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EARTHSHINE: TRAJECTORY AND PROPULSION

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EARTHSHINE: Trajectory and Propulsion

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Abstract

EARTHSHINE is a proposed deep space mission to be designed and developed exclusively by the UK. The mission will rely on proven small satellite technology and will be constructed primarily using commercial off the shelf components to minimise costs and production time. The mission is proposed to investigate if radiation from the Sun and galactic cosmic rays affect the Earths climate. The proposed mission uses an operational halo orbit around the first Earth-Sun liberation point and the purpose of this study was to investigate suitable low cost methods for transfer to the operational orbit.

A short review of existing liberation point missions is presented and a simple trajectory analysed to estimate the delta-V budget for the mission. The choice of a suitable launch vehicle is discussed and a minimum energy transfer strategy selected. Further detailed analysis of the trajectory was undertaken by performing simulations using STK software. A suitable flight proven propulsion system was chosen to perform the delta-V manoeuvres.

The transfer was analysed for a midnight launch on Ariane 5 ASAP into GTO. A simple 20N monopropellant engine was chosen to perform six perigee burns to increase the apogee radius of the GTO. The delta-V budget for the transfer was estimated to be 850m/s including a 10% contingency. The transfer will take approximately 360 days from first perigee passage of the GTO until a small impulsive manoeuvre will be required to insert into a large amplitude halo orbit. Hydrazine propellant mass of 40.5kg was determined for the transfer.

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Nomenclature

ACE	Advanced Composition Explorer
ASAP	Ariane 5 Structure for Auxiliary Payload
AOCS	Attitude and Orbital Control System
COTS	Commercial Off The Shelf
EARTHSHINE	Earth-Sun-Heliosphere Interactions Experiment
GAIA	Global Astrometric Interferometer for Astrophysics
GTO	Geostationary Transfer Orbit
ICE	International Cometary Explorer
ISEE-3	International Sun-Earth Explorer-3
LEO	Low Earth Orbit
HOI	Halo Orbit Insertion
MCC	Mid Course Correction
OSM	Orbit Shaping Manoeuvre
RAAN	Right Ascension of Ascending Node
SOHO	Solar and Heliospheric Observatory
A_y	Amplitude of halo orbit in y-direction [km]
i	Inclination [deg]
Zp	Altitude of perigee [km]
Za	Altitude of Apogee [km]
wp	Argument of perigee [deg]
v_{∞}^{E}	Earth relative speed outside Earth sphere of influence [m/s]
μ_S	Gravitational parameter of the Sun [m^3/s^2]
r_E	Average distance of Earth from Sun [m]
r_{L1}	Average distance of L1 from Sun [m]
μ_E	Gravitational parameter of Earth [m^3/s^2]

r_p	Radius of perigee of Earth orbit [m]
v_s	Velocity of spacecraft [m/s]
v_p	Velocity of spacecraft at perigee [m/s]
a	Semi major axis [m]
Δv	Delta-V [m/s]
m_0	Initial spacecraft mass [kg]
I_{sp}	Specific impulse [s]
g_0	Acceleration due to gravity of Earth [m/s^2]
m_p	Propellant mass [kg]
T	Thrust [N]
$\dot{m}$	Mass flow [kg/s]
t_b	Burn time [s]
B	Blowdown ratio
P_{gi}	Initial gas pressure [Pa]
P_{gf}	Final gas pressure [Pa]
V_{gi}	Initial ullage volume [m^3]
V_{gf}	Final ullage volume [m^3]

1. Introduction

EARTHSHINE is a proposed small satellite, low budget mission to answer key questions about how the Earth's climate and space environment are influenced by the Sun. The mission will be the UK's first deep-space mission and will use small satellite technology at large distances from the Earth (EARTHSHINE, 2003).

The mission is proposed to test the controversial hypothesis that radiation from the Sun and galactic cosmic rays influence the Earth's climate by affecting the amount and the type of cloud. Recent research provides increasing evidence of correlations between radiation and climate and the EARTHSHINE mission is designed to test some of the proposed mechanisms and correlations.

The baseline mission proposal uses just two payloads (Gibb, 2004), a combined visible and infra red camera and a cosmic ray detector. The primary functions of the two instruments are summarised in Table 1-1. The two payloads work together to provide the required information on the possible link between radiation and the climate.

Instrument	**Primary Function**
Amon-Ra	Visible and infra red cameras used to monitor Earth albedo in the visible and infra red wavelengths providing information on properties and occurrence of clouds.
Horus	Cosmic ray detector to measure and differentiate between galactic and solar cosmic rays.

Table 1-1: Payload choice for EARTHSHINE mission

The baseline operational orbit will be a halo orbit around the Earth-Sun L1 Lagrange point. This orbit provides an uninterrupted view of the dayside of the Earth allowing continuous measurements of cloud cover and incident cosmic rays. The L1 Lagrange point is outside the Earths magnetosphere and so offers a unique vantage point to monitor incident radiation and alongside Earth observations, to discover the shielding effect of the magnetosphere. The spacecraft will be 3-axis controlled during the operational orbit with a minimum operational lifetime of two years.

The principle driver for the mission is a tight cost envelope of just £5 million excluding payload, launch and operational costs. EARTHSHINE must therefore be built using commercial off-the-shelf (COTS) components to minimise costs and all the aspects of the mission must tie in with the low budget philosophy.

The project has been run in parallel with a study of the EARTHSHINE mission concept by industry professionals. During the project access to the initial EARTHSHINE mission proposal and technical description was readily available.

1.1. Project Objective

The aim of the project was to design a baseline mission capable of obtaining the science objectives whilst meeting the low budget constraints on the mission. The project was undertaken by a team of eight students and the work required to meet the objectives was divided equally between the members of the group, as summarised in Table 1-2.

Work Package	Tasks	Name
100	System Engineering	S. Palmer and Y. Sevcenco
200	Operational Orbit	J. Reveles
200	Trajectory and Propulsion	A. Helliwell
200	Attitude Control	Y. Sevcenco
400	Structural and Thermal	C. Wood
500	Power and Communications	C. Cramond
600	Science	N. Gibb
600	Payload	L. Lubino
700	Alternative Mission Concepts	S. Palmer

Table 1-2: Work package allocation

1.2. Project Strategy

The aim of the Trajectory and Propulsion work package was to investigate and to determine the most cost effective method for transfer of EARTHSHINE to the operational orbit. This involved:

- Choice of suitable low cost launch vehicle
- Analysis of minimum energy transfer trajectory to operational orbit
- Choice of propulsion system to meet the low cost constraints of the mission

1.3. Report Outline

This report has been arranged to give the reader an indication of the evolution of the trajectory and propulsion options throughout the project. The chapters are arranged to show how the initial analysis of the problem was performed, progressing through to more detailed analysis and final trajectory and subsystem design for EARTHSHINE.

Chapter 2, *Liberation Point Missions*: Provides an overview of liberation point missions and the techniques used to transfer to halo orbits around the collinear liberation points.

Chapter 3, *Launch Vehicle Considerations:* Discusses the options available for launching the EARTHSHINE spacecraft in relation to the tight cost budget for the mission.

Chapter 4, *Initial Trajectory Analysis*: Discusses how the initial analysis of the trajectory was performed and how the initial estimate of the delta-V budget for the transfer was obtained.

Chapter 5, *Propulsion System Options:* Summarises the reliable, low cost commercially available propulsion options available for the mission.

Chapter 6, *Detailed Trajectory Analysis and Propulsion Choice:* Summarises the main results and findings of the report. Discusses the final choice of propulsion system and demonstrates the transfer strategy for the mission using simulations performed in STK.

Chapter 7, *Propulsion System Components:* Outlines the choice of components for the propulsion system that meet the low cost constraints of the mission. Discusses how the chosen system could affect the transfer analysis results.

Chapter 8, *Comparison with Industry Proposal:* Briefly discusses the similarities and differences in the results of this report and the current industry proposal for the EARTHSHINE mission.

Chapter 9, *Conclusions:* Briefly concludes the results which have been obtained and comments on further work that would be required if more time was available for the study.

2. Liberation Point Missions

In 1978, the International Sun-Earth Explorer-3 (ISEE-3) was the first spacecraft to travel to and maintain operational orbit in a halo orbit around the first Sun-Earth liberation point. Since this initial pioneering mission, several spacecraft have orbited around L1 and L2 and there are a number of planned missions which will travel to the collinear liberation points.

2.1. The Liberation Points

The EARTHSHINE mission baseline uses a halo orbit around the L1 Lagrange point as the operational orbit. The Lagrange points are equilibrium or liberation points in the Earth, Sun and Moon system where the forces on a spacecraft from the large gravitational bodies balance out.

In the solar rotating coordinate system shown in Figure 2-1, the equilibrium points occur where the velocity and acceleration components are zero and the forces on the spacecraft are balanced. The centrifugal force balances with the gravitational force of the two primaries (Folta, 2003).

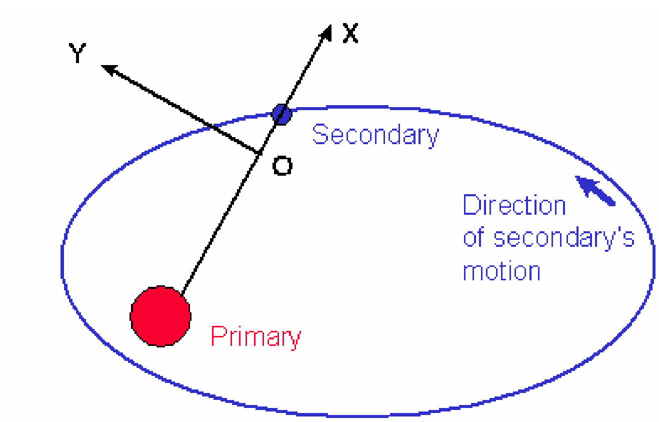

Figure 2-1: Rotating liberation point coordinate system where the primary body is the Sun and the secondary is the centre of mass of the Earth (STK, 2004)

There are five liberation points in the Sun-Earth system (likewise in the Earth-Moon system). The positions of the five equilibrium points are illustrated in Figure 2-2.

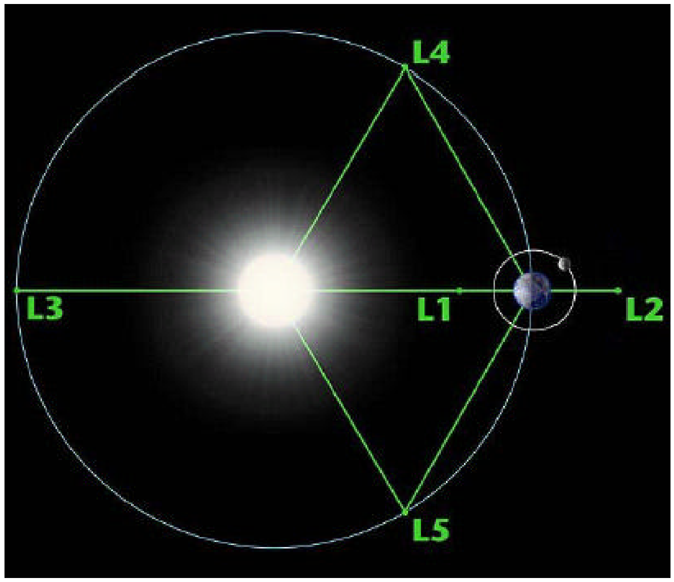

Figure 2-2: Positions of the five liberation points in the Sun-Earth-Moon system (Kim, 2003)

The collinear liberation points L1, L2 and L3 are inherently unstable whilst the two triangular points L4 and L5 are stable. L1 is the liberation point chosen for the EARTHSHINE mission because it provides an uninterrupted view of the dayside of the Earth. The L1 and L2 points are approximately 1.5 million kilometres either side of the Earth on the Earth-Sun line.

2.1.1. Liberation Point Orbits

Liberation point orbits are obtained through analysis of the three body problem containing the Earth, Sun and spacecraft. The solutions of the equations of motion reveal two distinct motions about the Lagrange points

The lissajous orbit has a different angular velocity in the Z direction than in the X and Y and as a result, the orbit oscillates about the Y-axis. The halo orbit is a special solution of the motion where the angular velocity in the Z direction is the same as that in the X and Y directions. The halo orbit is periodic with a period of motion about the Lagrange point of just less than six months. The EARTHSHINE mission baseline uses a large amplitude halo as the operational orbit.

2.2. Technique used for Transfer to L1 and L2

Transfer to the two liberation points closest to the Earth is performed in a similar way. A typical L2 transfer trajectory is shown in geocentric inertial and solar rotating coordinates in Figures 2-3 and 2-4 respectively.

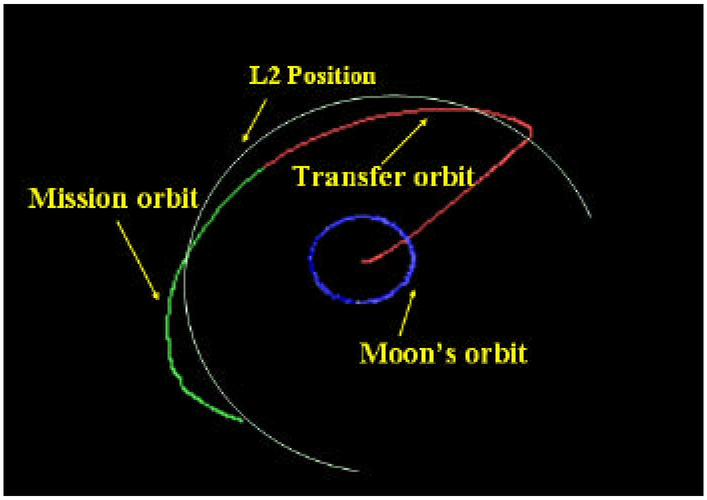

Figure 2-3: Transfer in geocentric inertial coordinate system (Folta, 2003)

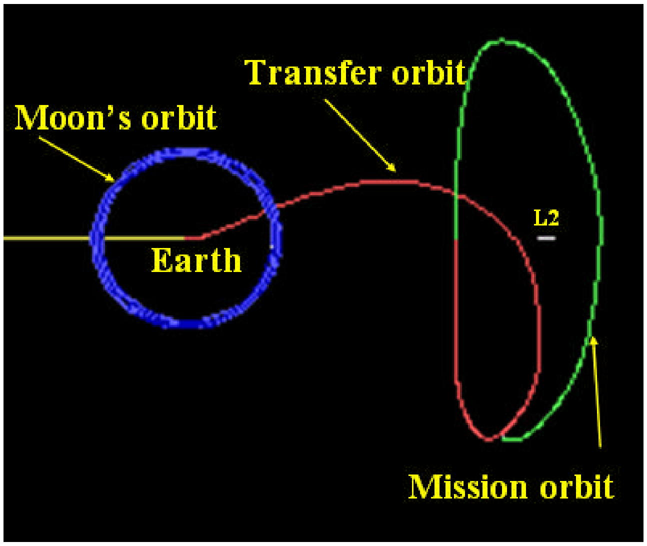

Figure 2-4: L2 transfer in solar rotating coordinate system (Folta, 2003)

In the geocentric inertial coordinates, the L2 point orbits the Earth. The spacecraft transfers rapidly to L2 distance before turning naturally due to the attraction of the Earth. The halo orbit is obtained at the end of the transfer phase possibly with an insertion delta-V. Insertion into the halo orbit is obtained when the spacecraft velocity in the X direction is zero at the Z-X plane crossing in the rotating coordinates. This can be seen in Figure 2-4 where the trajectory line is 90 degrees to the Z-X plane crossing (Earth -L2 line).

