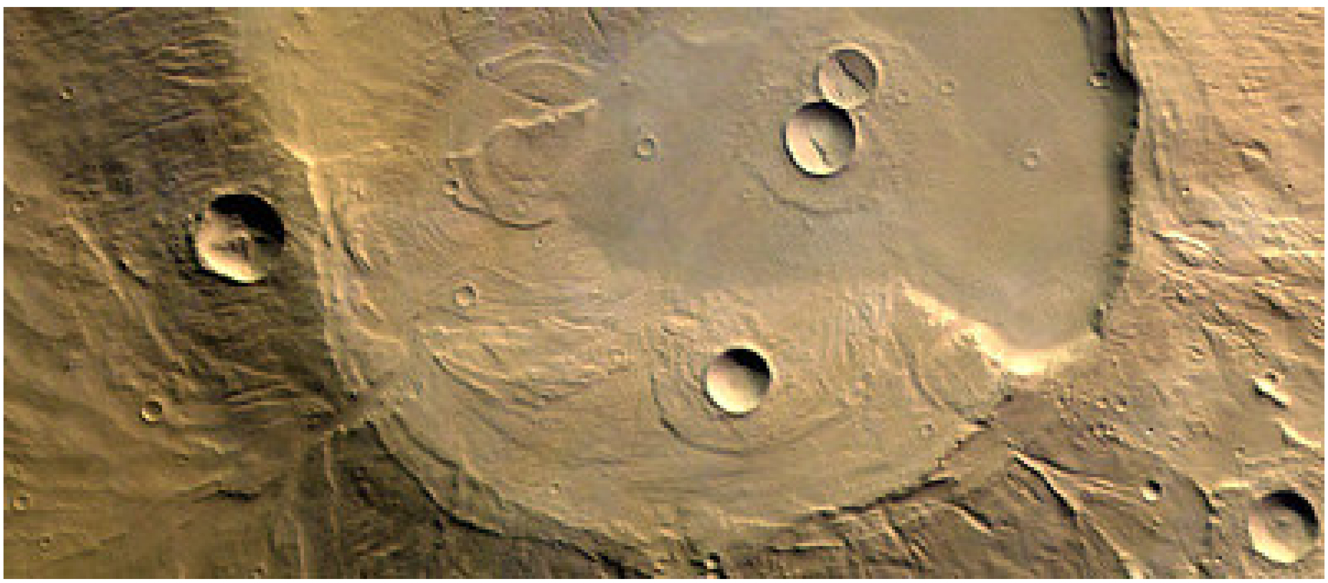
Project Troy

A Strategy for a Mission to Mars



Reaction Engines Limited

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1. Introduction

There is in the USA, Europe and Russia growing government support for an attempt around 2030 to mount a manned mission to the planet Mars.

Mars is the second most accessible astronomical body for human exploration, the Moon of course being the first. Mars is regarded as having great scientific importance, offering increased understanding of planetary sciences, cosmogony, and the origins of life. It is in many respects the most similar planet to the Earth, but sufficiently different to be a critical test of basic theories in these fields. Mars may become the second planet to be a permanent domicile for the human race, but that may lie centuries in the future.

There has been a steady stream of 38 automatic spacecraft to the 'Red Planet' over the past 40 years and a great deal has been learned from these. However a comprehensive expedition can only be cost effective with human explorers, but at a very high total cost. While automatic spacecraft have a low total cost and have therefore been more affordable, the limited data they can gather makes them less cost effective. The automated spacecraft have provided adequate data to now plan a low risk manned mission to the Planet.

There have been many studies of potential Martian missions since the 1950s, beginning with the landmark study by W. von Braun in 1952. This study gave the first indication of the true scale and cost of the undertaking. In the intervening years, government agencies in several countries in conjunction with industry have continued to assess the feasibility of Martian exploration in terms of cost, technology and logistics. In 2004 the USA declared that it has embarked on a structured program involving a return to the Moon in 2018 leading to a mission to Mars around 2030. Planning for the latter however remains vague.

An indication of a possible US approach can be given by consideration of the NASA Design Reference Mission (DRM) study of 1992-93 and its subsequent evolution in 1997. This mission used no low Earth orbit operations or assembly and did not rely on a lunar outpost or other lunar operations. Short transit times to and from Mars and long Mars surface stay times were achieved by using conjunction-class missions. Six crew members ensured an adequate manpower and skills mix.

In the initial mission a heavy-lift rocket capable of launching 240 tons to low Earth orbit was used. In the modified mission, a Shuttle-derived rocket capable of boosting 85 tons into Earth orbit was assumed, thus eliminating the costly large heavy lift rocket development. For comparison, the Saturn V payload was 140 tons, with the proposed Ares V planned to orbit a similar amount.

The mission was split over two launch opportunities. At the first launch opportunity three Mars spacecraft and three nuclear propulsion stages would be launched on six rockets. Each spacecraft would dock with its propulsion stage in Earth orbit and then launch towards Mars. The three spacecraft were a cargo lander, and Earth return vehicle (ERV) orbiter and an unmanned Habitat

Aspects of NASA Design Reference Mission to Mars lander. These spacecraft were estimated to weigh between 60 and 75 tons each. The cargo lander would carry a Mars ascent vehicle (MAV), an in situ resource utilisation propellant factory and hydrogen feedstock and 40 tons of cargo, including a pressurised rover vehicle.

The ERV would aerobrake into Mars orbit, while the cargo lander and Habitat would land on the surface of Mars. The cargo lander would then start to manufacture methane and oxygen for rocket fuel for the MAV and a 600 day cache of life-support consumables.

The crew would follow at the next Mars launch opportunity 26 months later, accompanied by unmanned vehicles supporting the next expedition or providing backup for those already on Mars. This would involve a further six launches.

This time the Habitat would be manned by a six person crew. The second Habitat would land near the unmanned first Habitat. The two Habitats would be moved together and then linked. A stay of 600 days on Mars was planned. During this time the crew would carry out several 10 day rover traverses ranging up to 500 kilometres from the base.

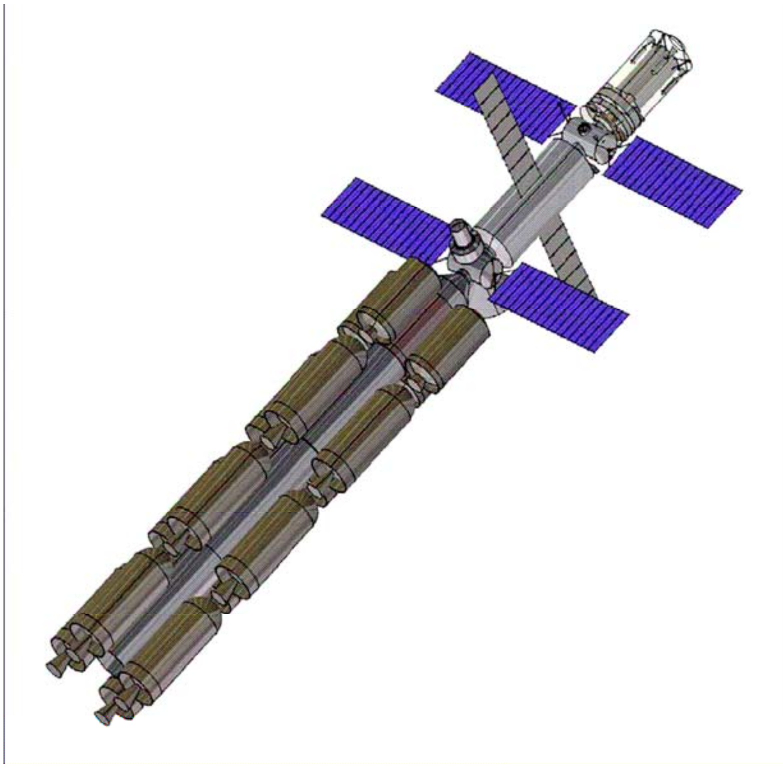
At the end of this stay the crew would lift off from Mars in the MAV and dock in Mars orbit with the ERV. The ERV would then leave Mars orbit for Earth, still docked with the MAV. Near Earth the crew would transfer to the MAV capsule and detach from the ERV which would pass into a solar orbit. The MAV would perform re-entry into Earth's atmosphere and carry out a parachute landing.

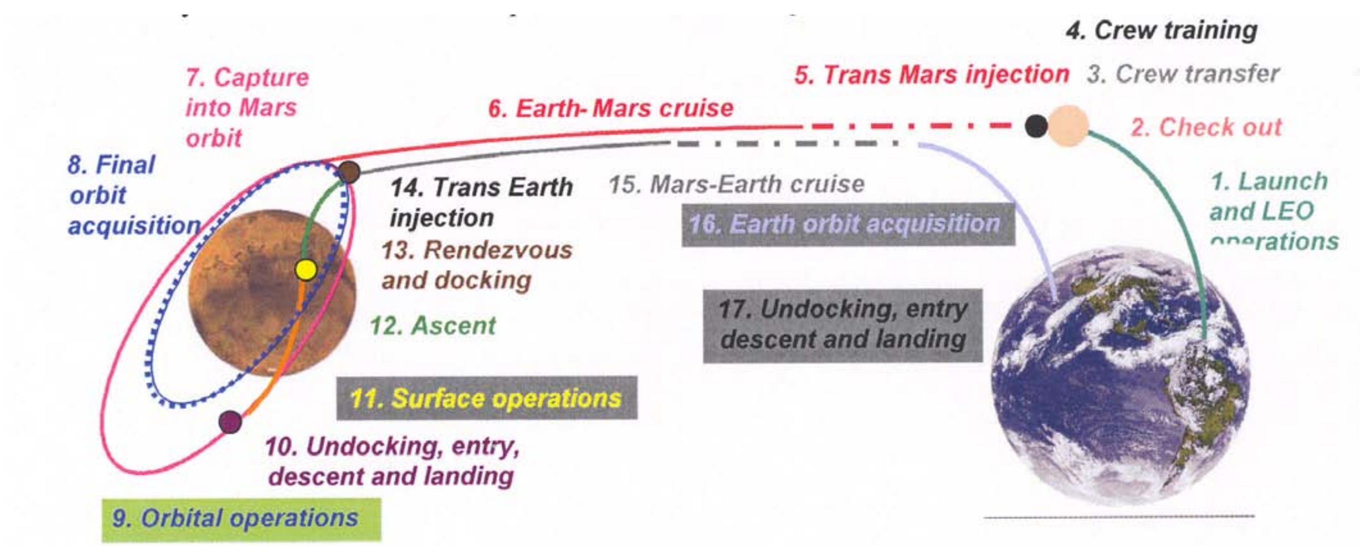
ESA set out ideas for a Mars mission, intended to be the first step in an iteration cycle, in an Overall Architecture Assessment in 2004. In contrast to the NASA mission no previous cargo mission with surface infrastructure or consumables was assumed, and in situ resource utlisation was not considered for either propellant or food. Development of a new heavy-lift launcher is avoided by assuming that the Russian Energia booster can be brought back into production more cheaply.

