.1 Personality Test

Since none of the group had worked together before it was decided to hold off on assigning project management roles whilst the group got to know each other and each members varying styles of work. Subsequently after a number of weeks it was decided to perform personality tests in the form of Belbin's Self-Perception Test. The test involved each member completing a questionnaire with points being allocated based on how they answered. From these answers the Belbin theory determined which roles a person was best suited for, based on their creativity, leadership characteristics etc. The results of the test are shown below in Table 10:

Role	Company Worker	Chairman	Shaper	Plant	Resource Investigator	Moniter Evaluator	Team Worker	Completer Finisher
Abhishek Thapa	Very High	Average	Average	High	Low	Low	Low	Average
David McDiarmid	Average	Average	High	Average	Low	Very High	Low	Very High
Ross Macdonald	Low	High	Very High	Low	Low	High	Average	High

Table 10 - Belbin's Test Results

Belbin's theory states that teams tend to perform better with a good balance of roles and styles of work as opposed to an 'Apollo' team of all highly intelligent members. For example, a good mix of introverts and extroverts or an equal balance of creative and practical workers. As can be seen above the group was fairly well mixed balanced in terms of how they tended to work, however, it must be noted than Belbin's theory is normally applied to larger teams. Since the group only consisted of 3 members it meant the group was not as well rounded as would have been ideal, for example all group members scored low on Resource Investigator. A resource investigator is typically the most sociable and likeable group member who serves to gel the team together to work for a common goal. This test not only highlighted suitable management roles but also areas in which the team may lack expertise and, thus, must pay special attention to.

15.3 Selection of Roles

15.3.1 Project Management Roles

It was important, when selecting project management roles, to not base decisions solely on the results of the personality test and instead compare the results to observations each group member had made of the others. Ultimately, following discussion at a group meeting the Project Management roles were decided as shown in Table 11 below

Team Member	Role	Job Description
Ross Macdonald	Project Manager /Task Allocation	Person in overall charge of the project, responsible for allocating tasks and ensuring they are completed in line with project specifications. Also responsible for team synergy and conflict resolution.
David MacDiarmid	Communications Manager	Responsible for ensuring meetings are regular and that each part of the project linked effectively to the mission architecture. Also responsible for making each groups members work available to the other via group online storage.
Abhishek Thapa	Bookkeeper	Responsible for keeping regular meeting notes and ensuring tasks were carried out by the deadlines set by the project manager. This involved creation of a Gantt chart to monitor progress.

Table 11 - Team Member Roles

It is clear that given the nature and the scope of the project that a traditional 'Project Manager' role was not the most suitable setup. Instead some of the traditional roles of a project manager, such as ensuring that deadlines were met, were delegated to each group member. This was done primarily to ensure that each member had a role in keeping the project moving efficiently and so reduce time wasted. Since each part of the project had links to at least one other area it was crucial that each member of the group self-managed, in a manner, by critiquing the work of the others and how it related to their section. This is explained further in the selection of technical roles.

15.3.2 Selection of Technical Team Roles

Suitable selection of technical roles within the project was equally as important as project management roles. In the early weeks of the project before roles were defined the team tasked themselves with reading the literature associated with not only

MMET's but the history of space travel to LEO and beyond. Each member was to collate the information they had collected and present it to the group in an informal style. This step was carried out after the selection of group management roles and its purpose was to determine which area of the project each team member favoured and so assign initial tasks. The project split relatively simply into 3 sections:

- Earth to LEO payload transfer – Abhishek Thapa
- Earth based tether to Lunar Orbit transfer – David MacDiarmid
- Lunar Orbit to Lunar Surface transfer – Ross Macdonald

Each team member was allocated an area of focus, Abhishek was assigned to Earth to LEO transfer since he had previously studied this topic in his work with High-Altitude Balloon (HAB) technology previously and David was assigned to the orbital transfer based upon the fact he was studying a similar subject in a Spaceflight Mechanics class. In reality the areas would overlap to a large extent meaning each member would have some involvement in each technical area of the project. However, for the initial work required, this delegation was considered the most efficient method of encompassing the entire scope of the project. Ultimately, each person was in charge of their section of the transfer and so any changes to the mission architecture involving two sections had to be cleared by both members before being carried out.