All of the missions which have travelled to the liberation points so far have used a dedicated launch vehicle to insert the spacecraft into the transfer trajectory. Several manoeuvres are then required to position the spacecraft in the correct position for insertion into the halo or lissajous orbit (Sharer, 1996).

The first burn, called the Mid Course Correction (MCC) manoeuvre, occurs a few days after launch to correct for inaccuracy in launch vehicle performance.

The second manoeuvre is called the Orbit Shaping Manoeuvre (OSM) and is only required if there are restrictions on the Sun, Earth, vehicle angle during the transfer. These restrictions may be imposed on the spacecraft if it is not desirable to have the spacecraft drifting too far from the Earth-Sun line for power collection or communication reasons.

The final burn is the Halo Orbit Insertion (HOI) manoeuvre. .The delta-V requirements for insertion into a halo orbit are directly related to the amplitude of the halo orbit as shown in Figure 2-5. Essentially, a smaller halo orbit requires more energy and hence a higher delta-V to insert into it. The size of halo orbit used is generally chosen as a compromise between the delta-V budget, the communication requirements and meeting the science objectives of the mission.

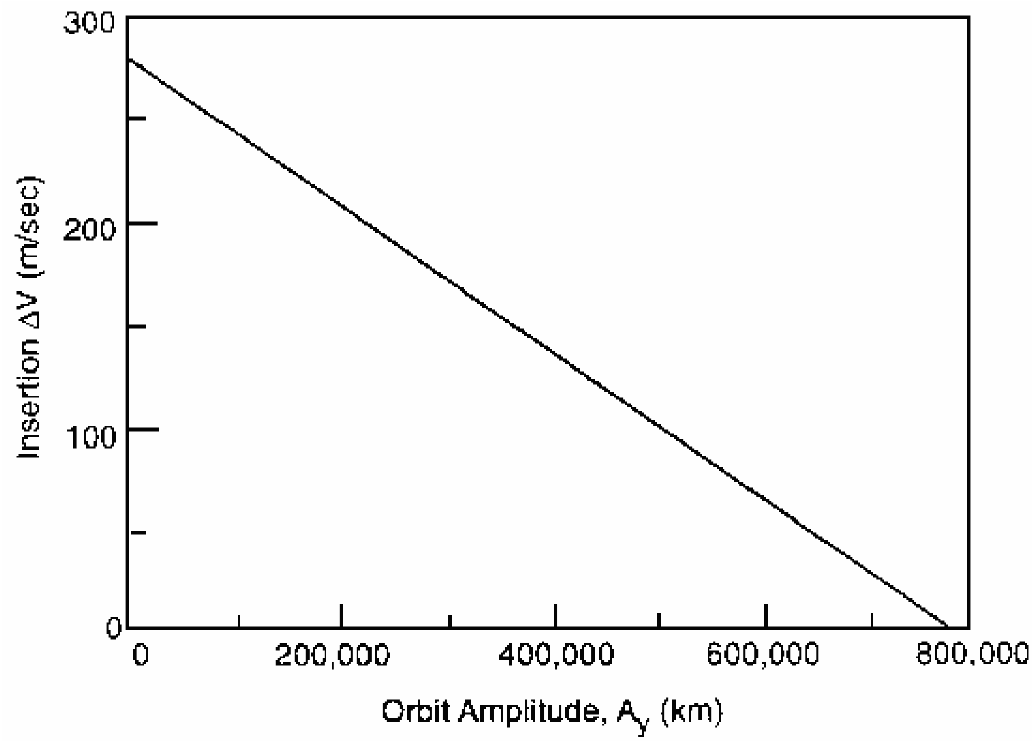

Figure 2-5: Relationship between insertion delta-V and amplitude of halo orbit (Farquhar, 1998)

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2.3. Existing and Proposed Liberation Point Missions

The technique used to travel to the L1 and L2 liberation points is essentially the same for each mission. Slight differences occur between each mission depending on the size of halo or lissajous orbit and on whether restrictions are imposed on the Sun, Earth and vehicle angle. Information on these missions was useful for understanding the trajectory to L1 and for obtaining an initial estimate of the transfer delta-V budget for EARTHSHINE.

2.3.1. ISEE-3

The Interplanetary Sun-Earth Explorer-3 (ISEE-3) was launched on August 12 1978 by Delta rocket #144. The transfer phase took 100 days during which three manoeuvres were performed to shape the transfer appropriately for insertion into the desired operational orbit.

The first manoeuvre cost 17.9m/s and was performed less than one day after the launch to remove the effects of inaccuracy in launch vehicle performance. The second Mid Course Correction manoeuvre was performed after 25 days and was essentially an orbit shaping manoeuvre of 24.8m/s. Finally, a delta-V of 14.0m/s was provided after 100 days to insert the spacecraft into a halo obit with Y-amplitude of 666,670km (Farquhar, 1998).

ISEE-3 performed four orbits around the halo orbit with station keeping costs of just 8.5m/s per year. The L1 point is inherently unstable and so the spacecraft needed to provide some energy to maintain in an optimal halo orbit. This energy however was minimal.

ISEE-3 monitored interplanetary conditions successfully from L1 before travelling to the geomagnetic tail and visiting a comet under a new name of International Cometary Explorer (ICE).

2.3.2. ACE

The Advanced Composition Explorer (ACE) was launched in 1997 to determine and compare the elemental and isotopic composition of several samples of matter including solar corona, interplanetary material, local interstellar material and intergalactic matter.

ACE was launched on a Delta 2 rocket into a near escape trajectory. A MCC manoeuvre was performed after 2 days, an OSM after 17 and the HOI burn was performed 117 days after launch.

The Halo Orbit Insertion cost over 140m/s due to a small amplitude halo being required. The mission constraints required a halo of between 5 and 10 degrees as seen from Earth. This 5 degree minimum was the solar exclusion zone which had to be avoided to prevent communication black out due to radio wave emission from the

Sun. This solar exclusion zone is standard for all missions. The 10 degree upper limit was specific to the ACE mission (Sharer, 1996).

2.3.3. GAIA

The Global Astrometric Interferometer for Astrophysics (GAIA) is a proposed L2 liberation point mission to acquire information on the composition, formation and evolution of the galaxy.

The proposal for the transfer of GAIA has now changed, however the initial proposal used Ariane 5 for launch into GTO. GAIA would then require its own on board propulsion system to inject it into the transfer trajectory. This would be performed by an apogee raising sequence with a nominal impulsive delta-V requirement of 750m/s (GAIA, 2000).

The operational orbit for GAIA is a lissajous orbit with total transfer delta-V of 1150m/s from a typical GTO provided by Ariane 5.

2.4. Summary of Liberation Point Transfers

The delta-V budget for transfer to a liberation point orbit is largely dependent on the launch vehicle used and the operational orbit required. Theoretically, a large amplitude halo orbit could be obtained from an accurate direct injection launch, with no delta-V manoeuvres required for the transfer. However, in reality small manoeuvres are always required to correct for erro rs in the launch vehicle performance.

The delta-V budget is increased if a smaller amplitude halo orbit or lissajous orbit is required. If a direct launch cannot be obtained, then the velocity budget is much higher than the dedicated launch vehicle approach. Transfer fom GTO requires an extra 750m/s to travel toward the liberation point. This velocity would normally be provided by the launch vehicle in a direct injection launch.

3. Launch Vehicle Considerations

There are two possibilities when choosing a suitable launch vehicle for a liberation point mission. The first option uses a dedicated launch vehicle to insert the spacecraft directly into the transfer trajectory. The direct injection method is employed by the majority of liberation point missions to keep the delta-V budget for the transfer as low as possible. The second option is for the spacecraft to be launched into parking orbit around the Earth. The spacecraft would then require its own propulsion to transfer it towards the L1 point. The delta-V budget for this technique would be much larger than for direct injection, however the launch costs are much reduced because it would be possible to share the launch into parking orbit with another payload.

3.1. Direct Injection or Shared Launch

The primary aim when choosing the launcher is to minimise costs. The EARTHSHINE mission is low budget and so launch costs must be kept to a minimum. Often, the most effective way to reduce the launch price is to share the launch with another spacecraft. For example, if two or more spacecraft require the same orbit from the launch then the two spacecraft can be launched together at a shared launch cost. The biggest disadvantage with shared launch is that the primary payload always has control of the launch and therefore the secondary payload cannot specify a different parking orbit to the primary payload.

The most desirable launch for EARTHSHINE would use the upper stage of the launch vehicle for direct insertion into the transfer trajectory. This would minimise the manoeuvres the spacecraft would need to perform, keeping the delta-V budget low and the propulsion system relatively simple. However, it would generally not be possible to share a direct injection launch to L1 with another payload and so this option would require a costly dedicated launch vehicle.

The Dnepr launch vehicle marketed by Kosmotras appears to be the cheapest launch vehicle available for a direct launch into a trajectory towards L1. The launch cost in 2003 was fixed at $13 million (Kim, 2003).

There are many options available for shared launch into parking orbit around the Earth. Most launch vehicles that provide shared launches operate into Low Earth Orbit (LEO), whilst some vehicles provide shared launch capabilities into Geostationary Transfer Orbit (GTO), primarily for the commercial applications of geostationary orbit.

For transfer to L1, launch into GTO offers a reduced delta -V budget when compared to transfer from LEO due to the higher energy state of the GTO. Transfer into a L1 halo orbit from GTO can be performed for less than 1000m/s (Hoff, 2002) however transfer from LEO costs over 3000m/s (Kim, 2003). For transfer from a parking orbit, launch into GTO is therefore more favourable due to the significantly reduced delta-V budget compared to launch into LEO.

Launch into GTO is generally more costly than launch into LEO. The extra energy of the GTO must be provided by the launch vehicle hence launchers providing access to GTO are often large and expensive. However, the Ariane 5 launch vehicle has a facility for launching small spacecraft along with a primary payload, and offers much reduced launch costs over the majority of shared launches into Earth parking orbit.

The Ariane 5 Structure for Auxiliary Payload (ASAP) has the capability to launch micro or mini satellites as auxiliary payloads alongside the primary payload. The ASAP facility can be operated in three configurations which allow either eight micro satellites with maximum mass of 120kg, four mini satellites of less than 300kg or a combination of mini and micro satellites to be launched simultaneously (Ariane, 2000).

The launch cost for a micro satellite of 120kg into GTO was estimated in 2003 to be $5–6 million (Kim, 2003). This offers a significant cost saving over a direct injection launch, but restricts the mass and dimensions of the spacecraft significantly. The spacecraft will also require a substantial on board propulsion system to transfer from the parking orbit. However, the cost savings are so significant that Ariane 5 ASAP will be used as the baseline launch vehicle for EARTHSHINE.

For launch as a micro satellite on Ariane 5 ASAP, EARTHSHINE will be restricted to a mass of 120kg and external dimensions of 0.6m x 0.6m x 0.7m (Ariane, 2000). Launch as a mini satellite increases the available mass to 300kg, however launch costs and the amount of propulsion required would be higher for the mini satellite. With such a low cost budget for the mission, the 120kg micro satellite was chosen as the baseline.

3.2. GTO provided by Ariane 5

The GTO parameters provided by Ariane 5 will essentially be dictated by the primary payload. However, to accommodate typical attitude constraints for the apogee burn of commercial spacecraft, Arianespace have fixed a standard midnight launch window for double launches. The perigee of the GTO is then pointed away from the Sun direction, with the apogee pointed towards the Sun (GAIA, 2000). A GTO of this orientation was chosen as the baseline for the orbit EARTHSHINE would be placed in by Ariane 5 ASAP.

The standard Ariane 5 GTO has the following orbital parameters (Arianespace, 2000):

- Inclination i = 7 deg
- Altitude of Perigee Zp = 560km
- Altitude of Apogee Za = 35,890km
- Argument of Perigee wp = 178 deg

The longitude of the first descending node is usually positioned around 10 deg west as shown in Figure 3-1.

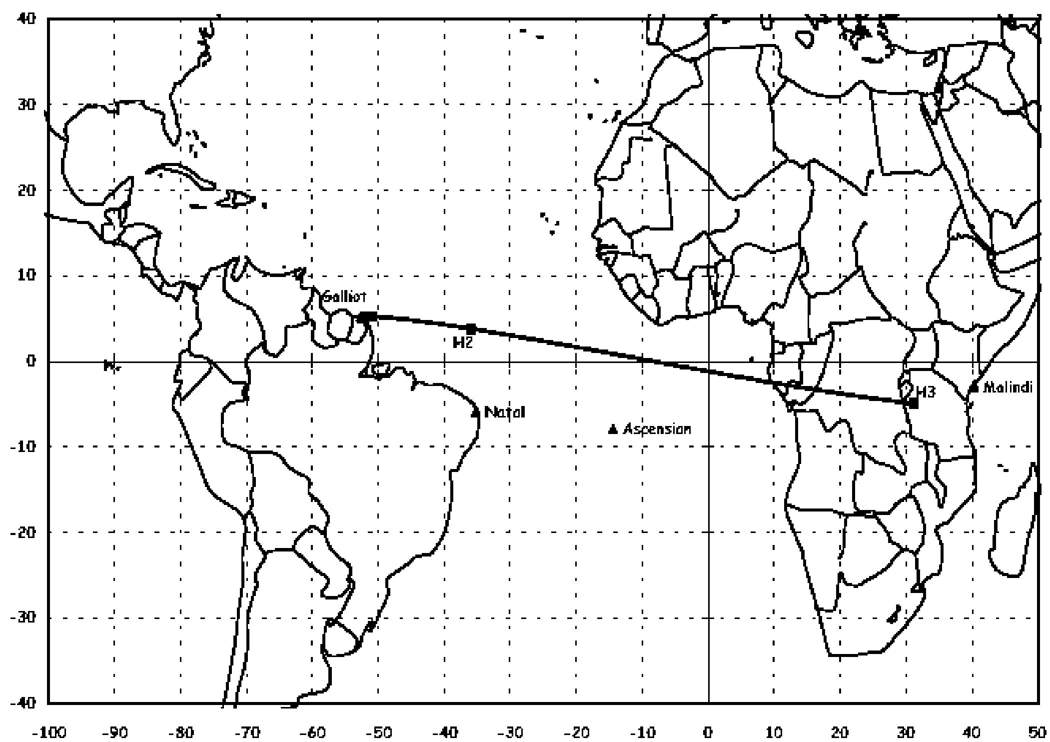

Figure 3-1: Typical ground track of Ariane 5 commercial launch (Arianespace, 2000)

4. Initial Trajectory Analysis

A simple initial analysis of the transfer was performed to estimate the delta-V budget for the transfer. This initial calculation of the delta-V budget was used to make estimates of the type of propulsion and mass of propellant required for transfer from GTO. These initial values were important for determining a baseline design for EARTHSHINE.