A single vehicle makes the trip to Mars. The Transfer Habitation Module (THM), which houses the astronauts during their journey and in orbit around Mars, has a mass of 67 tonnes. The forward docking node of the THM is where the Mars Excursion vehicle (MEV) is located on the outward journey. The MEV comprises a Descent Vehicle, a Surface Habitation Module and a Mars Ascent Vehicle. It has a mass of 46.5 tonnes. Aft of the THM is an Apollo-style Earth Re-entry Capsule (ERC) weighing 11.2 tonnes.

A Propulsion Module is attached aft of the THM, divided into three stages. The Trans-Mars injection stage propels the complete vehicle out of low Earth orbit and onto a transfer to Mars. The Mars Orbit Insertion stage lowers the vehicle into orbit on arrival at Mars. The Trans-Earth Injection stage sends the THM and ERC back to Earth at the end of the mission.

The total mass at departure from Earth is 1357 tonnes. Of this, 1083 tonnes is propellant. The ship is assembled in Earth orbit, and a total mass of 1541 tonnes must be placed in orbit for the first mission, including construction platforms which can be re-used for subsequent missions. A





Aspects of ESA Human Missions to Mars mission architecture total of 25 main assembly launches planned for each Mars mission. Two or three more are required to deliver the crew and top up the propellant tanks. At a rate of one launch every two months, the construction phase is expected to take 4.6 years.

The mission is a conjunction class. There is a low energy transfer orbit to Mars lasting about 7 months, a long duration stay of 18 months, a low energy return to Earth of another seven months, for a total of two years and eight months away from Earth.

The design of the Mars Excursion Vehicle is based on the assumption that three of the six crew members descend to the surface of Mars to spend about 30 days there. Seven scientific excursions would be performed, but even with the use of a rover exploration is limited to within 5 kilometres of the lander. These three crew return to the THM and spend the rest of the year and a half at Mars in orbit. The other three crew members do not land at all.

This short surface duration is driven by the desire to select the simplest mission case. A long duration stay would require more resources and infrastructure to support the astronauts on the surface. This would imply an extra cargo mission to take all the extra infrastructure and lead to a requirement for a high precision crewed landing to rendezvous with this cargo ship on the Martian surface.

In all of the recent studies employing current launcher technology two problems are evident, the cost of lifting the mission elements to orbit and the logistical problems associated with the limited flight rate of expendable vehicles from existing facilities. Europe would like to be a participant in the manned exploration of Mars, but in the current financial climate Europe probably cannot afford its ambition in this area. From the information available, independent observers have also questioned whether the USA can.

In this report Reaction Engines has investigated the role that the SKYLON spaceplane might play in rendering a Mars mission more achievable in terms of logistics and cost. In order to examine the problem a SKYLON compatible Mars mission has been studied in some detail to assess the nature of the payloads and traffic flows involved. The study has been given the working title ‘Project Troy’. However, it is stressed that the heart of the study is to examine the role of SKYLON in such an undertaking, the Mars mission being effectively a by-product of that study.

A mission to Mars will be an expensive undertaking with a primary objective to answer some of the deepest scientific and philosophical questions currently being asked by mankind. However, it is clearly desirable that there should also be a practical spin-off from the expenditure of such a large sum of money. The creation of a reusable transportation system which will go on to reduce the cost of space activity by over an order of magnitude long after the Mars missions are achieved would be a suitable legacy from such a laudable undertaking.

2. Project Troy

2.1 Mission Architecture

The architecture described here was the final choice following consideration of several vehicle configurations and propellant choices for the Mars Transfer Stage (MTS) and the Earth Return Stage(s) (ERS). The guidelines for the selection were;

- maximum coverage of Mars
- safety of the crew
- minimisation of cost

This study favoured a two phase mission with an automated phase I (the Precursor) in which equipment, including the surface habitats and power supplies, are delivered to Mars approximately two years prior to the phase II manned mission (the Principal). This strategy enables a working surface base and also orbital facilities to be established and checked out in advance so that an aborted exploration has the maximum chance of survival and a range of predetermined back-up options at its disposal.

The Mission would depart from a low Earth orbit, the Operations Base Orbit (OBO) in which all of the vehicles would be assembled in the vicinity of a dedicated space station, the ‘Operations Base’, which would provide for workforce accommodation, assembly facilities and propellant storage.

Hohmann transfer orbits between the Earth and Mars were selected to maximise the payload and hence the return from the mission. The length of stay at Mars would be of the order of 15 months, together with long coasting periods in space (9 months), as a consequence of this choice. The only practical abort option is therefore to complete the mission, accepting that adequate provisions would already be established in the vicinity of Mars to effect a first opportunity return.

All the stages would employ LO_2/LH_2 propellants. This necessitates careful design of the stages for long term space storage of LH_2. The selection of these propellants minimises the mass to be lifted into Earth orbit. Consideration was given to methane as a more storable fuel, but was rejected due to the large mission mass increase. The large Earth Departure Stage (EDS) is designed to be reusable so that it can be employed to boost both phase I and phase II vehicles. It does this by staging short of escape velocity and passing into a highly elliptical synchronous orbit with the OBO, to which it returns after the boost phase. The final injection of the vehicle into Martian transfer orbit is performed with the MTS engines.

For the duration of the stay on Mars O_2 and CO propellants and reactants would be manufactured from the Martian CO_2 atmosphere using a small nuclear power supply. These would be employed to propel the single stage Ferry vehicle used for transfer from and to Martian orbit and between locations on the surface. They would also be used in solid oxide fuel cells to power surface vehicles for exploration.

The mass of the mission was found to be approximately halved if an aerobraking Earth return was adopted, as opposed to a propulsive capture. Although the latter option would enable some hardware recovery this was not economically attractive compared to the large mass reduction using aerobraking. Atmosphere entry velocities were found to be only slightly higher than for the Apollo missions and so atmospheric assisted capture into low Earth orbit was adopted, with a subsequent rendezvous for recovery of crew and returnables.

The number of vehicles in each phase of the Troy mission is an arbitrary choice, clearly, the fewer the lower the cost. However, having set up the infrastructure, additional vehicles do not increase the mission cost pro rata. Two vehicles would provide great redundancy, while three would enable over 90% of the Martian surface to come within range of exploration if each crew set up base in an optimum position. This would probably require three precursor missions to deliver the full equipment load for each mission. Thus the Troy mission assumes three precursor vehicles and three manned vehicles, the latter having six crew each, giving a total exploration team of 18 people.

This may seem extreme at first sight, but it should be recalled that over a period of 4 years 18 people were placed into the environment of the Moon, 12 of whom went to the surface. This involved the launch of approximately 30,000 tonnes of vehicles.

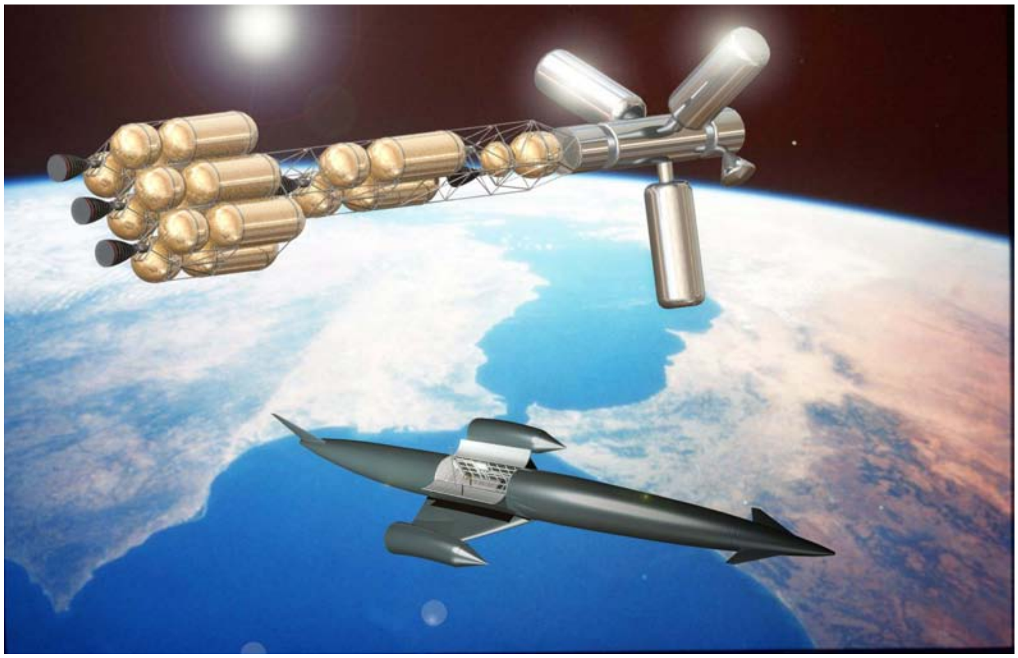
2.2 Mission Dynamics

The dynamical parameters of a mission to Mars are dependent on the particular date of the mission, mainly due to the considerable eccentricity of the Martian orbit (0.093365 compared with 0.016727 for the Earth) which means that although the two planets have similar relative longitudes approximately every two years, the relative distance varies by almost a factor of 2.