15.4 Gantt chart Schedule

A work breakdown structure (WBS) was utilised after the submission of the statement of purpose, as shown on the left side of Appendix 18.2 (cost analysis and report write-up omitted to show critical path of the technical mission architecture design). This change from the initial project planning aided in better organising the tasks and more importantly, according to the project objectives to be met (defined as deliverables in the statement of purpose). This was particularly effective, as each member had a sound understanding of the specific tasks and sub-tasks that were required to meet each of these objectives/deliverables. The deliverables were marked as milestones in the Gantt chart to emphasize its importance visually and hence, avoid any detraction from the overarching goals of the project. As the requirements of the tasks became clearer, the WBS structure was further refined and hence, made the allocation of human and time resource, to each task, much easier.

The critical tasks and the critical path was determined for the design of the mission architecture only, as cost analysis and report write-ups were largely recurring activities which ran in parallel over the course of the project. As shown by Appendix 18.2, the critical path originated from the LEO to LLO transfer stage activities. This highlighted the importance of this transfer stage of the mission architecture in the completion of the project and also brought to mind that only one person was initially assigned for this critical part of the project. As it was clear from the Gantt chart that there were not any major tasks running over the time period allocated for the LEO to LLO stage (only recurring cost analysis activities, which stem from the LEO to LLO stage), the remaining two members also took up some of the sub-tasks, which the person originally assigned coordinated. As a result of this resource management, the initially set deadline for the whole LEO to LLO transfer stage was met successfully (Appendix 18.3). This was also particularly effective, as the bulk of the learning from the project subject was obtained from this transfer stage.

The Gantt chart was also used to look at the effects of pushing back and splitting the activities. For example, it was desired to allocate time for the first semester exams and for job application (which was not initially taken into account). From the Gantt chart (Appendix 18.2), two critical tasks were found to lie in this time period (Detailed analysis of Earth and Moon orbits). However, the team realised that the time allocated for these tasks, initially, were an overestimation and the decision to look at simplifying the model meant that the task could be split and not pushed back (Appendix 2). This meant there was some time after the ‘break’ to look at simplifying the model.

The progress of the project was monitored at regular intervals, by the use of progress lines (Appendices 18.2 & 18.3). This gave an indication of which tasks were lagging behind, so that it could be focused on and which were ahead, so the time could be effectively distributed to other tasks.

15.5 **Meetings and Minutes**

Project meetings between group members were not set at regular intervals e.g. every Monday, however from the initial meeting a date for the next was always agreed based upon the work being carried out in the interim and the need for a physical meeting. This date was generally proposed by the communications manager. These meetings featured a progress update from each group member after which the other members would highlight any areas in which the work impacted theirs. The meetings served as deadline days for particular tasks to be carried out as well as serving to establish the progress of the project as a whole and the work still required to be done. It was then the project manager’s role to allocate tasks to be completed by the next meeting, however, these more often than not were usually decided by general group consensus. The bookkeeper was responsible for ensuring minutes were taken at each meeting, an abbreviated example of standard meeting minutes is shown in Appendices 18.4 and 18.5. Meeting were also held by the group with Professor Cartmell on a semi-regular basis, although these meetings tended to be more formal with rigid agendas. These meetings served to communicate to the client how the project was progressing and, if required, re-establish the scope of the project as previously mentioned. These changes to expected deliverables were usually brought about by time constraints rather than complexity.

15.6 Risk management

A 5x5 risk matrix was used to evaluate potential risks and their impact. These risks involved curtail aspects of the project outline not being completed. These risks were chosen from the WBS which showed the task dependence and interaction with other tasks. Also included within the risk matrix was team member absence. If any risks were found to be high or severe risk action would have been taken to either reduce the likelihood of it occurring or its impact, so as to reduce the risk. The risk matrix can be seen in table 12.

Aspects that could cause the project to overrun included the loss of data. This was reduced greatly by backing up important files with the use of a Google Drive folder, and personal backups.

Impact / Likelihood	Insignificant	Minor	Moderate	Major	Severe
Almost Certain					
Likely		1 week absence of member			
Possible					
Unlikely				>1 weeks absence of member	>3 weeks absence of member
Very Unlikely				Loss of data that would impact project	

Table 12 - Risk Management Matrix

Risk Significance	Colour
Low	
Medium	
High	
Severe	

16 Group Working

16.1 Reporting Lines – Group Discussion of small group size affecting roles

Due to the small group size, the reporting lines were less hierarchical and more level and cooperative. Each member had an equal standing and expertise for conducting a group project. However, the benefit of discretising the project organisation was clear. Therefore, instead of a leader in the original sense where the leader would have the final say on a matter, conflicts were resolved objectively. This allowed the structure of the project to be managed effectively.