4.1. Patched Conic Approximation

The patched conic method is a standard technique used for analysing interplanetary trajectories. The technique involves splitting the transfer into three different stages in which only motion about the central body need be considered. The method also assumes that all planetary orbits are circular and that the transfer is of the Hohmann minimum energy transfer type. Strictly, the patched conic approximation cannot be applied to a liberation point mission because the approximation breaks down when more than one gravitational body needs to be considered. However, as a first estimate the technique provides a valid indication of the delta-V required to transfer from GTO towards L1.

The first stage of the patched conic method is the geocentric phase in which the spacecraft is in GTO around the Earth. The spacecraft would be given a boost at perigee to transfer it into the trajectory towards L1. The burn will be performed at perigee because at this point the spacecraft will be travelling at the fastest point in the parking orbit and so a smaller delta-V will be required to move into the transfer trajectory.

The second stage is the heliocentric phase in which the satellite moves in the sphere of influence of the Sun. This is the transfer trajectory and is a minimum energy Hohmann transfer.

The third stage is the arrival phase which normally translates to the arrival of the spacecraft in the sphere of influence of another planet. For a liberation point transfer there is no arrival phase and so only the first two stages needed analysis in this case.

The heliocentric phase was analysed first. The L1 point was assumed to orbit the Sun at the same speed as the Earth but at a distance of 1.5 million kilometres closer to the Sun. The Hohmann transfer in the heliocentric phase is shown in Figure 4-1.

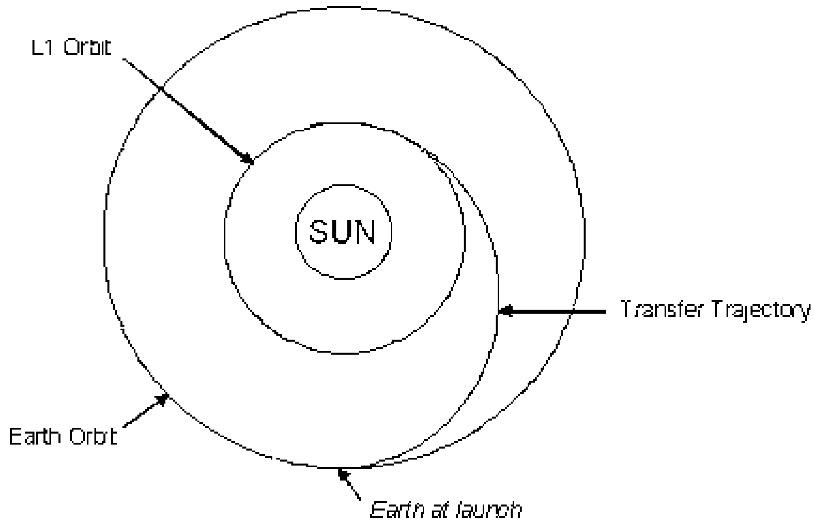

Figure 4-1: Schematic of patched conic transfer orbit in heliocentric phase

The required Earth relative speed of the spacecraft as it exits the Earth sphere of influence, is given by Equation 4.1 (Fortescue, 2003),

$$v_{\infty}^{E} = \sqrt{\frac{\mu_S}{r_E}}\left(\sqrt{\frac{2r_{L1}}{r_E + r_{L1}}} - 1\right) \tag{4.1}$$

where μ_S is the gravitational parameter of the Sun and r_E and r_{L1} are respectively the orbit radii of the Earth and the L1 point around the Sun. The required Earth relative speed v_{∞}^{E} was calculated to be -75.2m/s.

Assuming zero potential energy as the spacecraft leaves the Earths sphere of influence, energy conservation was used to calculate the velocity of the satellite required in the sphere of influence of the Earth to achieve an Earth relative speed of - 75.2m/s at infinity. Equation 4.2 represents the energy conservation of the system (Fortescue, 2003).

$$\frac{1}{2}v_S^2 - \frac{\mu_E}{r_p} = \frac{1}{2}v_{\infty}^{E\,2} \tag{4.2}$$

where μ_E is the gravitational parameter of the Earth and r_p is the radius of perigee of the GTO in the geocentric phase. v_S is the velocity of the spacecraft required and was calculated to be 10718m/s.

The velocity of the spacecraft at perigee of the GTO is 9933m/s which was calculated using Equation 4.3,

$$v_P = \sqrt{2\mu_E\left(\frac{1}{r_p} - \frac{1}{2a}\right)} \tag{4.3}$$

where a is the semi major axis of the GTO and v_p is the velocity of the spacecraft at perigee.

Finally, the impulsive delta-V, ?v, required to transfer from perigee of the GTO to the transfer trajectory was calculated to be 785m/s using Equation 4.4,

$$\Delta v = v_S - v_p \tag{4.4}$$

The initial estimate of the operational halo orbit used for the mission required an insertion delta-V of between 200m/s and 300m/s (Reveles, 2004). This requirement added to the delta-V budget of the mission giving an initial velocity budget estimate of between 985m/s and 1085m/s.

4.2. Existing Mission Delta-V Budgets from GTO

All of the liberation point missions to date have transferred to the Lagrange points using a dedicated launch vehicle. The delta-V budgets for these existing missions are therefore not representative of transfer from GTO. However, the initial GAIA proposal was to launch on Ariane 5 into GTO and so the delta-V budget for this mission provided a good indication of the velocity budget required for EARTHSHINE.

Using the patched conic method, the delta-V required to move from perigee of the GTO into the transfer trajectory was calculated to be 785m/s. Approximately, the delta-V budgets for the existing missions could be adjusted using the 785m/s to provide an estimate of the likely transfer delta-V budget form GTO. This is summarised in Table 4-1.

Spacecraft	Mission Delta-V Budget (m/s)	Theoretical Delta-V from Perigee of GTO (m/s)
ISEE-3	58	843
WIND	472	1257
SOHO	114	899
ACE	393	1178
MAP	109	894
GAIA	1150	1150

Table 4-1: Theoretical delta-V budgets for transfer from perigee of GTO (Folta, 1998. GAIA, 2000)

4.3. Initial Estimate of EARTHSHINE Delta-V Budget

Although only approximate, the patched conic method was used to obtain a delta-V of between 985m/s and 1085m/s for transfer from GTO. The velocity budget of the transfer depends significantly on the size of halo orbit however a delta-V of 1000m/s is representative of a typical transfer to a liberation point orbit.

The initial estimate of the delta-V budget was chosen to be between 1000m/s and 1400m/s. This large range was chosen so the baseline propulsion system would be able to cope with any unexpected increases in the velocity budget as a more rigorous analysis of the trajectory was performed.

5. Propulsion System Options

EARTHSHINE requires a propulsion system to transfer from perigee of the GTO into the L1 trajectory. An insertion delta-V may then be required and station keeping manoeuvres will be needed throughout the operational lifetime of the mission. Transfer from GTO typically will not require any MCC manoeuvres because errors in launch vehicle performance can be removed during the perigee burn.

The propulsion system must meet the low cost constraints of the mission and so must use reliable COTS components. The main type of propulsion that meets this requirement is chemical propulsion.

Electric propulsion is becoming more widely used with flight tested components now available. However, electric propulsion systems typically only offer very low thrust. The technique that would need to be employed to transfer to L1 using electric propulsion would be a continuous low thrust trajectory. This would increase transfer time significantly and the spacecraft would traverse the potentially harmful radiation belts on many occasions. It would not be possible to provide an impulsive insertion manoeuvre for the halo orbit using electric propulsion. Chemical propulsion can typically provide large thrusts from reliable flight proven hardware and so was chosen as the most effective propulsion system for EARTHSHINE.

5.1. Chemical Propulsion Options

Chemical propulsion systems utilise either solid or liquid propellants. The solid motor offers a simple, non-reusable system for single impulsive manoeuvres. Liquid propellant engines are restartable and can be used for several different burns throughout the mission. They can provide more control than solid motors, however are generally more costly and complex. Table 5-1 summarises the characteristics of chemical propulsion.

	Solid Propellant	Liquid Monopropellant	Liquid Bipropellant
Specific Impulse (s)	280 - 300	150 - 230	290 - 450
Thrust Range (N)	$50 – 5x10^6$	0.1 – 400	$5 – 5x10^6$
Restart	No	Yes	Yes
Pulsing	No	Yes	Yes
Throttling	No	Yes	Yes

Table 5-1: Main characteristics of chemical propulsion systems (Wertz, 1999. Brown, 1995)

5.1.1.Solid Motor

A solid motor uses solid propellant to provide a single burn for applications where a single impulsive manoeuvre is required. Solid propellants are typically powdered aluminium with ammonium perchlorate as the oxidiser (Wertz, 1999). They are safe and storable which is desirable for launch on Ariane 5 ASAP. Secondary payloads carrying a large amount of fuel offer a risk of explosion to the primary payload during launch. Solid propellants minimise this potential risk.

EARTHSHINE could use a solid motor to provide the necessary delta-V to insert it into the transfer trajectory from perigee of the GTO. This would be provided by one high thrust burn. The motor could then be jettisoned or could remain fixed to the spacecraft during transfer and operational life.

If a solid kick motor was used, then EARTHSHINE would still require an additional propulsion system for any other manoeuvres which may be required. Liquid propulsion would be required for possible insertion into the halo orbit and station keeping during operation. A solid motor cannot be controlled throughout its burn and so MCC manoeuvres would probably be required to correct for inaccuracies in its performance.

Ariane 5 ASAP has the capability for a thruster nozzle to protrude from the base of the spacecraft. However, a solid motor would be significant in size and would be difficult to accommodate in the ASAP launch facility.

A solid motor was therefore not chosen for EARTHSHINE because the burning cannot be controlled, the motor cannot be restarted and the size of the motor may be too large to accommodate in the ASAP launch facility.

5.1.2.Liquid Monopropellant Engine

A liquid monopropellant engine is the simplest and cheapest form of liquid chemical propulsion. Hydrazine is the most commonly used fuel which is thermally and catalytically decomposed over a platinum or iridium catalyst to provide thrust. Hydrazine is readily and safely stored as a liquid under pressure, with boiling and freezing temperatures of 387 and 275 Kelvin respectively (Fortescue, 2003).

A monopropellant engine can be restarted several times and hydrazine could potentially be the fuel for all the propulsive manoeuvres required for the EARTHSHINE mission. However, the thrust levels required for the perigee burn and station keeping are significantly different (EARTHSHINE, 2003) and so different thrusters would be required, linked to the same propellant tank.

5.1.3.Liquid Bipropellant Engine

A bipropellant engine is a more advanced liquid propellant engine which uses the reaction between the fuel and an oxidiser to produce thrust. There are a variety of possible fuel and oxidiser combinations that each provide significant increases in specific impulse and hence performance over a monopropellant engine. The increase in performance and saving in propellant mass is however cancelled out somewhat by the increase in mass, complexity and cost of the bipropellant system.

The most common fuel and oxidiser combination is monomethylhydrazine and nitrogen tetroxide (Wertz, 1999). The bipropellant system requires the fuel and oxidiser to be stored separately, with precise propellant management to ensure accurate mixing of the fuel and oxidiser in the combustion chamber.

5.2. Initial Estimate of Propulsion Budget

The initial estimate of the delta -V budget for the transfer was calculated to be between 1000m/s and 1400m/s. This velocity requirement will be provided by a liquid propellant engine. The initial estimate of the propellant budget was performed using a specific impulse of 310 seconds. This would be representative of an efficient bipropellant engine.

A form of the Tsiolkovsky rocket equation shown in Equation 5.1 (Wertz, 1999) was used to determine the initial estimate of the mass of propellant required for EARTHSHINE.

$$m_p = m_0 \left[1 - \exp\left(\frac{-\Delta v}{I_{sp} g_0} \right) \right] \quad (5.1)$$

where m_0 is the initial spacecraft mass (120kg for launch on ASAP), ?v is the velocity increase required, I_{sp} is the specific impulse of the engine and g_0 is the acceleration due to gravity of the Earth. The propellant mass, m_p was calculated to be between 34kg and 44kg for the delta-V range required.

The specific impulse chosen for the initial propellant estimate was slightly too high for commercially available bipropellant engines and was significantly greater than the specific impulse of a typical monopropellant engine. However, the delta-V budget was over estimated and so the range of 34kg – 44kg provided a suitable first estimate of the propellant mass required for the EARTHSHINE mission transfer.

6. Detailed Trajectory Analysis and Propulsion Choice

The patched conic method was used to provide a suitable first estimate of the delta-V budget for the transfer. However, to provide a more accurate estimate of the delta-V budget a more detailed analysis of the trajectory was required.

6.1. Transfer Strategy from GTO

EARTHSHINE will be the first spacecraft to transfer to the L1 liberation point without the use of a dedicated launch vehicle. The transfer strategy required for the mission is therefore unique. To minimise the mass of propellant that must be carried on the spacecraft, a minimum delta-V transfer technique will be required.

A minimum delta-V transfer can be obtained by performing the transfer using natural drift (Hoff, 2002). Halo orbits are very sensitive to perturbations and a spacecraft orbiting around L1 can easily move out of the orbit into Earth passage if no control or station keeping manoeuvres are performed. It is possible to take advantage of this natural motion for the transfer by moving form the Earth on a trajectory similar to the natural drift passage from the halo.

This transfer technique works only for large halo orbits with amplitude greater than 750,000km. For halo orbits of this size, insertion can be performed with minimal if not zero insertion delta-V (Figure 2-5). To meet the science requirements, EARTHSHINE would ideally be situated as near to the Earth-Sun line as possible. However this would require a smaller halo orbit with a larger insertion delta-V. The science can still be obtained from a large amplitude halo orbit and so a halo of amplitude 750,000km or greater was chosen as a compromise between the transfer and the science objectives.

Insertion into a large amplitude halo orbit can be achieved by increasing the apogee of the GTO to a radius of 1.2 million km (Hoff, 2002). Using this technique, the spacecraft will only have to provide the delta-V to increase the apogee to the correct radius. A small insertion delta-V may be required for the halo, but no other manoeuvres will need to be performed. This technique therefore offers the minimum delta-V transfer for the mission.

6.1.1. Thrust Limitations

The apogee of the GTO will be increased by performing a manoeuvre at perigee. Due to the limited dimensions and mass of the spacecraft, the liquid propellant engine will be limited to a thrust of 20N. Some engines are available that offer higher thrust but these are generally more costly and the size of the unit increases with increasing thrust making accommodation in the spacecraft difficult.

The total delta-V required to raise the apogee of the GTO to 1.2 million km was determined by calculating the perigee velocities of the GTO and the required Earth orbit with apogee at 1.2 million km. The perigee velocities were calculated using Equation 6.1 (Fortescue, 2003),

$$v_p = \sqrt{2\mu_E\left(\frac{1}{r_p} - \frac{1}{2a}\right)} \tag{6.1}$$

where μ_E is the gravitational parameter of the Earth, r_p is the radius of perigee and a is the semi major axis of the appropriate orbit. For the GTO parameters stated in section 3.2 the perigee velocity v_p, was calculated to be 9935m/s. The perigee velocity for the orbit with apogee at 1.2 million km was calculated to be 10688m/s giving an impulsive delta-V increase of 753m/s to move from GTO into the transfer orbit.

The thrust, T is given by Equation 6.2 (Fortescue, 2003),

$$T = I_{sp} g_0 \dot{m} \tag{6.2}$$

where I_{sp} is the specific impulse, g_0 is the acceleration due to gravity of the Earth and $\dot{m}$ is the mass flow from the exhaust.