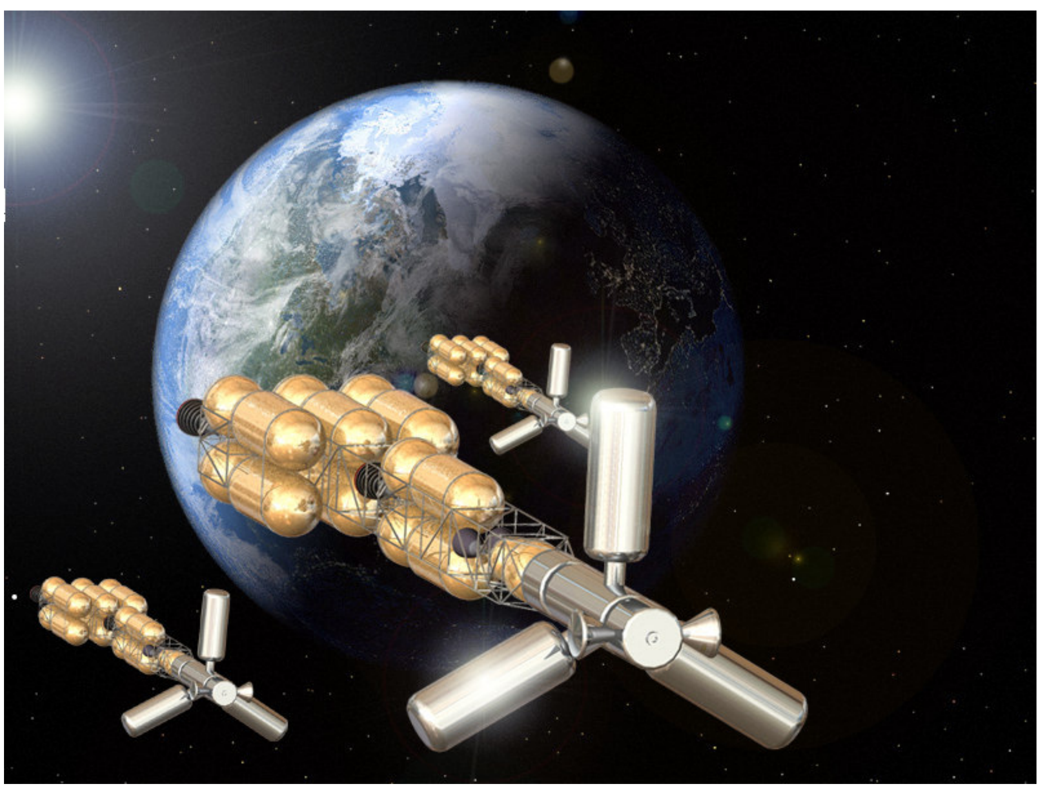
No consideration was given to fast transfer missions since a cursory glance at the vast literature on Mars missions shows that, even with a Venus swing-by, they can only be achieved with current propulsion at the expense of a great increase in launch mass. A perfect Hohmann transfer was assumed and no attempt was made to trade flight time and consumable mass. These are all optimisations to be made by a proper study dedicated to a real mission.

In this study, the manned mission was taken to be launched in 2028, being both the earliest practical and also corresponding with a quiet Sun and (in principle) lower risk of Solar flares. No study was performed of alternative departure dates, since the objective here is not to study a mission to Mars as such, only its feasibility. The precursor mission was therefore taken as the opportunity in 2026, two years before the principal mission.

The leading characteristics of the transfer orbits are shown in Tables 1 & 2



Troy on-orbit assembly



Troy Earth departure

Table 1 Basic Hohmann Transfer Parameters

Phase I *Precursor mission*	**departure date**	**arrival date**	**transfer time (d)**	**departure hyperbolic excess (m/s)**	**arrival hyperbolic excess (m/s)**
Outbound	5Nov2026	27Jul2027	264	3107	2067
Phase II *Principal mission*					
Outbound	6Dec2028	14Aug2029	251	2711	3578
Homebound	8Nov2030	17Aug2031	282	1983	3599

Table 2 Propulsion Requirements

Phase I *Precursor mission*	**Impulsive Injection ΔV (m/s)**	**Impulsive Capture ΔV (m/s)**	**Departure altitude km**	**Capture altitude km**	
Outbound	3620	2397	367	400	
Phase II *Principal mission*					
Outbound	3518	2594	367	400	
Homebound	1801	3759	400	400	

Each vehicle has three stages, an Earth departure stage (EDS) which imparts most of the required escape velocity, a Mars transfer stage (MTS) which adds the remaining Earth escape velocity and carries out the braking manoeuvre into Mars orbit and an Earth return stage (ERS) which performs the escape from Martian orbit and puts the mission on its Earth return trajectory. Recovery of the EDS in low Earth orbit following the boost is complicated by the precession of the Operations Base Orbit due to the equatorial bulge.

The crew capsule is assumed to finally return to a 400km Earth orbit following aerobraking in the Earth's upper atmosphere where a SKYLON then rendezvous with the capsule to retrieve the crew and the payload. Entry into the Earth's atmosphere would be at 11.65km/s at 121.9km. A small circularisation burn would be needed to acquire the 400km orbit.

The OBO was selected to be resonant with the Earths rotation period so that SKYLON missions could be flown on a regular basis with repeating relative positions of the orbital assemblies and the launch sites. The resonance chosen was (46:3) with 46 orbits exactly coinciding with 3 Earth rotations relative to the precessing orbit plane. This requires a OBO at an altitude of approximately 367km, depending on the inclination of the orbit. Here the orbit is assumed to be at 28.5° so as to be efficiently accessible from both Canaveral and Kourou. This was simply an assumption for the purpose of unbiased performance analysis with no implied prediction of the geo-political structure of a Mars mission.

This orbital arrangement would allow a perfect launch opportunity ever 2.93325days, or about 124 missions/year. SKYLON could deploy 10,580kg from Kourou and 10,790kg from Canaveral in SSTO mode. In conjunction with the sub-orbital deployment stage (SODS) it could deploy over 21,000kg from either site. Thus a single SKYLON operating from one site could place over 1300 tonnes into the parking orbit while two vehicles flying from either site at each launch opportunity could place 2600 tonnes into the base orbit in just one year. This is sufficient to assemble the proposed mission in that time, as shown in Section 3.

The orbital dynamics of the OBO is potentially a significant problem. As stated above in relation to the EDS recovery, the nodes of the OBO precess retrograde due to the oblateness of the Earth. This is of the order of 7.2°/day and means that if the departure date is missed by several days the orbit will be in a completely different orientation to that required to inject directly into the correct heliocentric transfer orbit to Mars. This is considered an impractical constraint due to the significant possibility of delays in the departure and an intermediate manoeuvre has been assumed consistent with the EDS recovery dynamics. The geometrical situation shown in Figure 1.

The departure velocity is determined by the hyperbolic departure trajectory and has a direction in space which is very close to the plane of the ecliptic. At any moment in time this vector is at some angle k to the plane of the Operations Base Orbit, the velocity vector of which rotates around the normal vector to the orbit.

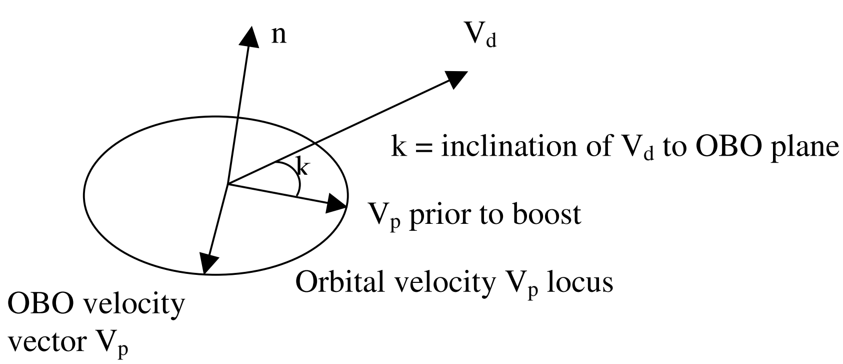


Figure 1 Relationship of departure velocity (V_d) to the OBO orbital velocity (V_p)

Assuming the OBO is at inclination i to the equator and the ecliptic is at inclination i_{ec} (the obliquity) then the relative angle of the OBO to the ecliptic ranges between $|i_{ec} + i|$ and $|i_{ec} - i|$. The maximum value that k can reach is then the greater of these two angles, i.e. $k \leq |i_{ec} + i|$.

The strategy is therefore to ignite the EDS when n, V_p and V_d are coplanar and transfer to a large radius from the Earth (approximately 400,000km), perform a plane change equal to k using the MTS and then add the remaining escape velocity on passage through perigee. The EDS remains in its initial orbit.

By selecting the apogee for the EDS to be resonant with the OBO nodal rotational period, the EDS completes several orbits before returning to the OBO orbit exactly when it can decelerate and return to the Operations Base. Tables 3 & 4 give the parameters for a EDS orbit which completes 5 orbits in nominally 50 days to return to the OBO.