The main reporting lines within the group were, primarily, determined by the technical area each member was responsible for, set out in section 15.3.2. With the large work involved in the project this allowed each group member to have a more detailed understanding of that specific technical area. This meant that each section could be reviewed in greater detail resulting in the optimum project result with the available resources and time.

Ultimately the body of work was to be completed for the client, Professor Carmell, to their desired specifications. Any potential conflicts that could not be resolved with the group would be taken to Professor Cartmell, who would make the final decision. Although advice was sought, all conflicts were resolved between the group members. The method utilised for conflict resolution can be seen in the next section.

16.2 **Conflict Resolution – Group**

Conflicts within groups can arise for a variety of reasons. Most notably a disagreement of the best way to proceed with the project or best way to tackle a certain task can arise easily and frequently. Different members within the group will have different reasons behind their ideas. To tackle such a conflict, should it arise, a controlled convergence matrix would be used to resolve the conflict. The controlled convergence matrix would allow potentially subjective views to be assessed so that an objective result could be reached according to set desirable criteria.

An example of this was when the method for transferring payload from the Earth's surface to the LEO had to be decided. A controlled convergence matrix was constructed with the primary criteria listed down the left hand side, according to the project objectives. The potential concepts are listed along the top and it was determined if they met the primary criteria. This can be seen in table 13:

Table 13: Controlled Convergence Matrix, Primary Criteria and Concepts

Concept / Primary Criteria	Conventional Rocketry	Conventional Rocket-integration. Balloon?	Conventional Rocket-integration. Plane	Non-conventional/ Reusable Rocketry	Single Stage to Orbit Space Plane
Availability by 2027?	Y	Y	Y	Y	N
Feasible trajectory design?	Y	Y	Y	Y	Y
Min Payload of 1000kg?	Y	N	Y	Y	Y
Primary Criteria Met	**Y**	**Y**	**Y**	**Y**	**N**

Only the concepts that met all primary criteria could be used within the mission architecture and these can be seen from table 13. The concepts that met the primary criteria were put into a second controlled convergence matrix to determine the best option. Although, the concepts that failed could not be used with the final mission architecture, they were inserted into the second controlled convergence matrix to provide a comparison and to determine if it had passed the primary criteria where they would rank when compared to the other concepts. The result of this is particularly interesting when the space plane is considered. The only primary criteria that the space plane failed was its availability at the planned time of the mission architecture. However, if this was not a factor the space plane ranks above all of the other concepts. The ranked controlled convergence matrix can be seen in table 14.

Table 14

Concept / Criteria	Conventional Rocketry	Conventional Rocket-integration. Plane	Non-conventional/ Reusable Rocketry	*Single Stage to Orbit Space Plane*	*Conventional Rocket-integration. Balloon?*
Trajectory simplicity	DATUM	-	S	+	-
Cost		S	+	+	+
Reusability		S	+	+	S
Weather dependency		+	S	+	-
Potential Impact of launch abortion		-	S	S	-
Maximum Payload Size		-	S	-	-
Available Launch sites		+	S	S	+
∑+		2	2	4	2
∑-		3	0	1	4
∑S		2	4	2	1
Total Score	0	-1	2	3	-2
Rank	3	4	2	1	5

16.3 Communication and File Sharing

The ability to access all the research conducted by individual group members was a key feature of the success of the project. Literature and files were shared using a joint Google Drive folder. This was a particularly useful when conducting research for the literature review, as a piece of information that would benefit another group member could be shared immediately and easily. This also allowed access to the project management documentation used throughout the project. This included meeting minutes and the latest version of the Gantt chart, which allowed each member of the group to remain up to date.

Good communication was vital, not only between the members of the group but with the project supervisor, Professor Cartmell. The main communication method used between the members of the group was a Facebook Messenger Group. This was an easy to use communication platform and additionally allowed the sharing of files such as PDFs and images if necessary. The project supervisor was kept up to date using email. This allowed a formal and clear communication platform for project updates and questions to be answered.