The burn time t_b is given by Equation 6.3 (Fortescue, 2003),

$$t_b = \frac{m_p}{\dot{m}} \tag{6.3}$$

where m_p is the mass of propellant.

To estimate the burn time required to provide a delta-V of 753m/s using a 20N thruster, Equations 6.2 and 6.3 were combined with the Tsiolkovsky rocket equation (Equation 5.1) to give Equation 6.4

$$t_b = \frac{m_0}{\dot{m}}\left[1 - \exp\left(\frac{-\Delta v}{I_{sp} g_0}\right)\right] \tag{6.4}$$

where m_0 is the initial mass of the spacecraft which was taken to be 120kg.

For a specific impulse of 250 seconds and a thrust of 20N, the burn time to provide a 753m/s increase in velocity was calculated to be 3890 seconds. This is approximately ten percent of the period of the GTO. The burn occurs over such a long period that the spacecraft will travel significantly through the curved path of the orbit at perigee during the burn. For simplicity the spacecraft will be inertially fixed during the burn so that the thrust is along the velocity vector at perigee. This means that for a long burn time, energy will be wasted by thrusting in the wrong direction as the spacecraft travels around the curved path. The situation is shown in Figure 6-1.

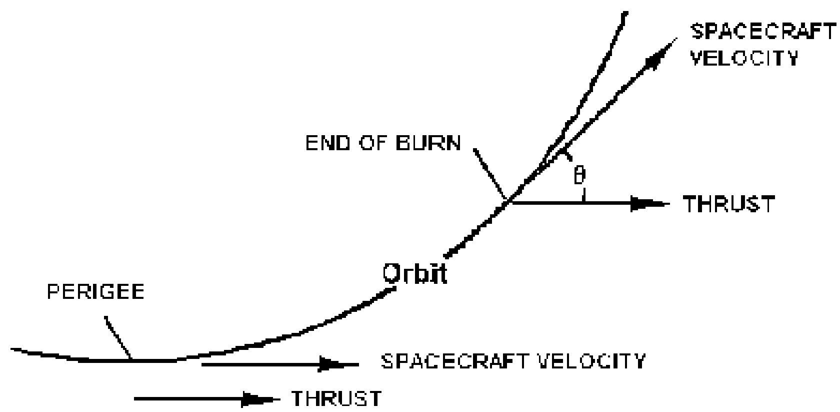

Figure 6-1: Change in velocity vector during finite burn leading to delta-V losses (Brown, 1995)

The wasted energy is referred to as delta-V loss and has the impact that for finite burns, a larger delta-V must be provided than for the theoretical impulsive case. Using one burn of a 20N engine it would not be possible to raise the apogee of the GTO to the correct level without a substantial mass of propellant because the delta-V losses would be too significant.

To limit the delta-V losses, the burn must be reduced in duration by splitting the manoeuvre into several shorter burns. Each burn would increase the apogee of the GTO until the target radius of 1.2 million km was obtained. As the number of burns is increased, the duration of each burn is reduced and the delta-V losses are minimised. Theoretically increasing the number of burns indefinitely, reduces the duration of the burn to near zero making the manoeuvre near impulsive. However, the more burns that are performed, the longer the apogee raising takes and the more the spacecraft will traverse the radiation belts. The complexity of the transfer would also be increased.

6.2. Apogee Raise Analysis using STK

The apogee raising was analysed using Analytical Graphics Inc. STK (Satellite Tool Kit) software package to determine the relationship between the number of perigee burns and the delta-V and propellant mass required. The choice of the specific type of liquid propellant for the propulsion system was also determined using the results of the simulations in STK.

The simulations were performed using 20N engine models of both a bipropellant and a monopropellant system. The bipropellant had a specific impulse of 290 seconds and the monopropellant was 224 seconds. Both engines were run as a pressure regulated system so the specific impulse and thrust were the same regardless of the mass of propellant consumed. The initial mass of the spacecraft was 120kg.

For simplicity of spacecraft control during each burn, the spacecraft was inertially fixed during the manoeuvre so the thrust was along the velocity vector at perigee.

The initial simulations were performed for burns starting at perigee. However the delta-V losses were particularly large for this configuration. The delta-V losses can be reduced by centring each burn on perigee so that the burn takes place an equal amount of time before and after perigee passage. Figure 6-2 shows the apogee raising simulation performed using STK.

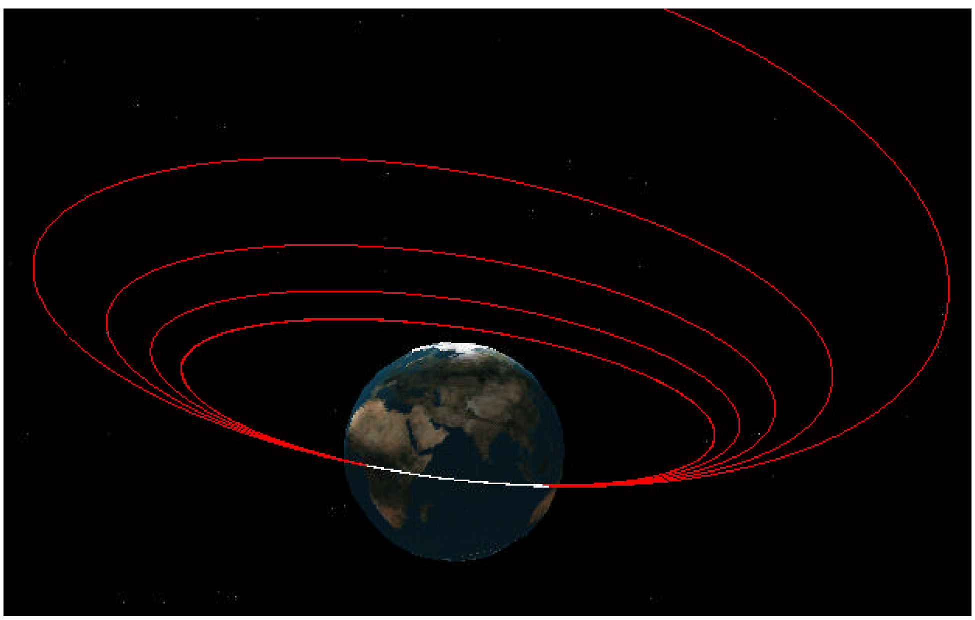

Figure 6-2: Apogee raising sequence simulation using STK

In figure 6-2, the white section behind the Earth shows the burn taking place at perigee. The length of this curved arc is reduced as the number of burns are increased.

6.2.1. Simulation Results

Figure 6-3 and Figure 6-4 summarise the results of the simulations performed in STK. Figure 6-3 shows the delta-V that is required to raise the apogee to 1.2 million km against the number of perigee burns performed.

Each burn was performed for equal duration and the length of each burn was adjusted accordingly until the correct apogee radius was achieved.

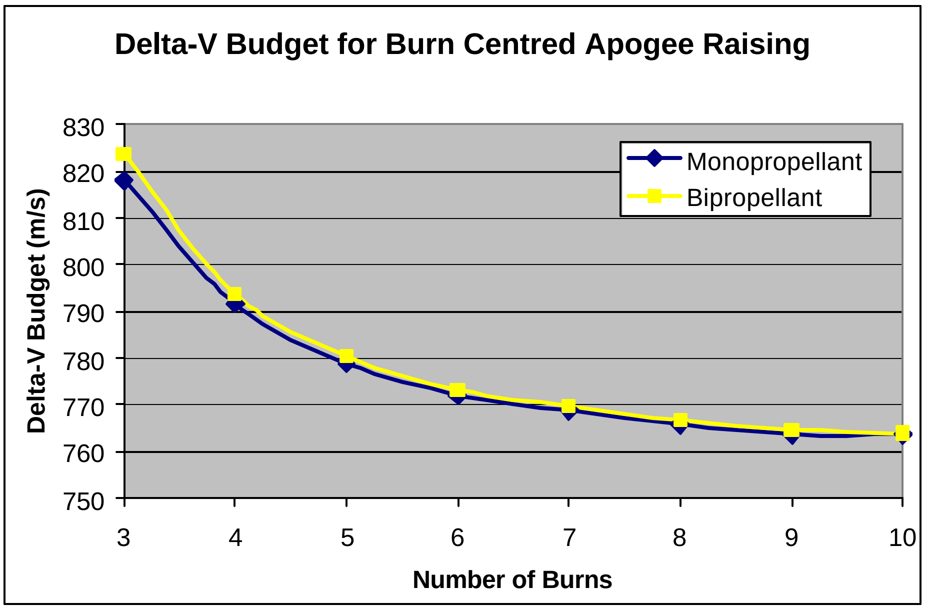

Figure 6-3: Delta-V budget required to raise GTO apogee to 1.2 million km against number of perigee burns performed

The delta-V budget required to raise the apogee of the GTO decreased as the number of burns was increased. The performance of the bipropellant engine was slightly worse in terms of delta-V due to longer burn times of the more energetic propellant for a given thrust. The difference between the two engines however was negligible.

The theoretical impulsive delta-V required to increase the apogee was 753m/s. The delta-V loss for three burns was around 70m/s which reduced to 10m/s as the number of burns were increased to ten.

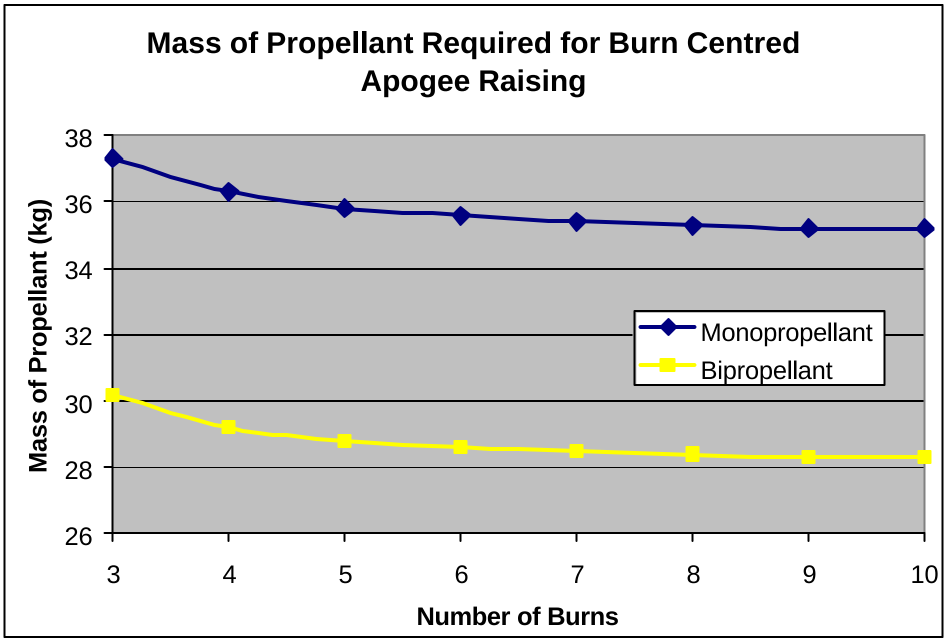

Figure 6-4: Mass of propellant required to raise GTO apogee to 1.2 million km

The mass of propellant required is summarised in figure 6-4. The propellant mass was calculated directly from the delta-V budget using Equation 5.1. The propellant mass required decreased as the number of burns were increased due to the decrease in the delta-V budget. The propellant mass however begins to level off much faster than the delta-V budget with no significant mass saving in performing more than six or seven burns. The bipropellant propulsion system offers a significant mass saving of around 7kg when compared to the mass of the monopropellant.

The simulations were performed using the Earth J2 propagator. The effects on the apogee raising from the Sun and the Moon were ignored. For the early apogee raises this affect from these bodies was minimal, however the later apogee raises would become affected by the presence of the Moon as the apogee radius approached the Moon's orbit. The influence of the Moon would need to be considered for a more detailed analysis of the apogee raising although the approximate model provided reasonable results.

The apogee radius achieved after each burn was different for each of the simulations performed. When each burn is undertaken for the same amount of time, increasing the burn time by one second can substantially increase the apogee radius for a large number of burns because the increased burn time must be added to each of the manoeuvres. The delta-V budget obtained was not representative of raising the apogee to exactly 1.2 million km but instead the apogee reached varied for each case. A larger apogee radius will require a larger delta-V regardless of the number of burns and so the delta-V was not entirely representative of the same apogee radius in each case. However, the difference in the achieved apogee radius was small and the errors introduced by this were minimal.

In reality, it would be possible to alter the individual burn times to target the exact radius required. However, this is difficult to analyse using STK because there are too many possible variations to the duration of each burn required.

6.3. Propulsion System Choice

From the apogee raising simulations, the bipropellant system saved around 7kg in terms of propellant mass. EARTHSHINE will be restricted to a launch mass of just 120kg and so it is important to choose a propulsion system that has the lowest mass. However, despite the propellant mass savings of the bipropellant system, a monopropellant engine was chosen for EARTHSHINE.

The bipropellant system saves propellant mass due to the higher specific impulse however the actual bipropellant system is heavier, more complex and more costly than a monopropellant propulsion system.

A bipropellant engine requires at least two tanks so the fuel and oxidiser can be stored separately. With such a small spacecraft, it is not possible to fit two spherical tanks into the limited dimensions. Instead, several cylindrical tanks are required for the oxidiser (EARTHSHINE, 2003). This increases mass and cost of the system because tanks are expensive and heavy. A monopropellant system requires just one centrally mounted cylindrical tank and a typical example can save over 4kg in mass when compared to the multiple tanks required for the bipropellant system.

Extra components are required for the bipropellant oxidiser part of the system which weigh over 1kg (EARTHSHINE, 2003). The overall monopropellant system including propellant would therefore overall weigh less than 2kg more than the bipropellant system. This is significantly less than the actual difference in propellant mass due to the heavier bipropellant components.

With such restrictive mass constraints, an increase in mass of 2kg is significant however there are other advantages of using the monopropellant system. The monopropellant system is significantly cheaper than the bipropellant with a large variety of flight proven components available. A bipropellant system requires precise pressure control and propellant delivery increasing the complexity of the system over a monopropellant engine which can be operated as a blowdown system with no strict pressure control. Bipropellant fuel also has tighter operating temperature limits than monopropellant hydrazine.

A monopropellant system requires just one propellant tank which can be placed in the centre of the spacecraft. With only one centrally mounted tank, the effect of decreasing propellant mass on the centre of gravity and moment of inertia matrix is simplified when compared to several tanks mounted away from the centre of the spacecraft. One centrally mounted tank could also be used to store the propellant for the attitude control and station keeping requirements.

A possible system to provide all the manoeuvres for EARTHSHINE would be a dual mode propulsion system that uses hydrazine as the fuel along with an oxidiser. The bipropellant would be used for the main apogee raising sequence but then only the hydrazine would be used for the attitude control and station keeping manoeuvres (EARTHSHINE, 2003). This has the advantage of providing the higher specific impulse of the bipropellant for the perigee burns but decreases complexity and associated mass for the other manoeuvres where the high specific impulse is not so important. However, this system still requires multiple tanks, is more costly, more complex and only saves a few kilograms in mass over the monopropellant system.

A monopropellant hydrazine propulsion system was chosen for EARTHSHINE that meets the low cost restrictions on the mission. Hydrazine is the most common monopropellant with a large variety of low cost, flight proven components available.