Table 3 Operations Base Orbit Parameters

Circular orbit altitude (km)	367.78
Period (min)	91.82418
Inclination (deg)	28.5
Nodal precession (deg/day)	- 7.20388
Orbital velocity (m/s)	7688.9

Table 4 EDS Orbit Parameters

Apogee altitude (km)	379,300
Perigee altitude (km)	367.78
Period (min)	14,416
Inclination (deg)	28.5
Nodal precession (deg/day)	- 0.011876
Perigee precession (deg/day)	+ 0.019316
Apogee velocity (m/s)	188.4
Perigee velocity (m/s)	10,779.9

On return of the EDS to the OBO the Operations Base is 87 seconds behind the EDS. The EDS therefore enters a 366.78 km x 508.04 km phasing orbit initially to allow the Base to overtake it by this amount. The first braking burn is for ΔV = 3051.3 m/s and the second for 39.7 m/s totalling 3091 m/s. This is also the ideal ΔV for the EDS boost, although because of the unknown thrust-to-weight ratio 5% extra is assumed for the boost gravity losses bringing the EDS departure ΔV to 3246 m/s.

For the worst possible combination of orbital parameters k = 51.95°. At the departure orbit apogee the MTS must add $\Delta V = 2 \times \text{Sin}(25.975) \times 188.4 = 165$ m/s to effect the required plane change. On passage through perigee the MTS adds a further 529 m/s for the precursor or 427 m/s for the principal missions to reach the required transfer orbit injection velocity. These are the worst case values for the 2026 / 2028 Mars opportunities.

The velocity increment summary for the Troy mission is as follows;

EDS departure	3246 m/s
EDS recovery	3091 m/s
MTS 2026 departure	694 m/s
MTS 2027 Mars capture	2397 m/s
MTS 2028 departure	592 m/s
MTS 2029 Mars capture	2594 m/s
ERS 2030 Mars departure	1801 m/s
Earth 2031 capture	3759 m/s (aerobraking)

2.3 Vehicles

The mass ratio for any propulsive part of the mission (event n) is $R_n = \exp\left(\frac{\Delta V_n}{V_{ex}}\right)$ where ΔV_n is the velocity increment for that event and V_{ex} is the effective exhaust velocity of the engines.

The inert mass of a stage is assumed to be given by $M_s = \lambda.M_p$ where sub s,p refer to stage and propellant respectively. M_f , M_i and M_u are the final, initial and useful masses respectively, the latter being the mass carried by the stage, which may be the payload or other stages. These masses are related for each of the three stages by the following relations:-

For the EDS; $\frac{M_u}{M_i} = 1 - \frac{(1+\lambda)(R_1 - 1)}{R_1(1-\lambda(R_2 - 1))}$ where R_1 and R_2 are the mass ratio's for the departure and recovery propulsive events.

The MTS loses mass ΔM between its propulsive burns at Earth departure and Mars arrival due to consumption of consumables.

Hence:- $M_i = \left[M_u + \Delta M\left(\frac{(1+\lambda}{R_2} - \lambda\right)\right]\left[\frac{(1+\lambda}{R_1 R_2} - \lambda\right]^{-1}$ where R_1 and R_2 relate to the departure and arrival propulsive burns respectively.

For the ERS; $\frac{M_u}{M_i} = 1 - \frac{(1+\lambda)(R-1)}{R}$ where now R is the mass ratio appropriate to the total installed stage ΔV.

The **phase II mission** was then derived from the following performance assumptions:-

- The engines of the EDS, MTS and ERS all deliver V_{ex} = 4600 m/s
- the structural factor $\lambda = 0.1$ for all stages
- the M_u injected into Earth transfer of the Phase II mission is 50 tonnes
- At arrival at Mars the Phase II mission is carrying an additional 50 tonnes of equipment and supplies plus 451kg of LH_2 to be lost through boil-off on the ERS.
- All electrical power is provided with $O_2 - H_2$ fuel cells

The analysis begins with the phase II ERS which has the following characteristics;

R	1.4792
M_u (t)	50.000
M_i (t)	**77.684**
M_f (t)	52.516
M_p (t)	25.168
M_s (t)	2.516

From the above assumptions the total M_u delivered to Mars by the MTS is:-

Earth Return Stage (t)	77.684
additional payload (t)	50.000
ERS LH_2 boil-off (t)	0.451
Total Mass (t)	**128.135**

The mass loss between MTS burns is estimated to be as follows:-

food (t)	1.098
oxygen (t)	1.524
LiOH (t)	1.831
fuel cell reactants (t)	8.870
ERS LH_2 boil-off (t)	0.252
Total Mass loss ΔM	**13.575**

From the ΔV's at Earth departure and Mars capture the MTS mass breakdown is:-

R_1	1.13734
R_2	1.75754
M_u (t)	128.135
ΔM (t)	13.575
M_i (t)	**300.410**
M_p (t)	144.273
M_s (t)	14.427

The EDS characteristics follow from the departure and recovery ΔV's:-

R_1	2.02517
R_2	1.95806
M_u (t)	300.41
M_i (t)	**781.985**
M_p (t)	437.795
M_s (t)	43.780

The EDS has a total of 437.795 tonnes of propellants, and the MTS a total of 144.273 tonnes. These are very close to a 3:1 ratio, especially allowing for an estimated 762 kg of LH_2 boil-off (which contributes to the fuel cell reactants in the outward journey). The EDS and MTS use standard modules comprised by two hydrogen tanks, two oxygen tanks and a high area ratio SSME engine. The EDS uses three of these modules and the MTS just a single one. The module tank dimensions are shown in Figure 2 The ERS uses a different size of tankage, also shown in Figure 2, with two high area ratio RL10-B2 derivative engines. All the tanks are insulated with multi-layer insulation with the hydrogen boil-off passing through intermediate screens to control the heat leak. The oxygen tanks are cooled by active heat transfer to the hydrogen tanks.

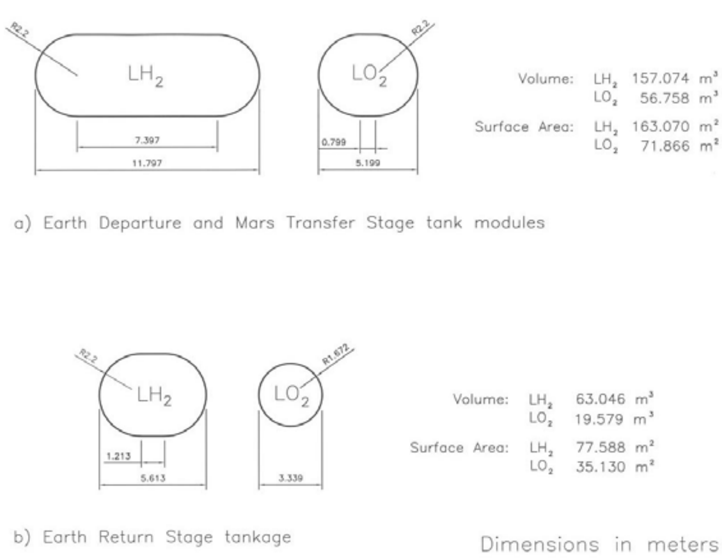


Figure 2 Nominal propellant tank dimensions

The **phase I Precursor mission** employs the same modules as the EDS and MTS for the phase II mission, and it is intended that the EDS is in fact the same stage reused. The mass breakdown of the Precursor is given in Table 5. The tankage dimensions in Figure 2 have been selected so as to accommodate the propellant requirements of both the Precursor and the Principal missions.

The phase II mission overall mass breakdown is estimated to be as summarised in Table 6.

A breakdown of the various stage parameters is given in Table 7.

Table 5 Phase I (Precursor) Troy Mission Mass Breakdown

Event	**Mass (tonnes)**	**Mass (tonnes)**
Departure mass	781.985	
ΔV = 3246 m/s MECO mass	386.134	
Propellant consumed V_{ex} =4600 m/s		395.851
EDS separated mass		85.724
MTS ignition mass	300.410	
ΔV = 694 m/s MECO mass	258.341	
Propellant consumed V_{ex} =4600 m/s		42.069
264 day transfer consumables		
2kW(e) fuel cell reactants		3.379
MTS excess LH_2 boil-off		0.387
MTS ignition mass	254.575	
ΔV = 2397 m/s MECO mass	151.185	
Propellant consumed V_{ex} =4600 m/s		103.390
MTS structure		14.427
Payload in Martian Orbit	136.758	

Table 6 Complete Phase II (Principal) Troy Mission Mass Breakdown

Event	Mass (tonnes)	Mass (tonnes)
Departure mass	781.985	
ΔV = 3246 m/s MECO mass	386.134	
Propellant consumed V_{ex} =4600 m/s		395.851
EDS separated mass		85.724
MTS ignition mass	300.410	
ΔV = 592 m/s MECO mass	264.133	
Propellant consumed V_{ex} =4600 m/s		36.277
252 day transfer consumables		
food		1.098
oxygen		1.524
LiOH		1.831
5kW(e) fuel cell reactants		8.870
ERS LH_2 boil-off		0.252
MTS ignition mass	250.558	
ΔV = 2594 m/s MECO mass	142.562	
Propellant consumed V_{ex} =4600 m/s		107.996
MTS structure		14.427
ERS mass	128.135	
451 day stay consumables		
food		2.292
oxygen		3.185
LiOH		3.822
1kW(e) ERS fuel cell reactants		3.175
ERS LH_2 boil-off		0.451
Martian surface equipment		37.526
ERS ignition mass	77.684	
ΔV = 1801 m/s MECO mass	52.516	
Propellant consumed V_{ex} =4600 m/s		25.168
ERS structure		2.516
Mars departure payload	50.000	
282 day transfer consumables		
food		1.228
oxygen		1.707
LiOH		2.049
5kW(e) fuel cell reactants		9.926
Space habitat		20
Atmospheric entry mass	15.090	
circularising ΔV 102.4 m/s MECO mass	14.317	
Propellant consumed V_{ex} =4600 m/s		0.773
Crew (6)	0.450	
Space suits (7)	1.540	
Water (recycled)	3.000	
Capsules(2) and returnables	9.327	