16.4 Reflection on Group Working

16.4.1 Strengths

The primary strength of the group was cohesiveness, through effective resource management. For example, the realization of the LEO to LLO transfer as the stem of the critical path of the project, led to the remaining group members to help out the member in charge whilst keeping an eye on upcoming major tasks in the timeline. This eventually led to the successful achievement of the deliverable associated, within the defined time allocated.

Another strength was the strong and continuous communication of the team over the duration of the project. All communication methods mentioned in the previous sections were essentially used to share the progress, challenges, future planning etc. regularly. This meant that the progress of the project was continually monitored by all members of the group, and the resource could be then managed accordingly.

16.4.2 Weaknesses and Improvements

At the beginning of the project there were moderate conflicting views upon, for example, final decision making. However, the group was able to realize this early on and decided upon using a controlled convergence matrix in selecting the best options for the given requirements. This considerably reduced any discrepancies and bias for future decision making as a group.

After the Interim assessment, the meetings were planned to be more structured: through specific agendas prepared prior to the scheduled meetings. This helped the following meetings to be concise and efficient, and adhere to the project objectives.

17 References

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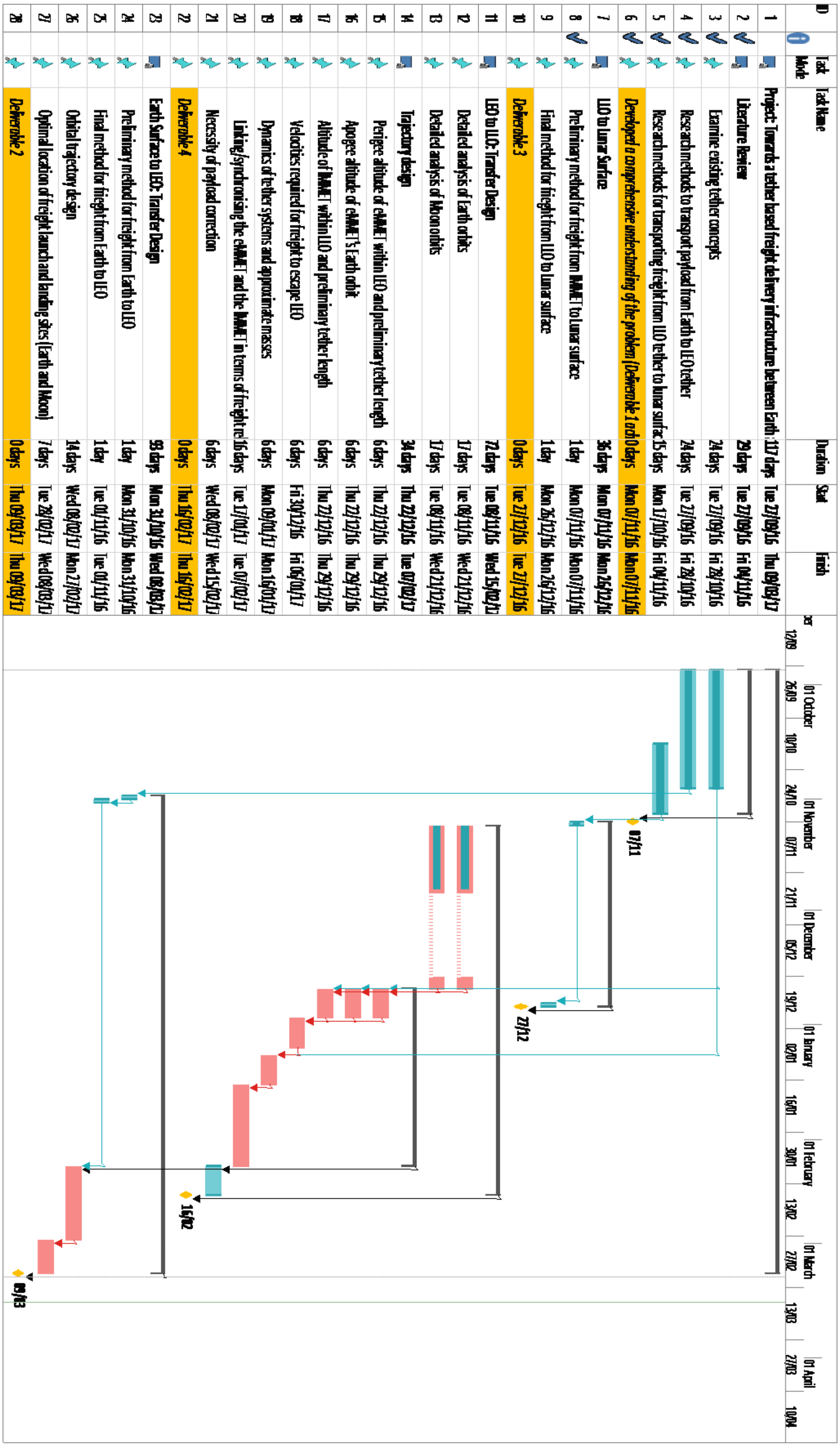
18 Appendices