6.4. Trajectory Analysis

From the STK simulations of the apogee raising a monopropellant engine was chosen to perform six burns of 651 seconds each to raise the apogee to the correct radius. The transfer strategy states that with the apogee increased to 1.2 million km, insertion into a large amplitude halo orbit is possible. Whilst this may be feasible, it will only occur if the spacecraft is in the correct position in the Earth-Sun system for halo orbit to be obtained. The transfer was therefore analysed using STK to investigate the trajectory afte r the final perigee burn had been performed.

6.4.1. Simplified Trajectory Analysis

The initial trajectory analysis was perfumed using STK. This is shown in Figure 6-5.in the rotating coordinates system looking down on the ecliptic with the Sun to the left of the figure and the halo orbit around the L1 point.

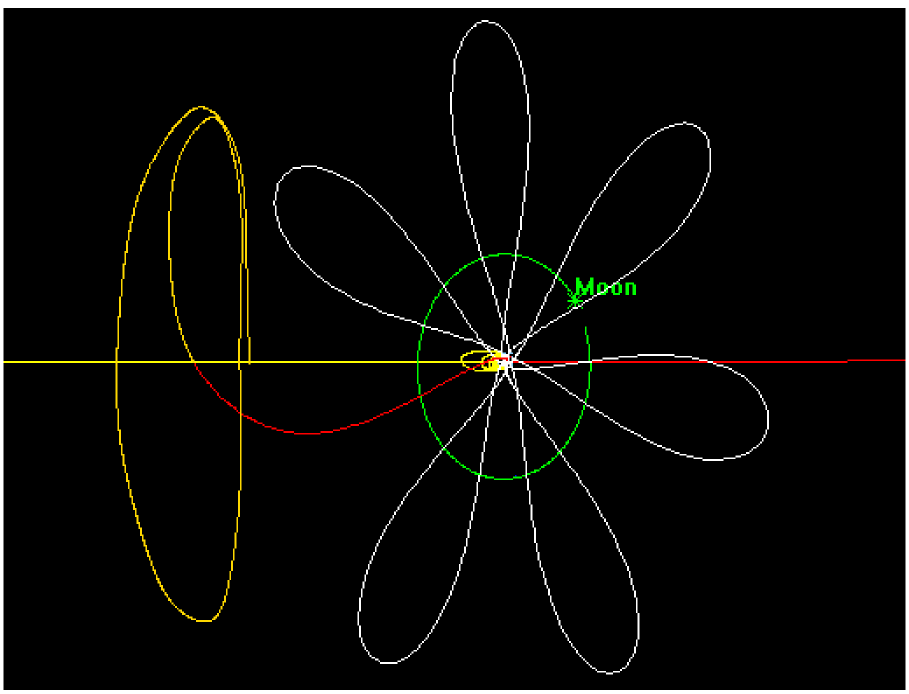

Figure 6-5: Simplified trajectory from GTO to L1 halo orbit shown in rotating coordinate system looking down on the ecliptic with the Sun to the left of the diagram. The Moon's orbit is shown around the Earth

The initial orientation of the GTO was centred on the Earth-Sun line with the apogee facing the Sun and perigee on the night side of the Earth. This was based on the assumption of a midnight Ariane 5 launch that would place the spacecraft into a GTO with similar orientation to this.

The apogee raising sequence took between eight and ten days depending on the number of orbits that were carried out. During this time, the Earth moved only a small distance around the Sun. However, after the final perigee burn the spacecraft took a further 28 days to reach apogee of the next orbit during which time the Earth moved approximately 28 degrees around the Sun in its orbit. The spacecraft was therefore no longer travelling towards L1 by the time it reached the apogee distance. It would be possible to perform a manoeuvre to target the halo orbit directly however this would require propellant which would not be available. Instead, the spacecraft remained in the highly eccentric Earth orbit (petal orbits) for one year as the Earth orbited around the Sun before drifting off towards L1.

For the simulation shown in Figure 6-5, the halo orbit was obtained with an impulsive manoeuvre of 19m/s as the trajectory crossed the Z-X plane (horizontal line in figure). Further impulsive manoeuvres were performed on the Z-X plane crossings to maintain the halo orbit. These manoeuvres were large due to the halo being skewed and not formed on the optimal path. This was because the inclination of the initial GTO had not been changed and so the spacecraft had to be forced to orbit L1 with increasingly large station keeping manoeuvres.

The trajectory analysis was performed using the simplified Earth J2 propagator for all of the transfer until the trajectory drifted in the direction of L1. For this part of the transfer, the Earth Full propagator was used to take the effects of the Sun and Moon into account so the halo orbit could be obtained. This simulation was therefore only used to represent the transfer and a more detailed analysis was performed taking all the perturbations of the Sun and Moon into account for the whole of the transfer.

6.4.2. Detailed Trajectory Analysis

Simulations were performed for a midnight launch on 1st February 2007. This date was chosen to allow suitable construction time for the spacecraft and to tie in with the science objectives of the mission. The trajectory is shown in Figures 6-6 and 6-7.

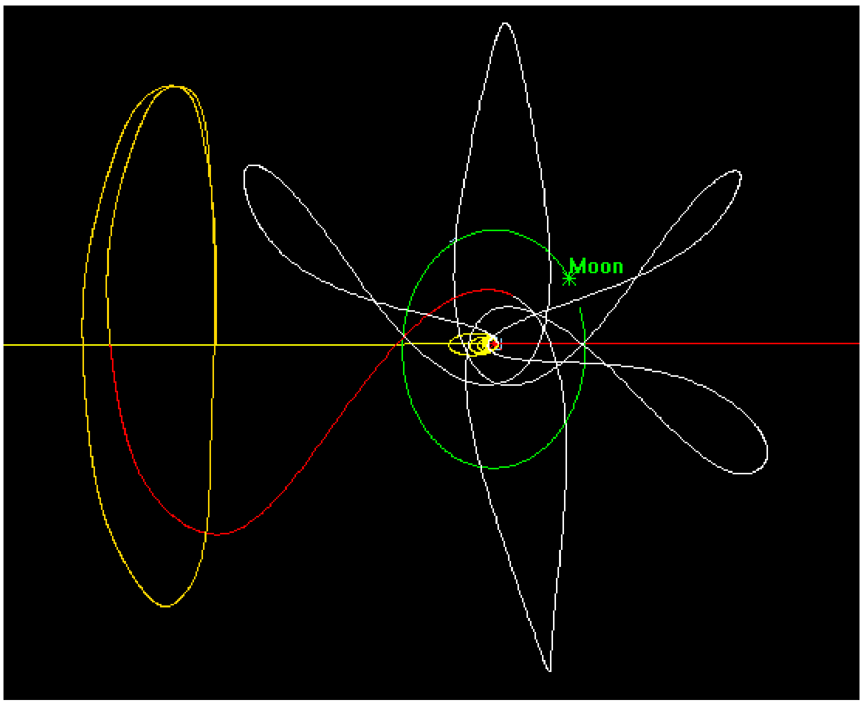

Figure 6-6: Detailed trajectory from GTO to L1 halo orbit shown in rotating coordinates system looking down on the ecliptic with the Sun to the left of the diagram. The Moon's orbit is shown around the Earth

The detailed trajectory analysis was performed using the CisLunar propagator for the apogee raising and the Earth Full propagator for the rest of the transfer. These two propagators consider the effects of the Sun and the Moon on the spacecraft during the transfer with the CisLunar propagator being used specifically for trajectories within the orbit of the Moon.

The initial apogee raising sequence was relatively unaffected by the influence of the Moon. However, after the final perigee burn had been performed, the spacecraft's motion through the petal orbits was strongly affected by the prescience of the Moon.

This affect varied considerably on the actual launch date depending on the position of the moon relative to the spacecrafts position in each orbit.

Due to the influence of the other bodies on the system, the perigee altitude of the petal orbits were raised significantly. The plane of the petal orbits also varied considerably from the initial GTO meaning no manoeuvres were required to actively remove the inclination. The out of plane motions of the petal orbits can be seen in the 3D trajectory shown in Figure 6-7. The number of petal orbits was reduced to five due to the perturbations from the other bodies.

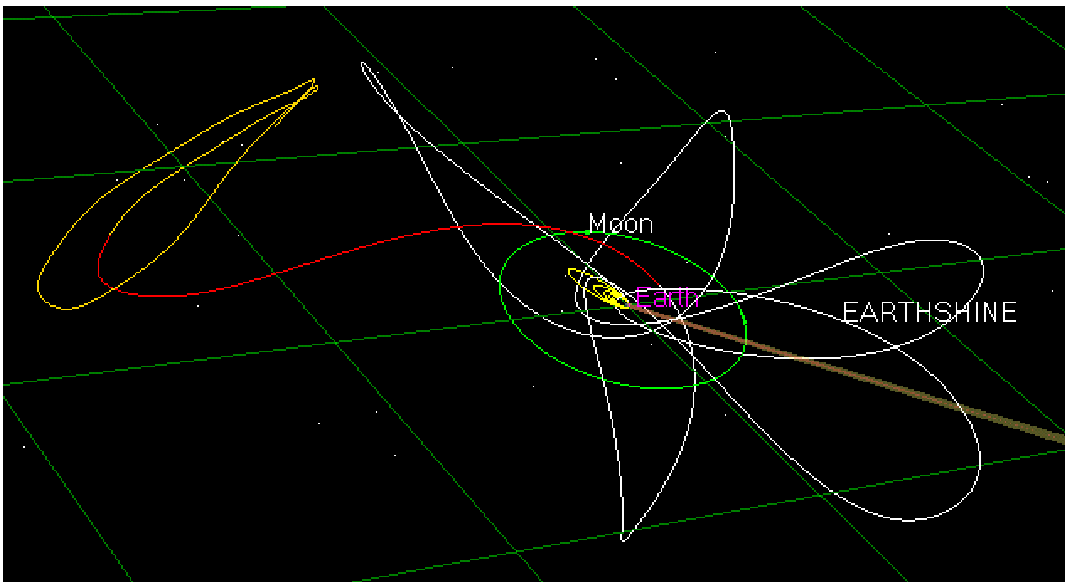

Figure 6-7: Detailed trajectory analysis shown in 3D in the rotating coordinate system

The halo orbit was obtained with an impulsive insertion delta-V of just 4m/s at the Z-X plane crossing. To remain in the halo orbit, impulsive manoeuvres were required on the Z-X plane crossings that cost just 5m/s per orbit. The mission requirement is for four orbits of the L1 point and so the station keeping budget from this particular simulation was just 20m/s for the duration of the mission.

6.5. Summary of EARTHSHINE Transfer

The total delta-V budget for the detailed trajectory analysis was approximately 776m/s. This included 772m/s for the apogee raising and 4m/s for the impulsive insertion manoeuvre. The apogee raising sequence used six burns as was decided from the results of the STK simulations. The first five burns were performed for 651 seconds and the final burn was reduced to 650 seconds. The results of the simulation performed using STK are summarised in Table 6-1 and 6-2. The initial GTO parameters were the same as shown in Section 3.2 with Right Ascension of the Ascending Node (RAAN) set at 318 degrees.

Orbit	Radius of Perigee (km)	Radius of Apogee (km)	Start	End
GTO	6937	42266	31/01/2007 23:59	01/02/2007 21:13
Raise 1	6937	49912	01/02/2007 21:19	02/02/2007 23:34
Raise 2	6938	61185	02/02/2007 23:40	03/02/2007 16:50
Raise 3	6941	79494	03/02/2007 16:55	04/02/2007 17:28
Raise 4	6959	114515	04/02/2007 17:33	06/02/2007 10:23
Raise 5	7095	208316	06/02/2007 10:28	10/02/2007 10:15
Petal 1	131607	1.10E+06	10/02/2007 10:21	14/04/2007 09:08
Petal 2	117027	1.10E+06	14/04/2007 09:08	02/06/2007 17:43
Petal 3	34247	1.00E+06	02/06/2007 17:43	01/08/2007 18:03
Petal 4	104114	1.20E+06	01/08/2007 18:03	03/10/2007 20:59
Petal 5	171611	1.20E+06	03/10/2007 20:59	24/11/2007 18:29
Trajectory Insertion occurs on 26 Jan 2008 06:04:31.05				

Table 6-1: Orbit parameters and transfer time results for the trajectory simulation performed in STK

Burn	Duration (s)	Delta-V (m/s)	Fuel Used (kg)
1	651	111.271	5.927
2	651	117.21	5.927
3	651	123.818	5.927
4	651	131.216	5.927
5	651	139.555	5.927
6	650	148.789	5.918
	Total Delta-V (m/s):	771.859	
	Total Fuel Used (kg):	35.553	

Table 6-2: Properties of each perigee burn for the trajectory simulation performed in STK

The transfer was very sensitive to deviations off the nominal path. To enable a halo orbit to be obtained, the STK simulation was run so that perigee passage of the GTO occurred 11 seconds before midnight. The GTO and first orbit raise were also travelled around twice. If the time of perigee passage was varied by a few seconds, then the halo orbit could still be obtained but only at the expense of a higher insertion delta-V. If any of the burn times were altered then the halo was generally not obtained. On occasions the spacecraft was even pulled toward L2 from the third petal orbit if the timing of the final burn was not as in the ideal case.

Transfer of this type does allow a halo orbit to be obtained for the minimum possible delta-V budget which is desirable to keep the propellant mass low. However the transfer was so sensitive to deviations from the optimal path that careful monitoring of each of the burns would be required. If a burn misfired slightly then the subsequent manoeuvres would have to be adjusted accordingly. This would be possible but may increase the delta-V budget from the ideal case.

The transfer was also very dependent on launch date and initial orientation of the GTO and so the analysis would need to be performed when more information on the launch was known. If the launch was delayed for any reason then it would be likely that the transfer would vary slightly even if the delay was only a few days. To get the timing correct for the minimum delta-V insertion it is possible to travel around the apogee raise orbits more than once to delay the final manoeuvre. This may be required because after the last burn the spacecraft will essentially coast until the halo insertion manoeuvre and so the timing of the final burn is critical to successful halo orbit insertion. The number of burns could also be changed to five or seven to alter the time taken by the apogee raising. If the timing of the final burn is not optimal then the halo may not be obtained. It may be possible to perform orbit shaping manoeuvres during the petal orbits but this would be at the expense of extra propellant being required.

6.6. EARTHSHINE Transfer Propulsion Budget

The delta-V required to raise the apogee of the GTO was determined to be 772m/s. Including an insertion velocity the total delta-V for the transfer could be as low as 776m/s. However if the transfer does not follow the ideal path then the insertion velocity may be higher.

To allow for a larger insertion delta-V or extra manoeuvres that may be needed the delta-V budget was estimated to be 850m/s. This includes a 10% margin. This allows a further 77m/s for insertion or any correction manoeuvres that may be required.

The propellant mass required for the transfer was estimated to be 40.5kg. This was calculated for a specific impulse of 224 seconds using Equation 5.1 and contains a 5% margin to allow for inefficiencies in the propulsion system.

7. Propulsion System Components

There is a large variety of low cost flight proven monopropellant propulsion components available. A 20N monopropellant thruster and hydrazine propellant tank were selected to meet the low cost constraints on the mission.

Table 7-1 summarises some of its key features of the EADS 20N Hydrazine Thruster Model CHT 20.