Table 7 Stage Characteristics as used on Phase II Principal Mission

Parameter ⇓ Stage ⇒	**ERS**	**MTS**	**EDS**
Useable propellant (t)	**25.168**	**144.273**	**437.795**
Useable LO_2 (t)	21.573	123.663	375.253
Useable LH_2 (t)	3.595	20.610	62.542
Boil off (t)	0.702	0.762	0.150
LO_2 residuals (1.5%)	0.324	1.855	5.629
LH_2 residuals (1.5%)	0.054	0.309	0.938
LO_2 tank volume (3% ullage) (m^3)	19.578	113.516	340.546
LH_2 tank volume (3% ullage) (m^3)	63.044	314.147	942.444
LO_2 tank surface area (m^2)	35.130	143.732	431.196
LH_2 tank surface area (m^2)	77.588	326.140	978.420
Tank wall thickness (mm)	0.825	0.825	0.825
Target stage burnout mass (t)	**2.516**	**14.427**	**43.780**
LO_2 tank mass	0.166	0.681	2.042
LH_2 tank mass	0.367	1.544	4.633
LO_2 tank insulation mass	0.176	0.719	2.156
LH_2 tank insulation mass	0.388	1.631	4.892
Engines (extended nozzles)	2 x RL10-B2+	1 x SSME+	3 x SSME+
Thrust (MN)	0.2224	2.135	6.405
Total engine mass (t)	0.570	3.349	10.047
Stage accounted mass +residuals (t)	2.045	10.088	30.487
Contingency factor (target mass/accounted mass)	1.230	1.430	1.436
burn duration (s)	521 (total)	311 (total)	284 (boost) + 30 (de-boost)
Vehicle acceleration range (m/s^2)	2.863 – 4.235	7.107 – 14.976	8.191 – 16.588

Notes: Tank wall aluminium alloy, working stress 400MPa @ 1.5 bar, density 2870kg/m3, Engineering factor2.0. Insulation areal density5kg/m^2. LO_2 density1152kg/m^3, vap pressure 0.8012bar @ 88K, LH_2 density 71.086 kg/m3 vap pressure 0.9352bar @ 20K. Propellant mixture ratio 6:1. Fuel cells produce 13.5MJ/kg reactants @ 298K. MTS H2 boil-off used in fuel cells.EDS boil-off in stage mass estimate.

2.4 Payloads

The full detail of the exploration payload is outside the scope of this paper except in that if SKYLON is to play a role it must be able to launch the equipment into low Earth orbit. More detail will be found in Annex 1 of the assumptions used here.

The payload of the precursor mission consists of all of the surface equipment and the transfer ferries. The available payload per vehicle is 136.8 tonnes and is considered to cover;

Surface habitat	30t
Nuclear power supply	10t
Propellant factory	15t
Two trucks	2 x 5t
Ferry (fuelled)	50t
Stores & equipment	19.8t
Assembly robot	2t

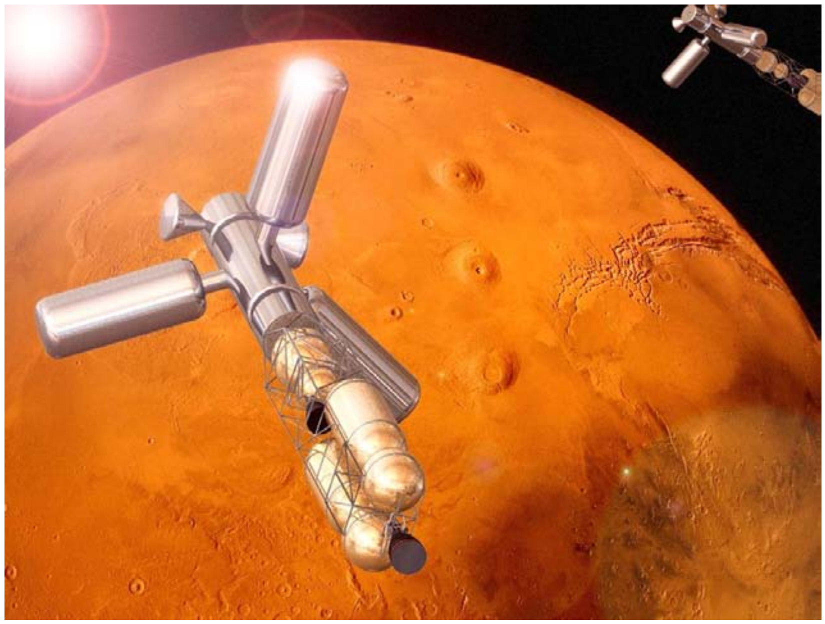
The stores include 2.3t of food, 3.2t of oxygen and 3.8t LiOH for scrubbing CO_2. The above masses include heat shields and landing equipment.

All of this, except the ferry will be deployed to the Martian surface and remotely activated and checked out prior to the launch of the Principal mission. The fully fuelled ferry will wait in orbit for the crew to arrive. The ferry carries sufficient propellant to immediately abort back to orbit should the stay on the surface prove untenable. Each mission vehicle, possibly three in total, would be identical.

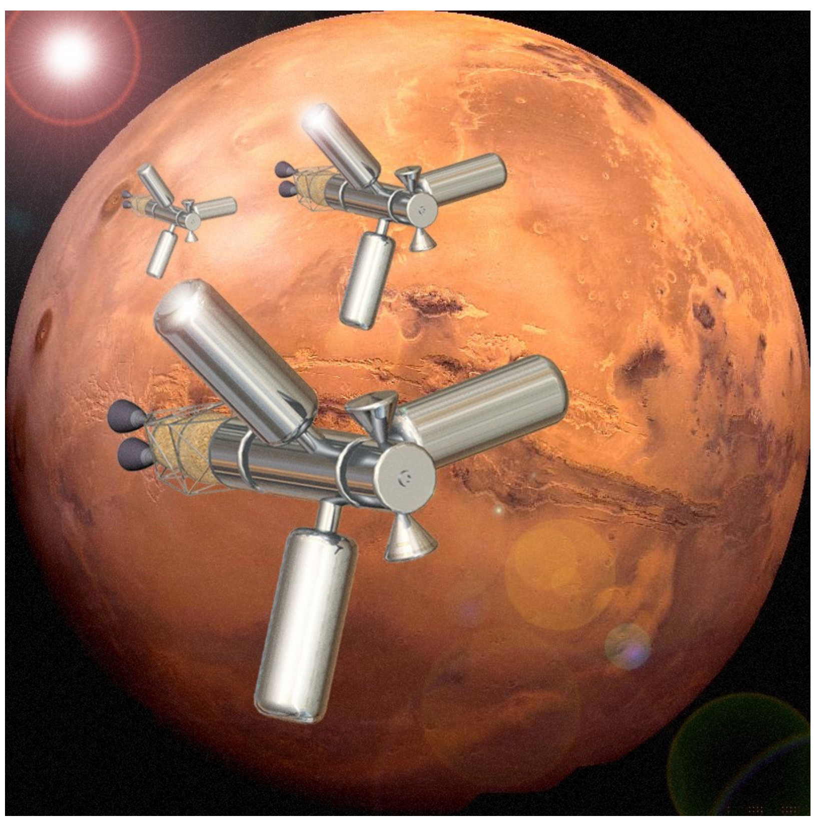
The payload of the Principal mission consists of 6 crew per vehicle, again possibly three in total. The main mass of the payload is the Space Habitat and the two Earth aerobraking capsules. The mission also carries 37.5t of equipment and stores to remain in the Martian environment. This will be selected such that it aids survival in orbit until the Earth transfer window opens, should a landing prove impossible. The breakdown might be;

Space Habitat	20t
Two capsules	2 x 4t
6 astronauts	0.45t
7 space suits	1.54t
Consumables & water	43.7t
Surface equipment & supplies	37.5t

This payload strategy makes available a considerable range of options for crew survival should accident, sickness or equipment failure strike the mission, with supplies available in orbit and on the surface capable of sustaining them until, either Earth return is possible or, a rescue mission arrives at the subsequent opportunity. If the three vehicle mission is adopted there should be the potential for two ERS's to return with all three crews and three entry capsules each, even if at the expense of the returnables. It may be prudent to increase the size and propellant loading of the ERS by 10 tonnes and reduce the surface equipment and supplies to 27.5 tonnes to ccover this eventuality.



Troy Mars arrival



Troy Mars departure

3 Construction Logistics

3.1 Lift Mass Summary

For the phase I mission it will be necessary to lift for each vehicle;

Liquid oxygen	510t
Liquid hydrogen	86t
Stage related hardware	49.3t
Mars payload (+propellants)	136.8t

The Phase II mission will require to lift for each vehicle;

Liquid oxygen	548t
Liquid hydrogen	91t
Stage related hardware	51.5t
Stores, equipment & crew	91.3t

However, in this case over 37 tonnes will be saved if the EDS is reused from the Phase I mission. If three vehicles are used in each mission they will, in each case, be identical and the lifted mass will increase to a total of 2346t for Phase I and 2234t for phase II allowing for stage reuse. For the propellant supply there will also be the parasitic lift of the propellant tanks in which it is carried, which could add a further 10% to the apparent propellant lift mass.

3.2 Launch Requirements

Apart from launch mass, the payload also must fit the SKYLON payload bay envelope of 4.6m diameter x 12.3m long and the process of calculating the number of launches required is determined by considering the combination of component envelopes and masses. However, from the previous section it is seen that about 70% of the lift mass is propellants and this will account for most of the launches and be determined by payload mass alone.

From the above considerations the number of flights is estimated per mission vehicle to be as follows;

Phase I

Item lifted	Number of flights
8 x EDS & MTS hydrogen tanks	8
8 x EDS & MTS oxygen tanks	4
4 x SSME's	2
Mars equipment	15
Liquid oxygen	53
Liquid hydrogen	9
Total number of flights	**91**

Phase II

Item lifted	Number of flights
2 x MTS hydrogen tanks	2
2 x MTS oxygen tanks	1
both ERS tanks	1
1 x SSME + 2 x RL10-B2	1
Payload & Mars equipment	10
Liquid oxygen	58
Liquid hydrogen	10
Total number of flights	**83**

For three vehicles in each mission the number of flights is 273 for Phase I and 249 for Phase II. Of this total of 522 flights, 390 are for propellant representing almost 75% of the total.