18.1 Optimal Launch Location

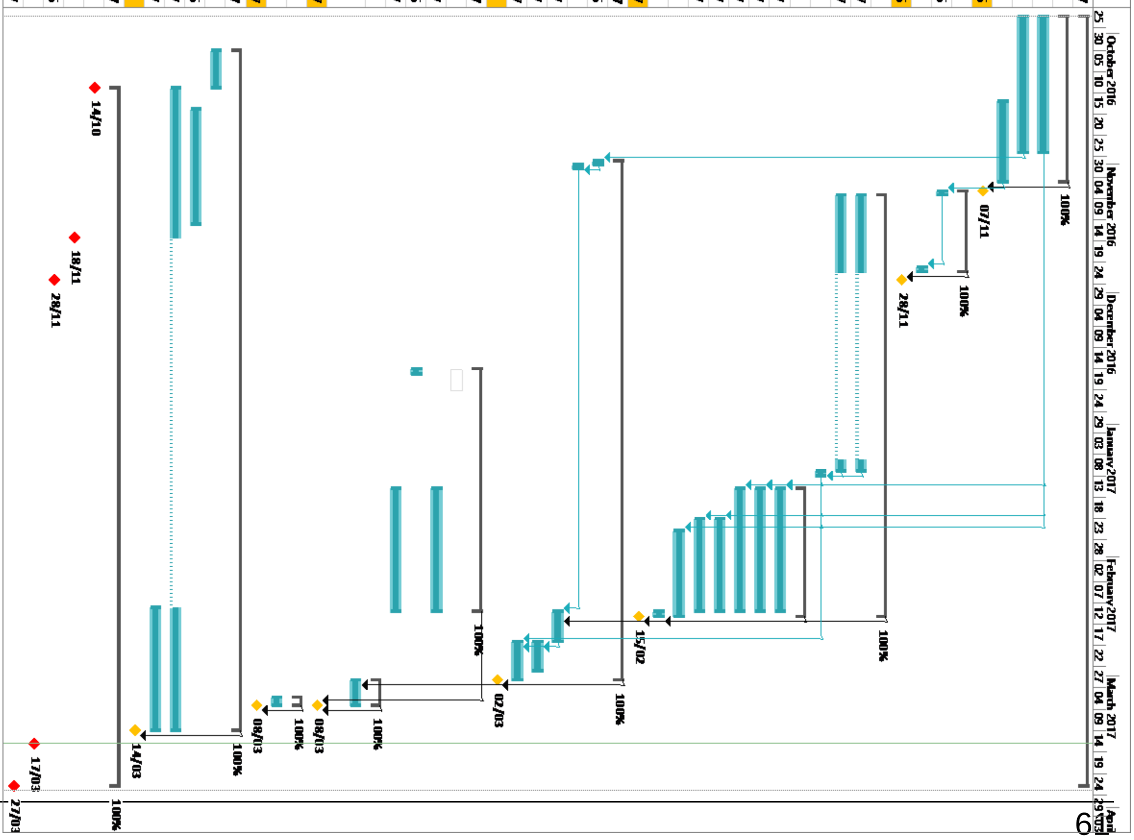
Latitude (deg)	Latitude (rads)	Inertial Azimuth (rads)	Inertial Azimuth (deg)	v_{xrot}	v_{yrot}	v_{rot} (m/s)
0	0	1.25175014	71.72	6722.40422	2374.25713	7129.3629
5	0.087266463	1.263523891	72.39458627	6751.62917	2289.47062	7129.24766
10	0.174532925	1.302346176	74.61893933	6840.34901	2007.71463	7128.90543
13.7374	**0.239762861**	**1.358250266**	**77.82200778**	**6947.36689**	**1596.78108**	**7128.50731**
15	0.261799388	1.386312089	79.42983182	6991.79989	1388.54565	7128.34657
18.28	0.319046187	1.570796327	90	7127.87018	4.6369E-13	7127.87018