20 N HYDRAZINE THRUSTER Model CHT 20

Characteristics	
Propellant:	Hydrazine
Inlet Press. Range:	5.5 to 24 bar
Thrust Range vac:	7.9 to 24.6 N
Isp vac:	224 to 228 sec
Total Impulse:	517,000 Ns
Cycle Life:	93,130
Accum. Burn Time:	10.5 hours
Overall length:	195 mm
Nozzle diameter:	33 mm
Mass:	0.395 kg

Table 7-1: Characteristics of suitable 20N monopropellant thruster (EADS, 2001)

The thruster is not strictly a COTS component but is built specifically for each mission. However, the thruster has a substantial flight proven heritage making it suitable for use on EARTHSHINE. The typical cost of a 20 Newton monopropellant thruster is between €70,000 and €100,000 (Bassewitz, 2004)).

The thruster offers a peak specific impulse of 228 seconds which is 4 seconds higher than the engine used to model the transfer in STK. The thrust also peaks at 24.6N which is higher than the simulations although the thrust can also drop to just 7.9N if the inlet pressure is reduced.

Table 7-2 summarises the key features of the PSI Hydrazine Propellant Tank 80274-1.

TANK CHARACTERISTICS (Metrics)			
Operating Pressure, Bar	25.99	Total Volume, l	59.98
Proof Pressure, Bar	39.02	Prop Volume, l	45.03
Cryo Proof	NA	Max Design Wt, Kg	6.01
Burst Pressure, Bar	51.99	Minimum Wall, MM	0.584

Table 7-2: Main characteristics of PSI Tank 80274-1 (PSI, 1999)

The tank has a propellant volume of 45.03 litres. Hydrazine has a density of 1g/cm^3 so the tank has the capacity to carry 45kg of hydrazine propellant. From the propulsion mass budget, 40.5kg are required for the transfer so the tank has capacity left over for station keeping and attitude control requirements. The AOCS propulsion requirements equate to approximately 2kg (Sevcenco, 2004) and station keeping is less than 1kg (20m/s with specific impulse of 224s). The total propellant required is less than the 45 litre capacity but it was decided that a full tank of hydrazine would be used to allow extra manoeuvres to make certain the halo was obtained or for extra station keeping if it was decided to extend the mission lifetime.

The tank operating pressure is higher than the maximum inlet pressure for the thruster however the two components are compatible due to pressure losses between the tank and the thruster. It would also be possible to run the tank at lower pressure if required.

The cost of the tank was estimated to be approximately $200,000 although this could be reduced if the tank was also required by another customer during the same period (Tam, 2004).

7.1. Effects of Blowdown Operation

The propellant tank is designed to be operated in blowdown mode rather than as a pressure regulated system. This means that the tank would be pressurised at the start of life but then would have no active system to maintain the pressure. As the burns are performed the propellant would be used up and the pressure in the tank would decrease. The tank has a total volume 59.98 litres to allow for the pressurant gas volume at the beginning of life.

The blowdown ratio, B of the tank was calculated using Equation 7.1 (Brown, 1995)

$$B = \frac{P_{gi}}{P_{gf}} = \frac{V_{gf}}{V_{gi}} \tag{7.1}$$

where P_{gi} and V_{gi} are the initial gas pressure and initial ullage volume respectively. P_{gf} and V_{gf} are respectively the final gas pressure and final ullage volume. The ullage volume is the volume of gas over the propellant volume which initially would be 14.95 litres for a full capacity of 45.03 litres. The blowdown ratio was calculated to be 4.01 if all of the propellant is consumed. For the use of around 41 litres of propellant for the transfer, the blowdown ratio was calculated to be 3.16.

For the initial operating pressure of 25.99bar, the tank pressure would reduce to just 8.22bar after the transfer stage. The thruster would operate at a specific impulse of 228 seconds and a thrust of approximately 25N when the pressure in the tank was high. However, by the time the pressure reached 8.22bar the thruster would be operating at just 11N (EADS, 2001).

This would significantly affect the performance of the perigee buns. The first perigee burn would be performed at a higher thrust than the simulations but the later burns would be operated at a much lower thrust level. For a smaller thrust, the burn time will be longer and hence the delta-V losses would be increased. This would be balanced out to some extent by the higher specific impulse and thrust of the early burns but the system would overall be likely to offer a worse performance than a pressure regulated system for which the simulations were performed.

An alternative tank could be used and operated as a pressure regulated system. This would enable the use of a smaller diameter tank because there would be no requirement for such a large ullage volume when full. The pressure regulated system however would require an active pressurant system that would add to the complexity, mass and cost of the system. Some of the simplicity of the monopropellant system would be lost if it was operated as a pressure regulated system.

An alternative system could use a tank design that can recharge the pressure of the tank on one occasion. For example, the tank would operate in blowdown mode for three of the burns but then would have the capability to restore the pressure to the original operating level before the tank would operate in blowdown mode again for the final three burns. This tank design would offer some of the advantages of the pressure regulated system without the increase in mass and complexity.

For cost and mass considerations the system would ideally be operated in blowdown mode although analysis would have to be performed to see what the effects of the decreasing thrust of the blowdown operation would be on the system and whether a semi or fully pressure regulated system would be required.

7.2. Propulsion System Design

The monopropellant hydrazine system comprises a centrally mounted tank, one 20N thruster and nominally six 1N thrusters for AOCS requirements (Sevcenco, 2004). The basic configuration of the blowdown system is summarised in Figure 7-1.

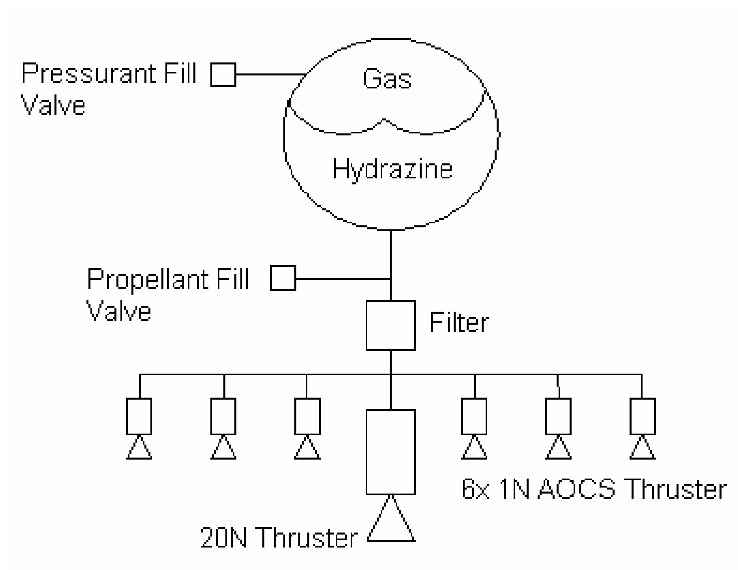

Figure 7-1: Schematic of propulsion system design

For blowdown operation, the tank would be filled with propellant and a pressurising gas before launch. The pressurant must be an inert gas and typically Helium or Nitrogen would be used. Helium offers the lightest system but leakage can be difficult to prevent and so Nitrogen is used if the weight of the system can allow it (Brown, 1995). The six AOCS thrusters and the main 20N thruster operate from the same tank and simply draw the propellant when required.

In addition to the tank and the thrusters, titanium pipework is required with a typical mass of 500g for a small spacecraft. Two fill and drain valves are required with mass of 60g per unit. A propellant filter would also used with a typical mass of 90g per unit.

Propellant latch valves and pressure transducers are also required with masses of 330g and 210g respectively (Oranje, 2004).

The dry mass of the propulsion system was estimated to be 10.44kg including the AOCS thrusters and a system margin on all of the components.

8. Comparison with Industry Proposal

The current design for the EARTHSHINE mission (EARTHSHINE, 2003), has been managed by EADS Astrium. The spacecraft will be launched on Ariane 5 ASAP into GTO although the overall mass of the design is 129.9kg which exceeds the normal 120kg upper mass limit for ASAP. The proposal uses the assumption that the mass could be increased to this level without decreasing launch opportunities significantly.

The mission analysis states two transfer methods for travel from GTO to L1 halo orbit. Following the final perigee burn, the short transfer takes approximately 60 days until halo orbit insertion. The long transfer takes 360 days. The short transfer requires a total delta-V budget of 1190m/s including 5% contingency whilst the delta-V budget for the long transfer is 850m/s.

The long transfer uses essentially the same trajectory as was calculated for this study and has been chosen for the industry proposal due to the lower delta-V requirements. It has also been chosen because it removes any variability in the Ariane launch window. The trajectory analysis performed in STK for this study has shown the transfer to be very sensitive to the orientation of the initial GTO and time of the simulation and so it is unclear why this transfer method can remove the variability in the launch window as has been stated for the industry proposal.

The propulsion system uses a bipropellant system for the apogee raising and monopropellant hydrazine for the attitude control and station keeping. Hydrazine is used as the fuel for the bipropellant functions making the propulsion effectively a dual mode system.

The propulsion system uses one centrally mounted propellant tank and three cylindrical oxidiser tanks. The total dry mass of the propulsion system is 17.9kg and the propellant mass required is 36.2kg. The monopropellant propulsion system chosen for this study has a dry mass of 10.4kg with a propellant mass of 45kg including a large margin to allow for extra manoeuvres. The total mass of the two systems is very similar with the dual mode bipropellant system saving approximately 1.3kg. The monopropellant system however will be less costly and much simpler in operation than the dual system used for the industry proposal.

9. Conclusions

The Ariane 5 ASAP launch vehicle was chosen as the most appropriate launch option for EARTHSHINE. The ASAP facility offers the cheapest launch into Earth orbit with the advantage of less energy required to transfer from GTO than from LEO.

The minimum energy transfer from GTO was demonstrated for a midnight launch on 1st February 2007. Six perigee burns will be performed using a 20N monopropellant thruster to raise the apogee to approximately 1.2 million km. The total delta-V provided by the burns will be 772m/s. The spacecraft will then remain in highly eccentric Earth orbit for 360 days before a small impulsive manoeuvre of 4m/s will be performed to insert into a large amplitude halo orbit.

The trajectory analysis demonstrated a minimum energy transfer strategy from GTO to large amplitude halo orbit. From the simulations it appeared that the transfer was likely to dictate the size of the halo orbit and that insertion into a specific halo orbit would not be possible unless a larger delta-V budget was available. The transfer was found to be very sensitive to deviations from the optimal trajectory raising some concerns as to whether this type of transfer could be used in reality within the limitations of real components.

A monopropellant hydrazine system was chosen to perform the propulsive manoeuvres. This system represented the best low cost propulsion system for EARTHSHINE using commercially available flight proven hardware. The simulations were performed for a pressure regulated system, however for simplicity the system could be operated in blowdown mode which would alter the thrust of the system as the propellant is consumed.

9.1. Further Work

The trajectory analysis would need to be performed in more detail to investigate the effects on the transfer for different launch dates and initial GTO orientation. The trajectory would need to be designed to remove the effects of launch window opening so that if the launch was delayed, the trajectory analysis would not have to be recalculated.

A pointing budget for the perigee burns should be obtained. The spacecraft orientation during each burn will be fixed so that the thrust is along the direction of velocity at perigee. The effect of misalignment of the thrust vector should be investigated to determine the pointing accuracy required for the burns. The situation could also be analysed to determine the effects of a burn misfire so a strategy could be developed to enable the halo orbit to still be obtained.

The halo orbit obtained from the trajectory should be analysed to determine the dimensions of the orbit. The halo orbit design was not strictly part of the Trajectory and Propulsion work package however the trajectory strongly dictates the size of halo orbit obtained and so the halo orbit model would need to be adjusted accordingly. Using STK, the station keeping requirements could be obtained and should be calculated using a finite burn model rather than impulsive manoeuvres which were used for the analysis in this case.

The effects of blowdown operation of the monopropellant system should be investigated carefully to determine how the transfer would be affected. If the decreasing thrust proved too significant that the propellant mass became too high then a pressure regulated system could be considered. However, the use of a pressure regulated monopropellant system would increase the mass and complexity of the system. The bipropellant or dual mode system would then need to be reconsidered to determine whether the monopropellant system still proved to be the most suitable propulsion system. The effects of using a higher thrust engine could also be considered if a suitable model could be found for similar mass, size and cost of the 20N thruster.

The launch options should be investigated further to determine whether there are any other launch vehicles available for EARTHSHINE. Ariane 5 ASAP appears to be the most cost effective launch however, if a direct launch toward L1 could be found then the propulsive requirements and complexity of the transfer would be significantly decreased. If a direct launch is not possible then an alternative launch to Ariane 5 should be considered in case this launch vehicle becomes grounded for any reason. An alternative launch into LEO would probably be available however transfer from LEO requires much more propellant and could not be performed using the current EARTHSHINE configuration. An alternative launch would have to be found into GTO which may not be possible meaning the launch would probably have to be delayed until Ariane 5 was available.

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Appendix A. Executive Summary, EARTHSHINE: Trajectory and Propulsion. Alasdair Helliwell

EARTHSHINE is a proposed deep space mission to be designed and developed exclusively by the UK. The mission will rely on proven small satellite technology and will be constructed primarily using commercial off the shelf components to minimise costs and production time. The mission is proposed to investigate if radiation from the Sun and galactic cosmic rays affect the Earths climate. The proposed mission uses an operational halo orbit around the first Earth-Sun liberation point and the purpose of this study was to investigate suitable low cost methods for transfer to the operational orbit.

A.1. Launch Vehicle Considerations

The most desirable launch for EARTHSHINE would use a dedicated launch vehicle for insertion into the transfer trajectory. The spacecraft would then only require minimal manoeuvres to enable insertion into the halo orbit. However, the use of a dedicated launch vehicle is costly. The most cost effective launch for the low budget mission would be a shared launch into Earth parking orbit.

Launch as a micro satellite on Ariane 5 ASAP was chosen as the most effective launch option for EARTHSHINE. For launch on the ASAP facility, the spacecraft will be limited to a launch mass of 120kg and will be inserted into GTO with apogee facing the Sun and perigee on the night side of the Earth.

A.2. Trajectory Analysis

For transfer from GTO, the strategy that will be employed is to raise the apogee of the GTO to approximately 1.2 million km. From this apogee radius, insertion into a large amplitude halo orbit is possible with minimal insertion delta-V (Hoff, 2002). This transfer strategy therefore offers the minimum energy transfer technique. The impulsive delta-V required to increase the apogee was estimated to be 753m/s.

A.2.1. Apogee Raise Analysis

The delta-V manoeuvres will be provided by an integrated liquid propellant engine limited to just 20N of thrust due to the restrictive mass and dimensions of the spacecraft. A long burn time would be required to raise the apogee using the low thrust engine and so to limit the delta-V losses of the finite burn, the burn will be split into several burns of shorter duration.

The apogee raising sequence was analysed using STK (Satellite Tool Kit) software to determine the optimum number of burns and the mass of propellant required. Figure A-1 shows the mass of propellant required for varying number of perigee burns. The monopropellant specific impulse was 224 seconds and the bipropellant was 290 seconds.