The structure of all stages and the in-space habitat present no major problems in being designed to be lifted to the operations base orbit by SKYLON. However it is evident from the preliminary studies in Annex A.1 that the equipment to be placed on the Martian surface (habitat, propellant factory, ferries etc.) need to be of larger dimensions than will fit as a single item into the payload bay in order to accommodate the atmospheric entry shields. These items will therefore need to be modular and assembled in Earth orbit.

It will be necessary to lift the operations base and development test hardware also.

3.3 Assembly Timescale & Cost

As discussed in section 2.2 the resonant base orbit allows 124 ideal flight opportunities per year from each launch site, which could be Kourou or Canaveral or both. At each opportunity, more than one vehicle could be launched from each site, especially once propellant loading has begun. It is therefore conceivable that the whole preparation for each mission with three vehicles each could be easily completed within 12 months provided that the activity on the ground and in orbit had been well planned in advance.

A possible flow chart of the whole TROY project and the development of the SKYLON vehicle is shown in fig. 3. Also given for comparison is the currently known tmescale for NASA's 'Return to the Moon' using the Ares launchers and the Orion spacecraft. Beyond 2026 the EDS and other Martian hardware could play a role in maintaining lunar activity.

Evaluating the actual cost of the Troy mission is very difficult, although reasonably accurate costs can be attempted for some of the mission elements. The following costs are all in Jan 2004 prices, that being the last update of the REL costing model.

Launch Costs

Item	**Cost: $million (US)**
SKYLON development	11,705
4 production vehicles (max 800 flights)	1,796
522 program launches	4,447
10 development launches	85
30 Operations Base launches	255
Total launch related cost	**18,288**

Total Troy Vehicle Costs (Phase I + Phase II)

Item (for 3 vehicles/phase)	**Cost: $million (US)**
EDS/MTS module development	6,084
EDS/MTS module production (19 items)	1,233
ERS development	1,247
ERS production (7 items)	145
Total propulsion hardware cost	**8,709**

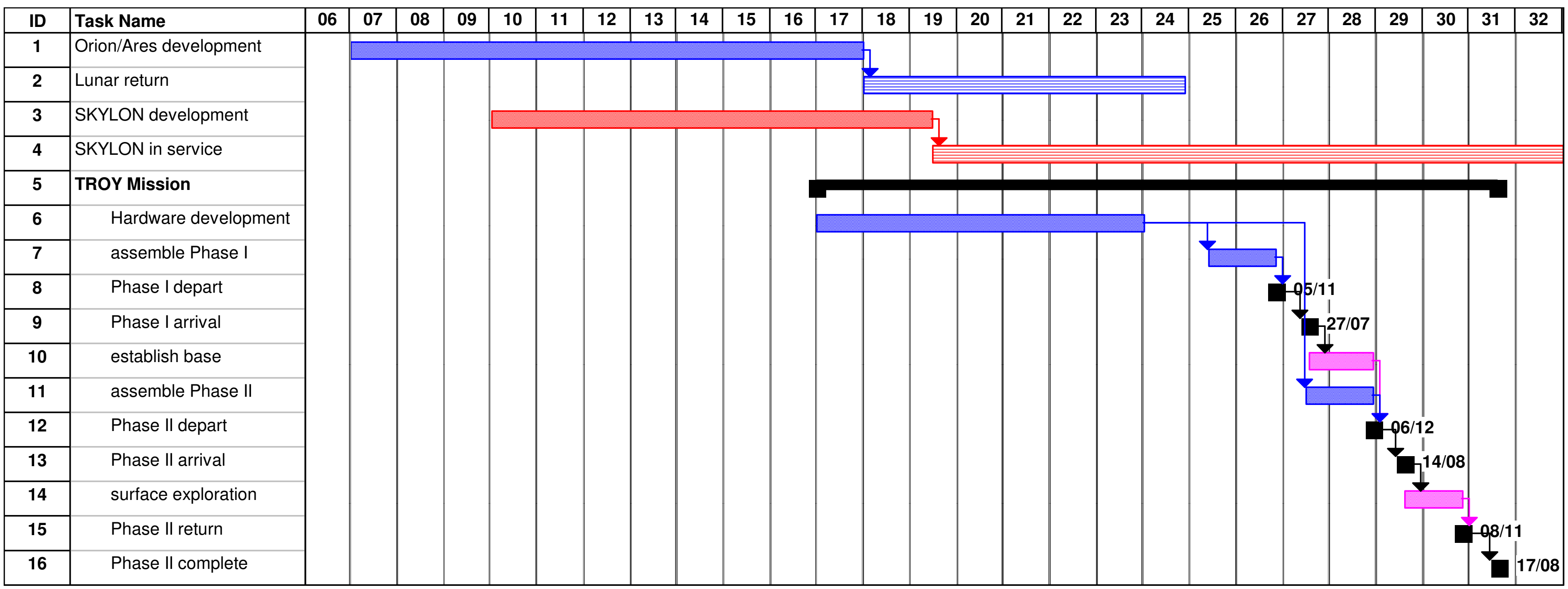
Manned Element Costs

Item (for 3 vehicles)	**Cost: $million (US)**
Operations Base Station	10,000
Mars Bases and Equipment	12,000
Space Payload and Equipment	5,000
Project Administration (nominal)	15,000
Total Manned Element Cost	**42,000**

The launch costs are considered to be reasonably accurate, probably better than 20% error. The Troy vehicle costs are less accurate but still probably about 40%. The manned element costs are dominated by their development costs and are the least accurate, and should be considered not better than 60% error. Thus the total cost of the undertaking will probably lie between $70 Bn and $100 Bn in 2004 prices. It should be noted that the costs are dominated by development and the additional cost of using three vehicles is nominal.

These figures should be compared with the $16Bn spent to date on unmanned probes (Annex A2.1). The compelling arguments for human exploration, rather than the robotic approach, are briefly touched on in Annex A2.2.

Fig.3 Timescale of 'Return to the Moon', SKYLON Development and the TROY Project



4 Summary and Conclusions

This study was performed to examine the role of SKYLON in making the exploration of Mars a practical and, especially, a safe undertaking for the exploration team. To this end a reasonably comprehensive initial study was carried out to identify the size and mass of the components of a credible Mars mission and to compare them in relation to the current performance capabilities of SKYLON.

It is concluded that not only is a vehicle of the SKYLON type able to perform this requirement, but it is also **essential if the exploration of Mars is to be practical, useful and safe**.

This conclusion is a consequence of the assembly time to lift some 2300t of hardware into orbit for each mission phase. It is of course true that a higher risk undertaking could be mounted using a single vehicle for each phase, or even back off to the point of not using a Precursor mission or planetary surface propellant manufacture. The mission would then fall within the scope of the launch characteristics of expendable rockets. However, this would be little more than a ‘Flags & Footprints’ undertaking of poor scientific return and at huge cost.

SKYLON would enable a mission spanning 14 months on the Martian surface by a distributed team of 18 explorers covering 90% of the planets surface. This team would be supported by a transport system of surface and flying vehicles ensuring that they were never isolated in the event of problems.

The mission would have surface and orbital resources enabling extended stop-over to wait relief should major equipment failures occur. In addition the use of a three fleet ‘Columbus’ approach would enable full crew return should a vehicle fail.

The exploration of Mars will be a huge commitment of resources and it would be a bonus if it left a benefit in its wake. The SKYLON vehicle has characteristics which would transform commercial access to space and this could be a very satisfactory legacy of a Mars mission.

Annex A.1 Mission Provisions

A1.1 Biological Requirements

Living Space

In this study it has been assumed that while in space there needs to be a minimum gross volume of $60m^3$ per person. There is some evidence that this may be too small and ESA, for example use $75m^3$ per person in their own Mars exploration study.

The SKYLON payload bay has a volume of $202m^3$ and so each module for the space habitat must be less than this, say $180m^3$. Thus two modules would provide $60m^3$ per person while three would provide $90m^3$ per person.

Nutrition

It has been assumed that a 75kg person needs 3000kcal/day in space and 3500kcal/day on the surface of Mars. Metabolism may be represented by;-

$$CH_2O_{0.5} + 1.25O_2 = CO_2 + H_2O \text{ (liquid)}$$

with 1mol of O_2 releasing 104.7kcal of heat.

Average low fibre food is assumed to have a calorific value of 5000kcal/kg and the mass has been increased 10% for additional fibre. Table A1.1 shows the balances for a 75kg person.

Table A1.1 Life Support Requirements (nominal 75kg person)

Input	**In Space (kg)**	**On Mars (kg)**
Dry food	0.660	0.770
Oxygen	0.917	1.070
Water	2.631	2.631
Output		
Urine	1.522	1.556
Faeces	0.155	0.182
Respiration & perspiration	1.522	1.556
CO_2	1.009	1.177
Heat	3000kcal	3500kcal

For a mission in 2028 with a crew of 6 spending 434 days in space and 451 days on the surface the requirement is 4200kg food, 583kg O_2 and 3000kg water assuming 500kg of water per person recycled. This represents about 19% of the total water throughput and includes utility use.

Space Suits

Each crew member has a suit with an assumed mass of 220kg. It is assumed that each vehicle carries a spare suit, the sum totalling 1540kg per vehicle.

Atmosphere

It will be necessary to remove CO_2 from the atmosphere. Although it is a crude technology which will surely be bettered it has been assumed here that lithium hydroxide (LiOH) will be used for this purpose to fix the CO_2 as Li_2CO_3 with the release of water. 1kg of CO_2 requires 1.091kg LiOH for its removal.

The total CO_2 produced by the 6 crew is 6418kg, requiring 7002kg of LiOH for its removal.

Respiration will produce 2625kg of water and CO_2 removal another 4502kg. Hence a total of 7127kg of stores will be converted to water. This will be used to make up for losses from the recycling of the 3000kg of water provided.