18.2 **Gantt Chart 1**



18.3 Gantt Chart 2

ID	Task Name	Duration	Start	Finish
1	**Project: Towards a tether based freight delivery infrastructure between Earth and Moon**	**129 days**	**Tue 27/09/16**	**Mon 27/03/17**
2	**Literature Review**	**29 days**	**Tue 27/09/16**	**Fri 04/11/16**
3	Examine existing tether concepts	24 days	Tue 27/09/16	Fri 28/10/16
4	Research methods to transport payload from Earth to LEO tether	24 days	Tue 27/09/16	Fri 28/10/16
5	Research methods for transporting freight from LLO tether to lunar surface	15 days	Mon 17/10/16	Fri 04/11/16
6	*Developed a comprehensive understanding of the problem (Deliverable 1 achieved)*	0 days	Mon 07/11/16	Mon 07/11/16
7	**LLO to Lunar Surface**	**15 days**	**Mon 07/11/16**	**Fri 25/11/16**
8	Preliminary method for freight from IMMET to Lunar surface	1 day	Mon 07/11/16	Mon 07/11/16
9	Final method for frieght from LLO to Lunar surface	1 day	Fri 25/11/16	Fri 25/11/16
10	*Freight transport method for LLO to lunar surface selected (Deliverable 3 achieved)*	0 days	Mon 28/11/16	Mon 28/11/16
11	**LEO to LLO: Transfer Design**	**71 days**	**Tue 08/11/16**	**Tue 14/02/17**
12	Detailed analysis of Earth orbits	17 days	Tue 08/11/16	Wed 11/01/17
13	Detailed analysis of Moon orbits	17 days	Tue 08/11/16	Wed 11/01/17
14	Ideal analysis of Earth and Moon orbits	1 day	Thu 12/01/17	Thu 12/01/17
15	**Trajectory design**	**22 days**	**Mon 16/01/17**	**Tue 14/02/17**
16	Perigee altitude of eMMET within LEO and preliminary tether length	21 days	Mon 16/01/17	Mon 13/02/17
17	Apogee altitude of eMMET's Earth orbit	21 days	Mon 16/01/17	Mon 13/02/17
18	Altitude of IMMET within LLO and preliminary tether length	21 days	Mon 16/01/17	Mon 13/02/17
19	Velocities required for freight to escape LEO	16 days	Mon 23/01/17	Mon 13/02/17
20	Dynamics of tether systems and approximate masses	16 days	Mon 23/01/17	Mon 13/02/17
21	Linking/synchronising the eMMET and the IMMET in terms of freight release timings	14 days	Thu 26/01/17	Tue 14/02/17
22	Necessity of payload correction	1 day	Tue 14/02/17	Tue 14/02/17
23	*Developed a synchronozed and an unsychronized model (Deliverable 4 achieved)*	0 days	Wed 15/02/17	Wed 15/02/17
24	**Earth Surface to LEO: Transfer Design**	**88 days**	**Mon 31/10/16**	**Wed 01/03/17**
25	Preliminary method for freight from Earth to LEO	1 day	Mon 31/10/16	Mon 31/10/16
26	Final method for frieght from Earth to LEO	1 day	Tue 01/11/16	Tue 01/11/16
27	Orbital trajectory design	5 days	Tue 14/02/17	Mon 20/02/17
28	Arrival synchronization at capture	5 days	Tue 21/02/17	Mon 27/02/17
29	Optimal location of freight launch and landing sites (Earth and Moon)	7 days	Tue 21/02/17	Wed 01/03/17
30	*Earth to LEO transfer design completed (Deliverable 2 achieved)*	0 days	Thu 02/03/17	Thu 02/03/17
31	**Cost Analysis**	**41 days**	**Mon 19/12/16**	**Mon 13/02/17**
32	~~Direct transfer of freight by conventional rocketry~~	~~5 days~~	~~Mon 19/12/16~~	~~Fri 23/12/16~~
33	Intitial system setup	21 days	Mon 16/01/17	Mon 13/02/17
34	Freight transport using complete system system	1 day	Mon 19/12/16	Mon 19/12/16
35	Materials	21 days	Mon 16/01/17	Mon 13/02/17
36	**Comparison Study**	**4 days**	**Thu 02/03/17**	**Tue 07/03/17**
37	Advantages and disadvantages of using tethers over existing methods in relation to cost and resources	4 days	Thu 02/03/17	Tue 07/03/17
38	*Deliverable 5 and 6 achieved*	0 days	Wed 08/03/17	Wed 08/03/17
39	**Logistics**	**2 days**	**Mon 06/03/17**	**Tue 07/03/17**
40	GNC and Telecoms	2 days	Mon 06/03/17	Tue 07/03/17
41	*Deliverable 8 achieved*	0 days	Wed 08/03/17	Wed 08/03/17
42	**Reports and Commercial**	**114 days**	**Wed 05/10/16**	**Mon 13/03/17**
43	Write Statement of Purpose/Contract	7 days	Wed 05/10/16	Thu 13/10/16
44	Write Interim Report	19 days	Wed 19/10/16	Mon 14/11/16
45	Website Design	46 days	Fri 14/10/16	Mon 13/03/17
46	Write Final Report	21 days	Mon 13/02/17	Mon 13/03/17
47	*Deliverable 9 achieved*	0 days	Tue 14/03/17	Tue 14/03/17
48	**Deadlines**	**116 days**	**Fri 14/10/16**	**Mon 27/03/17**
49	Statement of Purpose Due	0 days	Fri 14/10/16	Fri 14/10/16
50	Interim Report Due	0 days	Fri 18/11/16	Fri 18/11/16
51	Interim Assessments	0 days	Mon 28/11/16	Mon 28/11/16
52	Final Report Due	0 days	Fri 17/03/17	Fri 17/03/17
53	Final Assessment	0 days	Mon 27/03/17	Mon 27/03/17