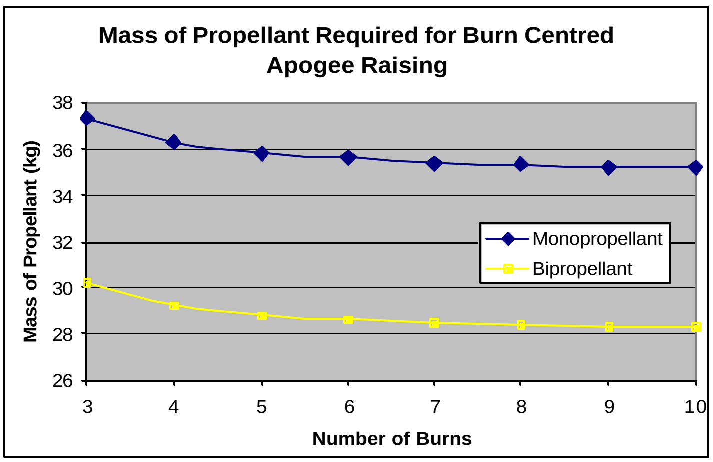

Figure A-A-1: Mass of propellant required to raise apogee of GTO to 1.2 million km

A.2.2. Propulsion System Choice

The use of bipropellant saves approximately 7kg in terms of propellant mass when compared to the monopropellant system. However, the actual bipropellant system is heavier, more complex and more costly than a monopropellant propulsion system. A suitable monopropellant system could save approximately 5kg in terms of component mass over the bipropellant system reducing the mass advantage of the more energetic bipropellant significantly.

A monopropellant system will require just one fuel tank. This could be placed in the centre of the spacecraft, simplifying how the centre of gravity and moment of inertia matrix will vary as the propellant was used up. The bipropellant system requires additional tanks for the oxidiser which can be difficult to accommodate in the limited dimensions of the spacecraft. The slight mass saving of the bipropellant system was not thought to be sufficient to justify its use on the spacecraft and so a simple monopropellant hydrazine propulsion system was chosen for EARTHSHINE.

A.2.3. Detailed Trajectory Analysis

The trajectory was analysed using STK as shown in Figure A-2.

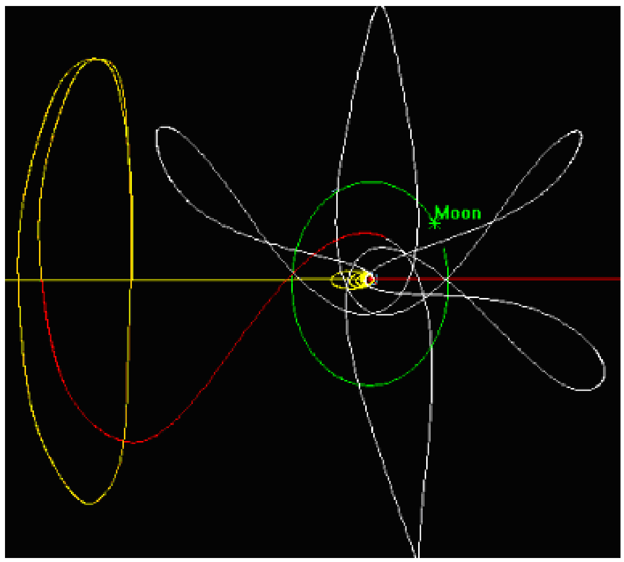

Figure A-A-2: Trajectory from GTO to L1 halo orbit shown in rotating coordinate s system looking down on the ecliptic with the Sun to the left of the diagram. The Moon's orbit is shown around the Earth

The trajectory analysis was performed for a midnight launch on 1st February 2007. The apogee of the GTO will be increased through six burns centred on the perigee of the orbit. The first five burns will be of 651 second duration with the final burn reduced to 650 seconds. After the final burn, the spacecraft will take approximately 28 days to reach the apogee radius of 1.2 million km. During this time the Earth will move around the Sun a significant distance so the spacecraft will arrive in the wrong position for halo orbit insertion. To avoid additional manoeuvres to target the halo orbit, the spacecraft will simply remain in the highly eccentric Earth orbit for ten months as the Earth orbits the Sun, before drifting off in the correct trajectory for halo orbit insertion.

A halo orbit inse rtion manoeuvre of 4m/s will be required on the Z-X plane crossing in the rotating coordinate system to insert into the halo orbit. Further manoeuvres of approximately 5m/s per orbit are required to maintain orbit around L1.

A.3. Delta-V and Propellant Mass Budgets

The delta-V budget for the transfer was calculated to be 776m/s which included 772m/s for the apogee raising and 4m/s for the halo orbit insertion manoeuvre. The delta-V budget for the transfer of the EARTHSHINE mission was determined to be 850m/s including a 10% margin to allow for any extra manoeuvres that may be required for deviations from the optimal trajectory. The hydrazine propellant mass required to provide the delta-V requirements was estimated for a specific impulse of 224 seconds to b e 40.5kg including a 5% margin to allow for inefficiencies in the propulsion system.

A.4. Conclusions and Further Work

A transfer strategy was demonstrated to enable EARTHSHINE to transfer to a large amplitude halo orbit using the minimum delta -V requirements. A reliable, flight proven monopropellant propulsion system was chosen to provide the manoeuvres with a total propellant mass of 40.5kg required for the transfer.

The transfer was found to be very sensitive to deviations from the nominal path and so work would need to be carried out to determine a pointing budget for the buns and to analyse how the trajectory would vary for different launch dates and initial GTO orientation.

The simulations were performed for a pressure regulated monopropellant system. However, for simplicity hydrazine propulsion systems are often operated in blowdown mode where there is no active control of the pressure. As the propellant in the tank is consumed, the pressure in the tank decreases. This has the effect of decreasing the thrust of the engine. The trajectory would need to be analysed for the decreasing thrust to see how the transfer would be affected.

A.5. Reference

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Appendix B. STK Software Simulations

The Astrogator function of the AGI STK software was used to analyse the trajectory. Some of the details regarding how the trajectory analysis was performed for EARTHSHINE are summarised below.

Figure B-1 shows the sequence of Astrogator segments used to construct the trajectory.

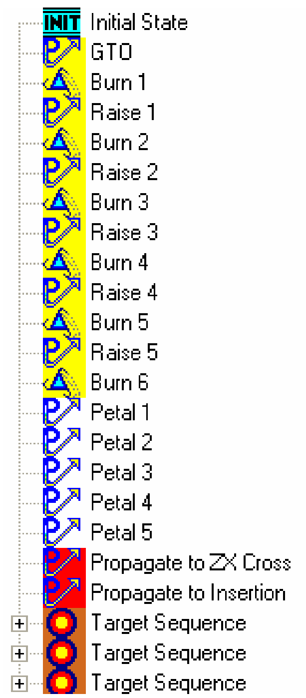


Figure B-1: Sequence of Astrogator segments for trajectory analysis

The 'Initial State' segment defines the initial orbit which is propagated using the 'GTO' Propagate segment. 'Burn 1' is the first apogee raise and 'Raise 1' is the orbit achieved following the burn. The apogee raising takes place through to the 'Burn 6' segment after which the 'Petal' segments are simply Propagate segments to simulate the spacecraft travelling through the petal orbits. The 'Propagate to ZX Cross' and 'Propagate to Insertion' are both the same and are required because the trajectory has to cross the ZX plane near the Earth before insertion occurs at the next ZX plane crossing. The last three 'Target Sequence' segments are required to insert into and remain in the halo orbit.

The 'Initial State' segment properties are summarised in Table B-1.

Segment	Properties
Initial State	Coord System: Earth Centred Mean J2000 Element Type: Modified Keplerian Orbit Epoch: 31 Jan 2007 23:59:49.00 Radius of Periapsis: 6938km Eccentricity: 0.718 Right Asc of Asc Node: 318deg Argument of Periapsis: 178deg True Anomaly: 360deg

Table B-1: Main Properties of Initial State segment

For the apogee raising, the Propagate segments 'GTO' through to 'Raise 6' were all performed using the CisLunar propagator with periapsis as the stopping condition.

Each of the 'Burn' segments were performed in the same way with the final burn reduced in duration slightly. The engine model used was a duplicate of the 'Constant Thrust and I_{sp}' model found in the Astrogator Browser. The engine was given a thrust of 20N and specific impulse of 224 seconds. Table B-2 summarises the properties of the Finite Burn segments.

Segments	Properties
Burn 1 to 6	Attitude Control: Along Velocity Vector Attitude Update: Inertial at Start Stopping Conditions: Duration Centre Burn: selected

Table B-2: Properties of Finite Burn segments

For the Burn segments, the Inertial at Start command for Attitude Update was used to fix the attitude of the spacecraft with thrust direction in the direction of velocity at perigee (the start of the next Propagate segment).

The Petal orbits were Propagate segments and all used the Earth Full propagator with the stopping condition at periapsis.

The ‘Propagate to ZX Plane Cross’ and ‘Propagate to Insertion’ segments both used the Earth Full propagator with the stopping conditions at the ZX plane crossing.

The Target Sequence segments are shown in expanded form in Figure B-2.

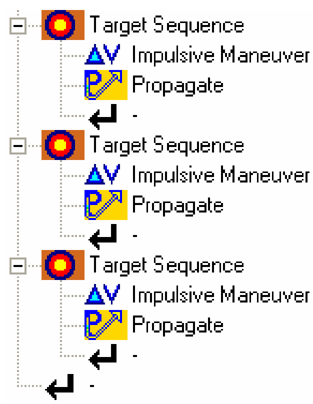


Figure B-2: Expanded Target Sequence segments

The Target Sequence segments were used to target a ZX plane crossing with no velocity in the X direction. This condition was required to insert into and remain in the L1 halo orbit. The Target Sequence calculates the impulsive velocity that is required to meet the target conditions. Table B-3 summarises the properties of the Target Sequences used.

Segment	**Properties**
Impulsive Manoeuvre	Attitude Control: Thrust Vector Thrust axes: VNC Vector Type: Cartesian X (velocity): selected
Propagate	Propagator: Earth Full Stopping Conditions: ZX Plane Cross Results: X component of Velocity in SEM L1 coordinate system Coord System: SEM L1
Target Sequence	Action: Run Targeter Edit Controls and Constraints: Cartesian X and Vx selected

Table B-3: Summary of Target Sequence properties

The SEM L1 coordinate system was set up for the Target Sequence. This was set up by duplicating the 'User Defined Coordinate System' in the Astrogator Browser. The Axes and Origin of the coordinate system were 'SEM L1' available under the MultiBody category.

In the STK scenario the Earth, Sun and Moon were inserted along with the model of the spacecraft. Table B-4 shows the 2D graphic projection settings that were used. The other properties not shown in the table were left blank or were not selected.

Projection	Type: Orthographic Display Coordinate Frame: BBR Display Height: 3e11 km Secondary Body: Sun Lat: 90deg Lon: -90deg

Table B-4: 2D graphic projection settings

Appendix C. Data Sheets

The data sheets of the chosen thruster and fuel tank are contained to provide information on the properties of the components.

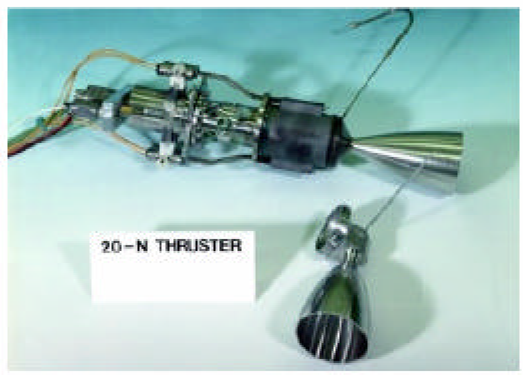
C.1. EADS CHT20N Monopropellant Thruster

CHT20N **20 N Thruster, Monopropellant**

Design

Thruster with shower head injection, catalytic decomposition chamber and expansion nozzle, propellant flow control valve, catalyst bed heater and temperature sensor.

Heritage



Design Heritage

Slight modifications and improvements of the first generation 20 N thrusters flown on Eureca (straight nozzle) and HAPS (nozzle tilted by 90°).

Flight Heritage of current generation

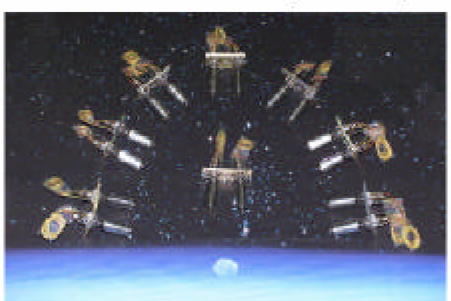
Two versions of the same qualification status are currently available:

[1]: Thruster with a straight nozzle

[2]: Thruster with the nozzle tilted by 90° against the thruster axis

S/C Program	Customer	Flight Units	Year
XMM[2]	ESA	10	1998
Integral[2]	ESA	8	1998
MetOp[1]	ESA/Eumetsat	49*	1999/2000

*: in part still under production



2

CHT20N	20 N Thruster, Monopropellant

Technical Data and Performance

Components:	Supplier	Characteristics
Flow Control Valve:	Manufactured in-house	Model: Solenoid valve with two mechanically independent seats and a single coil Voltage: 28 ± 4 VDC Power: < 13.3 W nominal
Catalyst Bed Heater: (4 per thruster)	Tayco Engineering, Cypress, USA	Model: Cartridge heater with single element Voltage: 28 ± 4 VDC Power: < 3.2 W nominal

Propellant

Monopropellant Grade Hydrazine (N_2H_4)

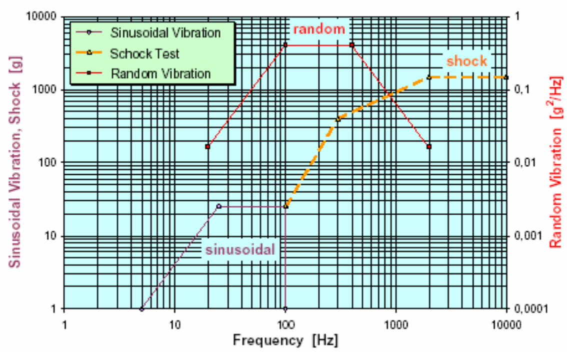
Performance and General Data

Supply Pressure Range:	5.5 bar	to	24 bar
Thrust Range:	7.9 ± 0.5 N	to	24.6 ± 1.0 N
Nominal Mass Flow Range:	3.52 g/s	to	10.22 g/s
Nominal SSF Spec. Impulse Range:	224 s	to	228 s
Minimum Impulse Bit Range:	0.165 Ns	to	0.370 Ns
Nozzle Area Ratio:	60		
Thruster Mass (without wires):	0.395 ± 0.015 kg		
Acceleration Design Load:	25 g		

Date of Qualification: 1997/8

Environmental Loads: (see diagram) 16.2 g_{rms}

Qualified Shock Loads: see diagram



Hot Firing Qualification

Total Impulse:	> 517 kNs
Total Number of Pulses:	> 93132
Total Hydrazine Throughput:	> 290 kg
Total Operating Time:	> 10.5 h
Longest Steady State Burn:	1.5 h
Number of Cold Starts ≤ 20°C:	36
Number of Cold Starts at 0°C:	12

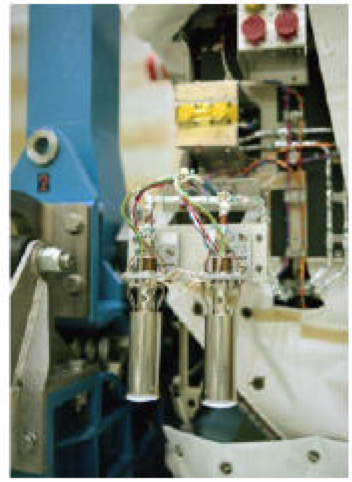
CHT20N 20 N Thruster, Monopropellant

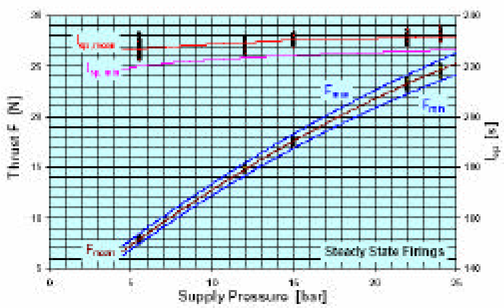
Special Features / Design Description

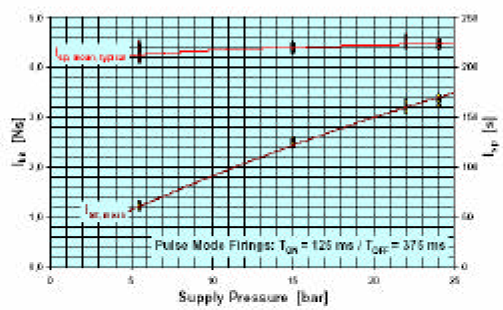
The thruster is equipped with a temperature sensor to measure the decomposition chamber temperature.