Temperature

The heat generated by the crew and CO_2 removal in space is 1045W which, together with the heat generated by electrical equipment, needs to be removed.

The in-space power requirements are determined by;

Electronics (navigation & computing)
Electrics (actuation and command)
Air Conditioning (CO_2 removal, H_2O condensation & purification)
Cooling System
Lighting
Stabilizing Wheels
Communications
Hygiene (toilet, shower, etc)

These are roughly estimated to be 5kW(e) on the outbound and return journeys and 1kW(e) while the crews are absent on the Martian surface. The heat rejection will be 6045W at 298K and hence requite a radiator area of approximately 15m^2 for emissivity of 0.9.

Gravity

Lack of gravity is linked with serious bone loss which can only be countered with rigorous exercise routines. If centripetal acceleration is used to mimic gravity it must employ a sufficiently large radius to minimise Coriolis accelerations, as the crew move around, to acceptable levels. Unfortunately there is no consensus as to what are acceptable levels, which possibly vary with the individual. If this route is employed it has been assumed here that as large a radius as is practical will be employed, taken to be 90m.

For an angular velocity of 0.2 rad/s the centripetal acceleration is then 3.6m/s^2, similar to that on Mars. Radial movements of 0.5m/s would give Coriolis accelerations of approximately 0.2m/s^2 or 5% of the apparent gravity. This is probably at the limit of what would be acceptable.

There is an engineering problem with perturbations of a rotating system as the crew move around. If the habitat modules are connected with cables there would also be the problem of spinning it up and slowing it down. It is assumed that momentum wheels would be employed to reduce these problems.

It is by no means clear that it will be necessary to provide artificial gravity for the duration of the Earth – Mars transfers.

A1.2 Radiation Protection

These issues are very complex and require a very detailed study. It may be that the radiation issues will dictate the age and sex of the crew members. Here the discussion is limited to overall description of the problem and proposed solutions.

Cosmic Ray Background

The mean particle energy is around 4000MeV/nucleon with an energy range of 10^2 – 10^{11} MeV. The flux is isotropic and constant. This represents a background dose rate of 5 – 12 rem/year, mainly by secondary radiation generated in the surrounding structure, which would be acceptable. However there would be about 6 primary 'hits'/cm^3 of tissue per day by heavy nuclei. Careful shielding with equipment and furnishings would reduce these by an order of magnitude.

Van Allen Belts of Earth

These are very intense belts of electrons and protons trapped in the Earths magnetic field. They will be traversed four times by the crews on the Troy mission and are not considered to represent a serious danger.

Solar Flares

Solar flares, consisting of protons and electrons, are frequent and lethal. They have an actual duration of 2-3 days but are most lethal within 5-10 hours of their occurrence, for a duration of a day. At the Earth dose rates can reach 100rem/hour with a total exposure of 400rem.

Water and hydrogen are ideal shielding materials. It is normally quoted that 300kg/m^2 of water or 35kg/m^2 of hydrogen is required to drop the absorbed dose to 25rem/flare.

The ERS carries 3000kg of water and 9926 kg of fuel cell reactants. It is proposed to distribute these around a 2m diameter by 2m long flare shelter and back fill with the water produced by the fuel cells. The 6 crew would be bunked around the walls of the cylinder during a flare, providing an additional measure of self shielding.

A1.3 Surface Produced Reactants and Propellants

An abundant supply of propellants for frequent flights to-and-from Martian orbit and between surface locations on the planet would greatly increase the scientific return on the huge resources which will have to be expended to bring such a mission about.

A possible solution is to use nuclear generated power in a fixed installation on the surface to manufacture chemical propellants from the Martian atmosphere. These would be liquid oxygen oxidiser and one, or more, of the non-stoichiometric carbon oxides including pure carbon itself, as fuel. These can provide specific impulse approaching liquid oxygen and hydrocarbons and are adequate for moving around the Martian environment. Martian orbit could easily be gained from the planets surface in a single reusable stage. They are also relatively dense propellants giving compact vehicle designs.

Carbon monoxide and carbon suboxide look the most promising practical propellants and manufacturing routes have been examined for both of these. Pure carbon, if it could be suspended in a slurry, could give better performance but has obvious storage and transport difficulties. Carbon monoxide is a cryogen and carbon suboxide has a tendency to polymerise without careful precautions. At the moment carbon monoxide looks the easier route for a first generation mission, while the suboxide may be more suited to later missions.

A preliminary investigation has been made of the characteristics of a reusable engine burning carbon monoxide and liquid oxygen. This is described in section A1.4.1.

ESA initiated a CO/O_2 solid oxide fuel cell (SOFC) development program in 2003 specifically for use on Mars and using reactants manufactured from the atmosphere. Applications of this technology to surface vehicles is discussed in section A1.4.2

The energy required to manufacture propellants is of the order of 6.5 MJ/kg at 100% efficiency, but more likely 30MJ/kg for a practical plant. A plant able to provide 100kW(e) for propellant production could therefore produce some 330 tones of propellants and reactants (210t CO, 120t O_2) over the duration of its operation, from its arrival with Phase I to the departure of the crew in Phase II.

Based on previous studies of nuclear power supplies using the Martian atmosphere for the heat rejection from the conversion cycle, the system mass (without shielding) would be approximately 10 tonnes. The shielding would be provided by the Martian regolith.

Apart from the power supply it will be necessary to land a propellant factory, liquefaction plant and storage facility. The flow rates are small, processing 0.0037kg/s of atmospheric CO_2. Because of the cryogenic nature of the propellants the tanks will need to be insulated and cooled.

The landing of a habitat, power supply and propellant factory will require a mobile robot capable of being controlled to deploy, bury and commission the power plant and connect the units together.

A1.4 Surface Mobility

It is desirable to provide the maximum surface mobility to a large exploration team placed on the Martian surface. It is also a requirement to return the team to orbit to rendezvous with, and transfer the crew and returnables to, the ERS for transfer to Earth.

The team should therefore have trucks and ferries powered by LO_2 – LCO manufactured as described in section A1.3. An outline of the characteristics of these vehicles is given in sections A1.4.1 and 1.4.2 below.

A1.4.1 Ferry

Each expedition will have a single stage vehicle capable of transferring all 6 crew between the Martian surface and Martian orbit. The fully fuelled ferry will be delivered to Martian orbit by the Precursor mission and remain there for the Principal mission to arrive.

The ferry will be an aeroballistic vehicle using the Martian atmosphere to decelerate with terminal propulsive braking. It must have sufficient propellants to immediately regain orbit should there be problem on arrival at the surface base.

The ferry will be required to perform in three different roles. The first as described above is to carry all of the crew to the surface and abort back to orbit if needed. The second, once the base is functional will be to carry payloads to orbit and then effect a return to the base after the delivery. The final flight in this role will be to return the crews to the ERS for the return home. The third role is a possible sub-orbital lob between surface bases should it be required.

For the purpose of this study the ferry specification in Tables A1.2 and A1.3 was derived. This meets the needs of the current investigation but is obviously not a definitive specification. The vehicle was examined using a model of the Martian atmosphere with proper trajectory modelling of the entry and ascent. The available abort ΔV is 4532m/s assuming 488m/s is required for the landing. The ΔV required for an abort back to polar orbit is 4454m/s.

The 95m^2 entry shield was determined from entry modelling and is important because of the need to segment it to fit the SKYLON payload bay. Above this size the segmentation of the structure becomes increasingly more difficult. The ferry will need to be assembled in Earth orbit as it is too big to fit into the payload bay.

Some of the leading dimensions of the ferry are shown in Table A1.2 In addition to the crew, the ferry can carry 2350kg to a Martian equatorial orbit or 1430kg to a polar one, with sufficient propellant to de-orbit and land back at the surface base. Removing 4 crew members and their space suit increases the additional payload by 1180kg.

Table A1.2 Ferry Dimensions

Heat shield diameter	11.0 m
CO tank diameter	4 tanks x 2.546 m (spherical)
O_2 tank diameter	2 tanks x 2.346m (spherical)
Engine bay diameter (3 engines)	4.6m
Cabin diameter	4.6m

Table A1.3 Ferry Mass Breakdown

Parameter	**Mass (kg)**	**Mass (kg)**
Nominal mass at de-orbit	**50,000**	
Useable propellant		41,190
Residual propellant		412
Basic empty mass with payload	**8,398**	
Structure and insulation		3,250
Engines		878
Heat shield (11m dia. SiC/CC)		1,000
Cabin & life support		1,500
Payload	**1778**	
Crew		450
Space suits		1320

A preliminary investigation has been made of the characteristics of a reusable rocket engine burning carbon monoxide and liquid oxygen. It would operate at high combustion pressures, up to 400bar in the preburner and 180 – 190 bar in the combustion chamber. The nozzle would have an area ratio of around 600:1. The configuration of the engine is shown in Figure A1.2