18.4 Meeting Minutes 1

Meeting 05/11/2016

Group Meeting

Work completed previously:

- All studied background of momentum exchange tethers as well as space travel in general
- Completed belbins management theory test

Agenda:

- Select management roles
    - Project manager – Ross
    - Communications manager – David
    - Bookkeeper – Ash

- Assign technical roles
    - Ash – LEO
    - David – interplanetary
    - Ross – Lunar orbit – Lunar surface

To do:

All:
Go away and do more focused study on our own areas with a view to selection of modes of transport?

Ash and Ross:
to present various methods of carrying out their transfer from Earth and Lunar surface at next meeting with a view to selection of concept.

David:
Start to examine the mechanics of the tether systems and how to orbits would synch in terms of perigees and apogees with the moon.

Next Meeting: 14th of November 2016

18.5 Meeting Minutes 2

Meeting 01/02/17

Meeting with Client

- Met Prof Cartmell to discuss progress and ask questions
  - Confirmed we are still on the right path
  - Integer calculation will be trial error
  - Emphasis on reducing the stress within the tether
  - Getting a low velocity disparity may not be possible with system proposed however this was expected. Likely to be between 0.2-1km/s

Group Meeting

Work completed:

- Ash has established method of synchronising payload transfer from earth to eMMET in orbit
- David has determined orbit path of the eMMET (elliptical) and the velocity required to achieve lunar transfer orbit.
- Ross looked at cost analysis in more detail, possibly not feasible in the time remaining, unsure if it would have been feasible anyway.

To do:

David
Continue with the synchronising of the tether systems and determine preliminary values for altitudes, all velocity changes, tether lengths and payload masses (1000kg) excel sheet
Ross
Feasible Lunar orbits. Is the Moon's orbital plane about the Earth feasible for long term stay? Further study of cost analysis and approximate costs that can be reasonably estimated.
Ash
Earth to LEO trajectory design (rough values of eMMET orbit)

- Vp_{CoM}=9km/s
- Va_{Com}=3.5km/s
- Vp_{Upper}=11km/s so deltaV=2km/s
- Vp_{Lower}=7.5km/s
- Rp_{com}=~200km so 40km tether Rp_{lower}=~160km

Take off sites need to accommodate inclination of about 18 to 28 degrees (at the moment)

Next Meeting: 9th of February 2017