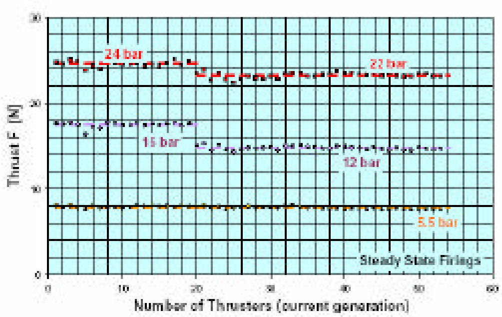
The 20 N thruster in both versions (straight nozzle or nozzle tilted by 90° against the thruster axis) has excellent and reproducible performance, where the mean values of thrust of the straight nozzle version tend to be slightly superior to those of the tilted nozzle version.

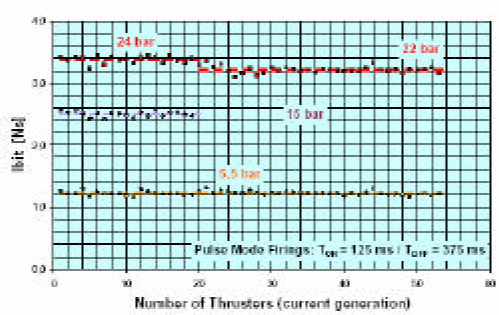
If desired, the thrusters can also be delivered together with brackets holding one or several thrusters, as was done for the HAPS, Eureca, or MetOp projects.



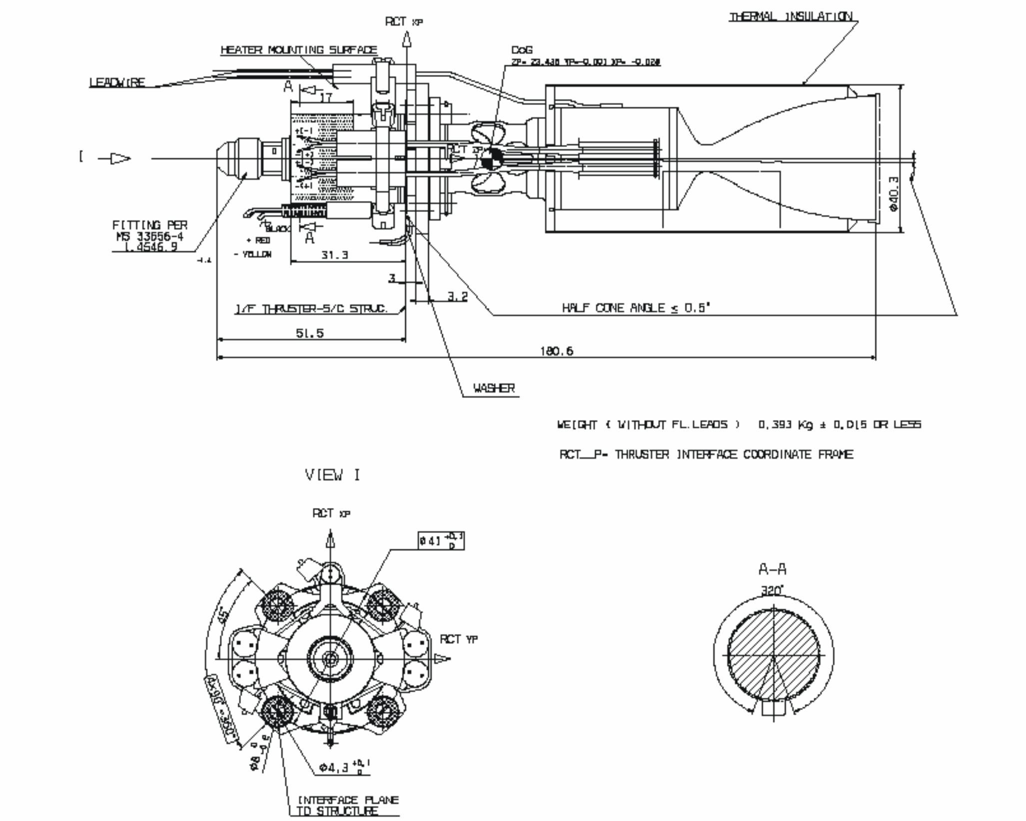




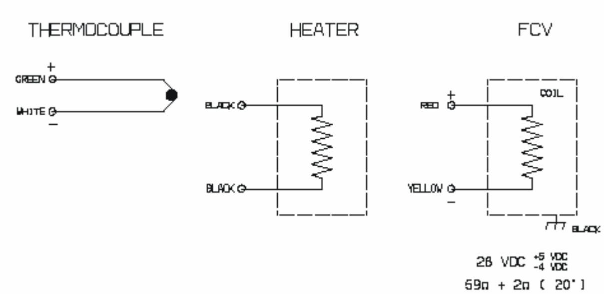




CHT20N **20 N Thruster, Monopropellant**



ELECTRICAL SCHEMATIC



C.2. PSI 80274-1 Monopropellant Hydrazine Tank

VISIT US ON OUR WEBSITE @WWW.PSI-PCI.COM

PRESSURE SYSTEMS INC

TANK TYPE	MOUNT	LOCATION
Diaphragm	Lugs, 2	Girth

This is a 19-inch spherical pressure vessel constructed of 6Al-4V titanium. Positive fuel expulsion is provided by a reversible ethylene-propylene terpolymer (AF-E-332) rubber diaphragm retained (welded in) at the sphere mid-plane. Mounting is accomplished by two (2) lugs parallel with and adjacent to the sphere mid-plane.

ISO 9001 (1994) REGISTERED

PSI Part Number 80274-1

SIZE: 19.06 inch ID Sphere
SIZE: 484-MM

PSI Analysis **Stress**

APPLICABLE DOCUMENTS

Acceptance Test Procedure	50-000217
Qualification Test Procedure	50-000217
Qualification Test Report	56-000078
Traceability	60-000042
Configuration Management Plan	60-000137
QBS Report	60-000138
Powder Blast	65-000030
Fusion Welding, TIG	90-000003
Radiographic Acceptance STD	90-000004
Radiographic Inspection	90-000006
Pre-Weld Cleaning	90-000022
Identification	90-000047
Electrochemical Etch	90-000049
Heat Treat	90-000082
Closed Die Forging	90-000083
Weld, Electron Beam	AMS 2681
Cleaning	CPP 3531

TANK CHARACTERISTICS

Operating Pressure, psig	377	Total Volume, ci	3,660
Proof Pressure, psig	566	Prop Volume, ci	2,748
Cryo Proof, psig	NA	Max Design Wt, lbs	13.25
Burst Pressure, psig	754	Minimum Wall, inch	0.023

TANK CHARACTERISTICS (Metrics)

Operating Pressure, Bar	25.99	Total Volume, l	59.98
Proof Pressure, Bar	39.02	Prop Volume, l	45.03
Cryo Proof	NA	Max Design Wt, Kg	6.01
Burst Pressure, Bar	51.99	Minimum Wall, MM	0.584

ACCEPTANCE
ENVIRONMENTAL TESTS

Random Vibration
Sine Vibration

DIAPHRAGM INFORMATION

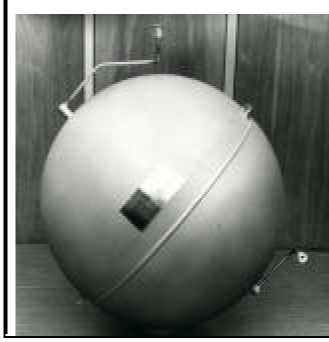
Diaphragm P/N		80-274007-1
Diaphragm Mold P/N		T-1981
Diaphragm Gross Wt		1.71
Diaphragm Matl Type		AF-E-332
Diaphragm, Material	Note 2	90-000075
Diaphragm Processing	Note 2	90-000079
N-Ray Inspection Procedure		1002

FORGINGS

FORGINGS P/N	SUPPLIER	Die No
80-182061-1 (2)	ARCTURUS	2696
RING FORGING	**RING SIZE, (Rough Machined)**	
80-274065-1, Retainer	19.06 +.12 OD x 17.55 -.12 ID x 1.5 +.12 Lg	
80-274063-1, Lug	21.25 +.06 OD x 19.31 -.06 ID x .85 +.06 Lg	

TUBE TYPE AND SIZE

TITANIUM	SIZE
80-274001-3, Outlet	.250 OD x .035 Wall
80-274002-3, Inlet	.250 OD x .035 Wall
	(6.35 x .88 MM)



Notes:

1: Tooling owned by PSI
2: Proprietary Document

PROGRAM INFORMATION

Program	EXOSAT
Customer	Marconi
Customer P/N	3002-42300-25-0
Original Job No	7820
Customer Installed Device	No
Customer Controlled Design	No

Appendix D. Email Correspondence

Appendix D contains some important email correspondence that provides information on the fuel tank and thruster provided by the company concerned. The emails also give indication of the likely cost for the components.

D.1. Information regarding Fuel Tank

The email below was received on 18/02/04 from Walter Tam of PSI regarding information on tank 80274 and possible prices for its construction.

Dear Mr. Helliwell,

You can buy an 80274 tank off the shell. This tank was recently purchased by another company in UK, therefore, all the drawings, documentation, and planning have been updated. I have attached a data sheet of this tank. However, the most recent version is our P/N 80460. It is essentially the same tank but for another customer. Interface dimensions should be almost identical.

Many of our customers want extensive deliverable documents, such as product assurance plan, configuration management plan, etc. that are expensive but do not serve any purpose. If they are requested, we have to charge non-recurring. However, if all you want is an off-the shelf tank without a lot of deliverable documents, the program will be less expensive. You will get an End Item Data Package, of course, and that should meet all your program needs.

Since I don't have a spec., I cannot determine whether the tank is suitable for your use. Please review the data sheet in terms of operating pressure, burst safety factor, propellant volume, mounting features, etc. I hope it will meet your needs.

As for analysis, we can provide you with the environments that the tanks is qualified to. If you provide us with your environments, we can also do a preliminary review for you. If analysis is needed, we will quote the analysis price in non-recurring.

For configuration control purposes, we assign a unique part number for each *individual customer. That means a new drawing, planning, acceptance test procedure, and cleaning procedure. Another option, although we have done it only once, is to ship a tank to you with another customer's identification number. Most aerospace companies don't do this because of their Quality Assurance Systems prohibit it. But if you don't care about tank identification, you can save a few thousand dollars.*

To build one tank will be comparatively expensive because the setup cost is not amortized over multiple tanks. I am currently quoting the same tank to another customer. It will be nightmarish to coordinate the two different and first-time customers, but I can try. If you are truly interested, I can combine yours with this other program, and that will reduce the price. To do this, however, both programs must start at the same time, so that I can buy forgings and do all the machining at the same time. Otherwise, you will have to pay for a single-tank price.

The ROM prices are (in US dollars):

1 tank @ $200K each,
2 tanks @ $165K each,
3 tanks @ $155K each.

Non-recurring prices vary. As I stated previously, it depends on your documentation requirements. I can work with you to determine an acceptable NRE price.

What are your pressurant tank requirements? What is your operating pressure, volume requirement, mounting, etc.?

If you need more information, please don't hesitate to contact me.

Best Regards,
Walter Tam
PSI
(323) 839-3219
waltert@psi-pci.com

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D.2. Information regarding Hydrazine Thruster

The following email was received from Henning Von Bassewitz of EADS Space Transportation on 3/03/2004. The correspondence suggests a typical price for a suitable monopropellant thruster and provided a detailed data sheet which is shown in Appendix C2.

Dear Mr. Helliwell,

Please find attached a copy of our data sheet 20N monoprop thruster CHT 20.
It may not be fully up to date with respect to the number of units we have made, but the performance data are still un-changed.
Prices are, of course, the most sensitive things in our business, since those products are not offered off the shelf. Each customer has some specific requirements and the number of ordered units is of major importance for the unit cost. Before we start with real production and acceptance testing, we have to establish the specific manufacture documentation, to prepare the manufacture and testing jigs and tools, to adjust the machines etc.. That means, we have a fixed block of cost before we have manufactured one single piece part, and then we have the pure unit manufacturing and testing cost.
Worldwide usual price of a 20N mono propellant thruster may be in the range of 70.000€ to 100.000€.
Please note that this is not an offer to you or to the University of Cranfield.

Best regards

Henning von Bassewitz

EADS Space Transportation
Propulsion & Equipment
Business Development

81663 Munich, Germany

Phone: +49-89-607-22148
Fax: +49-89-607-26882
Mobile / Cellular Phone: +49-171-8660592

D.3. Information regarding Propulsion System Components

This email was received on 15/03/2004 from Joost Oranje. The original email was from Henning Von Bassewitz of EADS and summarised the components required for a small satellite propulsion system in addition to the propellant tank and thruster.

Dear Mr. Oranje ,
Thank you again for your interest on our products.

With respect to your system design you should take into account, that we normally do not use high pressure tanks and pressure regulation systems for small S/C. This would have to be payed by too much dry mass. We pressurize the hydrazine tank to some 24 bar and operate the system simply in blow down mode over the whole life. Thrusters may be operated in a blow down pressure range of 4:1. The thrust, of course, goes down in nearly the same ratio, but not the specific impulse, provided you have a good thruster like one of us.

Thus you do not need a pressure regulator.

Make the propulsion system as simple as possible. We do not use redundant components, since all is very reliable.

For pipework you take 1/4 inch pipes, stainless steel or preferably titanium. Typical weight of total titanium pipework in a small S/C: 500 g.

Fill & Drain valves some 60 g / unit

Propellant Filter some 90 g / unit

Propellant Latch valves some 330 g / unit

Pressure Transducer some 210 g / unit
Please do not hesitate to send further questions in case you may have some.

Wee wish you great success for your study.

Best regards

Henning von Bassewitz