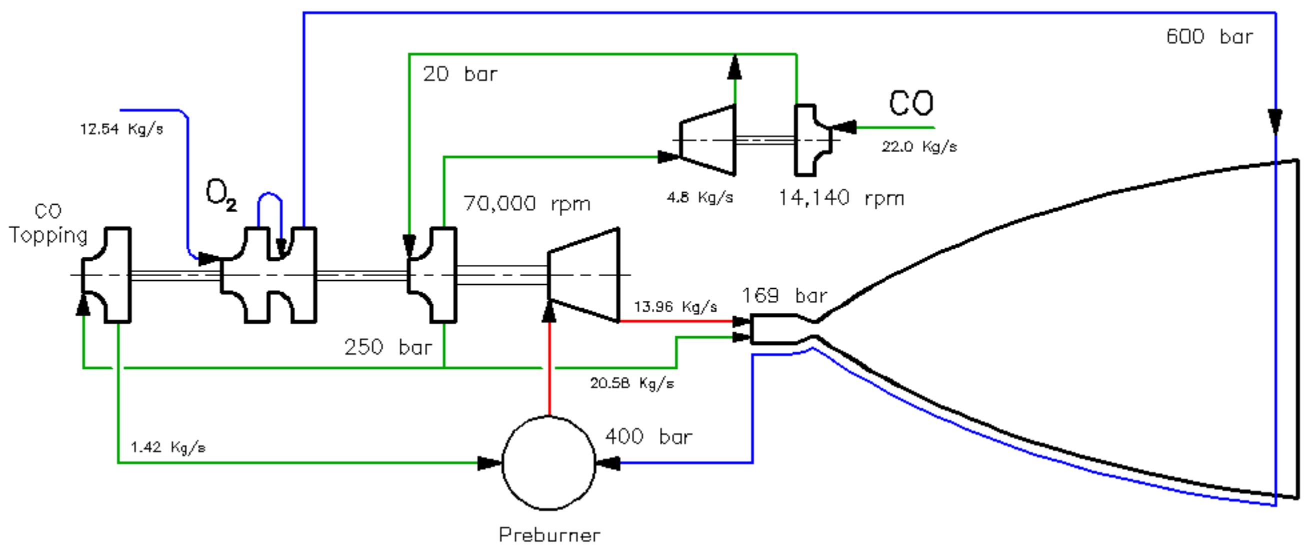


Figure A1.2 Seraph reusable engine

The chamber would be liquid oxygen cooled and the preburner would be oxygen rich. In these respects the engine resembles the P-111 built and tested by MBB in the 1960s. The oxygen rich preburner is selected to avoid soot production which leads to variations in engine performance run-to-run due to deposition on the turbine nozzle vanes.

The vacuum exhaust velocity would be approximately 2920 m/s. The engine studied would have a thrust of 10^5 N and would have a mass of 300kg, standing approximately 2.5m tall and with a 1.4m exit diameter.

The habitat and propellant factory would need similar engines for the final descent braking and these engines would serve as spares for the ferry since they would have no further function.

A1.4.2 Trucks

The range of a land vehicle is given by a variant of Breguet's range equation:-

$$S = \frac{\eta Q_s}{\mu g} \text{Ln}\left(\frac{M_1}{M_2}\right)$$

where S= range
η = the transmission efficiency
Q_s = the motor output power per unit fuel consumption
μ = the effective friction factor of the vehicle
g = gravity acceleration on Mars surface
M_1, M_2 = initial and final vehicle mass

Analysis of tracked military vehicles suggests $\mu = 0.08$, $\eta = 0.69$ are reasonable values. For a solid oxide fuel cell using cryogenic CO and O_2, $Q_s = 3.9 \times 10^6$ J/kg seems achievable (65% of theoretical maximum). At the Martian surface $g = 3.7295$ m/s^2.

For a mass ratio of 2 the range (S) is 6252km and a vehicle could, at its limit range up to 3126km from its base. If the basic truck weighed 5t fully equipped and crewed, the vehicle would need 5t of reactants for this journey. This brings 20% of the Martian surface within range of each base. By using two trucks, transferring 1.91t of reactants from one to the other after 1912km and returning the supply truck home, the range of the fully fuelled truck can be increased up to 4082km from the base, placing 32% of the Martian surface within reach. The three bases could therefore cover over 90% of the Martian surface, in principle.

The design of the surface vehicles, beyond estimating their mass and utility, is outside the scope of this report. It is assumed that they will fit into the SKYLON payload bay.

A1.5 Conditions on Mars

This section highlights some of the more important conditions which the expedition will encounter on the Martian surface. It is not intended to be comprehensive.

The mean Martian day is longer than on Earth by 39min 35sec and the axis of rotation is inclined from the normal to its orbit at 25° 12′ compared with 23° 27′ on Earth, giving a similar impression of seasons, although because of its eccentric orbit they are very unequal in duration. Northern Autumn lasts 142 days while northern Spring lasts 194 days.

Like the Earth at present, Summer occurs in the northern hemisphere while Mars is at aphelion. The Troy mission will arrive at the end of the northern summer season a few weeks before the Autumnal Equinox. The expedition will remain through the northern Winter and Spring and depart just prior to Mars reaching Aphelion.

During this period the southern ice cap will initially disappear and then begin to grow again towards a maximum diameter of 5900km. The northern ice cap will reach its maximum diameter of around 5000km and begin to shrink, although it never falls below about 320km diameter.

Mars has a severe planet wide dust storm season which begins early in the northern Spring, a few weeks after the arrival of the expedition and ends early in the northern summer. The total duration of the storm season is 140 - 150 days. This could be a significant design criteria on the mission occupying about 1/3rd of the duration of the stay on the surface. If necessary the expedition could delay landing until this season has passed.

The atmospheric pressure ranges between 6 – 10 mbar and is a function of the size of the ice caps which cycle CO_2 to and from the atmosphere. The atmosphere is mainly CO_2 (95%), nitrogen (2.7%) and argon (1.6%).

Temperatures are low on Mars, 185 –215K being typical at intermediate latitudes.

Annex A2 Martian Exploration

A2.1 Summary of Missions to Date

An analysis has been undertaken of all missions to the planet Mars flown to 1 January 2007.

There have been 38 missions to Mars between 1960 to the present date. 16 of these missions have succeeded (or partially succeeded) and 22 have failed. This gives an overall failure rate of 58%. The overwhelming majority of the missions have been conducted by the USA and the USSR (18 each), with the USSR having by far the highest failure rate - 83%, in comparison with the US failure rate of 33%.

The missions have been costed, in 2006 economic conditions, using stated programme costs (where available) and estimations (where no data were available). The total cost of missions to Mars is roughly estimated as $16 billion.

The following table summarises the missions flown to date:

	USA	USSR	ESA	JAPAN	TOTAL
TOTAL MISSIONS	18	18	1	1	38
FAILURES	6	15	-	1	22
SUCCESSES	12	3	1	-	16
TOTAL COST (millions US$, 2006 conditions)	$8,647	$6,680	$205	$350	**$15,882**

Notes on Cost Estimates

1. All data are from NASA.
2. Where total project costs have been stated (e.g. Viking, Mariner) these have been escalated to 2006 costs using the NASA Cost Estimator.
3. It should be noted that USSR costs are gross estimates, with the exception of launch vehicle costs, where modern equivalents (e.g. Soyuz, Proton) are priced on the current market (ref. FAA/CST). Spacecraft costs have been estimated on the basis of mass and complexity, and the cost to a contemporary western style economy with respect to high calibre scientists, engineers and technicians, and also laboratory and production facilities.

A2.2 Effectiveness of Human Exploration

Terrestrial environments pose many challenges to human exploration. Examples include vast arid deserts, the extreme cold of the Arctic and Antarctic, predator ridden dense tropical forests, and the high pressure deep oceans.

With the exception of the latter, it has never been suggested that it would be more cost effective to carry out the exploration by robots instead of humans and, indeed, the idea would be given short shrift. Even in the case of the deep oceans the trend is always to extend the depths to which humans can penetrate, having been preceded by tele-robotic devices in many instances. In this case the environment is forbidding to human presence, and yet humans now descend the deepest oceans supported by technology and continue to effectively extend knowledge of these regions.

The same arguments apply to space. However, in space the hindrance has not been the environment as such, but the actual cost of transportation into the region. While the cost effectiveness of a human explorer in terms of information returned for the expenditure incurred is not in serious doubt, the sheer total cost of human based exploration has proved daunting.

An independent assessment of the case for human space exploration has recently been made by the Royal Astronomical Society
(http://www.ras.org.uk/images/stories/ras_pdfs/Final%20Report%20October%202005.pdf).
Key conclusions relating to human versus robotic exploration are:

- “Scientific missions to the Moon and Mars will address questions of profound interest to the human race. These include: the origins and history of the solar system; whether life is unique to Earth; and how life on Earth began. If our close neighbour, Mars, is found to be devoid of life, important lessons may be learned regarding the future of our own planet.
- “While the exploration of the Moon and Mars can and is being addressed by unmanned missions we have concluded that the capabilities of robotic spacecraft will fall well short of those of human explorers for the foreseeable future.
- “However, we believe the essential scientific case at present for Human Space Exploration (HSE) is based on investigations on the Moon and Mars. We have identified 3 key scientific challenges where direct human involvement will be necessary for a timely and successful outcome.
    - Mapping the history of the solar system (including the atmosphere and dynamo of the young Earth) and the evolution of our Sun can be studied via the unique signatures left on and beneath the lunar surface. The possibility that bombardment by comets may have deposited organic molecules throughout the solar system can also be explored, with dramatic implications for the origins of life on Earth. Such investigations will require recovery and analysis of rock cores to depths of up to 100 metres in a variety of different geological settings across the surface of the Moon. We do not believe that a robotics approach alone can deliver this now or in the foreseeable future.

  - Pursuing the question of life on Mars is likely to involve human exploration no matter what the outcome of current and planned robotic missions may be. An early positive signal, indicating that life is readily able to exist on Mars, would further motivate plans for humans to go there. Conversely a continuing negative outcome from robotic investigation would leave open the possibility that life may have retreated below the hostile surface layers. Investigating this will require deep drilling to penetrate the permafrost, with subsequent analysis of rock and ice cores to seek signs of extant or extinct life. Again, we are not persuaded that a robotics approach alone can deliver this now or in the foreseeable future
  - If Mars is found to be a dead planet what lessons can be learned about the long-term viability of our planet to support life? Such a broad-ranging question is likely to require detailed planetary-wide exploration. The expert advice we have received is that such exploration could not be successfully carried out by robotic means alone. Humans are considered far better explorers than robots now and are likely to remain so for decades to come.
- "In summary, we find that profound scientific questions relating to the history of the solar system and the existence of life beyond Earth can best – perhaps only - be achieved by human exploration on the Moon or Mars, supported by appropriate automated systems."

The case has comprehensively been made that humans must follow where robot emissaries have paved the way. All that remains is to discover the will to carry out the task.