40th AIAA/ASME/SAE/ASEE Joint Propulsion Conference and Exhibit
11 - 14 July 2004, Fort Lauderdale, Florida

AIAA 2004-3492

Chemical Propulsion Systems for Low Cost Mars Sample Return

Dr A. M. Baker[*], Dr. I. Coxhill[†]
Surrey Satellite Technology Ltd

P. Henshall[‡]
University of Surrey Space Centre

The European Space Agency (ESA) instigated the Aurora Exploration Programme in 2001. A cornerstone of the initial robotic phase of Aurora is Mars Sample Return, MSR. ESA has designed the first iteration of a 'minimal complexity' (and cost) MSR mission centred around a dual Ariane V launch in 2011. Mission elements include an Orbiter, including an Earth return capsule, which will launch to Mars in mid 2011. A separate descent module / Mars ascent vehicle composite will launch direct to Mars later in 2011. A Descent Module (DM) will slow the composite to a controlled soft landing near the Mars equator, and will provide a science package, sample collection arm and supporting platform functions such as power during a 14 week surface stay. A two stage Mars Ascent Vehicle (MAV) will then launch a 0.5kg sealed Mars sample onto a rendezvous trajectory with the awaiting Orbiter. A combination of 3-axis control on the MAV upper stage and Orbiter propulsion will enable sample capture by the Orbiter, transfer to the upper stage and Earth Return Vehicle for launch back to Earth, arriving in 2014. This paper critically examines the demanding propulsion requirements of the Orbiter, DM and MAV mission elements for the ESA MSR mission. Suggested solutions from the inventory of European and non-European engines have been identified and are outlined. The Orbiter requires a high reliability high Isp engine of around 800N thrust and at least 50 restarts to minimise propulsive losses on the ~4ton platform during deceleration into Mars orbit. The DM system is required to perform close loop regulation of a retro rocket system from measurements of approach velocity. A total thrust of 8000-10000N has been estimated, with an engine capable of throttling to ~10% of maximum thrust with a rapid response time, an Isp requirement has not been determined. The DM thrust depends on the MAV, two MAV's of differing complexity and mass are being studied by major European contractors. The MAV engine will have a thrust requirement of between 2750 and 5500N, depending on the design selected. A comparable Isp to the Orbiter, in excess of 310s, is common to both, as is the survival a 14 week surface stay. All engines are intended to use storable liquid propellants.

I. Aurora and Mars Sample Return

The European Space Agency initiated the preparatory phase of the Aurora Solar System Exploration Programme in late 2001. Aurora's long term goal is to land a human crew on the surface of Mars around 2030. Human exploration will however be preceded by numerous robotic missions to Mars orbit and the Mars surface, accumulating science, geological and meteorological data, demonstrating technologies and building a permanent robotic outpost to support human exploration.

A cornerstone of the initial robotic phase of Aurora is Mars Sample Return, MSR. A mission concept for returning 0.5kg of materials from Mars, launching from Earth in 2011 (with a 2013 contingency date) has been developed by an ESA team using the Concurrent Design Facility, CDF at ESTEC. This is designed to be a 'minimum' mission, i.e.

- Minimum complexity and risk,

[*] Senior Propulsion Engineer, SSTL, University of Surrey, Guildford, GU2 7XH, England. Member AIAA
[†] Propulsion Engineer, SSTL, University of Surrey, Guildford, GU2 7XH, England.
[‡] PhD researcher, Surrey Space Centre, University of Surrey, Guildford, GU2 7XH, England

- Maximum payload mass delivered to Mars,
- Using European launchers available in 2011/2013, and European technology & components where possible.
- Maintaining a manageable mission duration and a realistic surface stay period,
- Avoiding Mars global dust storm season and superior conjunctions in critical mission phases,
- Satisfying planetary protection requirements

The CDF has designed a chemically propelled, split mission relying on twin launches of Ariane V with ESC-A cryogenic upper stages. A Mars Orbiter and Earth Return Vehicle (ERV) on a fully fuelled return upper stage are sent ahead of a 'composite' comprising a descent module (DM) and fully fuelled Mars Ascent Vehicle (MAV).

The two launch dates selected enable both mission elements to arrive at a similar time (June 2013), avoiding the dust storm season and critical operations at superior conjunction. However the DM/MAV is launched 4 months after the Orbiter/ERV, so can be retained on Earth in the event of an early failure of the Orbiter. This later launch is also a function of the launch site turnaround time between each Ariane V ESC-A launch.

Mission analysis has selected a launch in mid-2011 of the Orbiter and sample return vehicle, arriving 2 years later in June 2013. The DM/MAV composite are intended for launch in late 2011, arriving at a similar time to the Orbiter. separating from a small carrier vehicle which is sent into interplanetary space, and directly entering the Mars atmosphere prior to landing. The DM/MAV composite lands on the Mars surface between 5 and 15° S latitude in an area of scientific interest, and remains on the surface for 14 weeks. Return to Earth is scheduled for October 2013, with arrival of the ERV in August 2014.

The complete mission summary is available at http://ftp.estec.esa.nl/pub/Aurora/MSR.

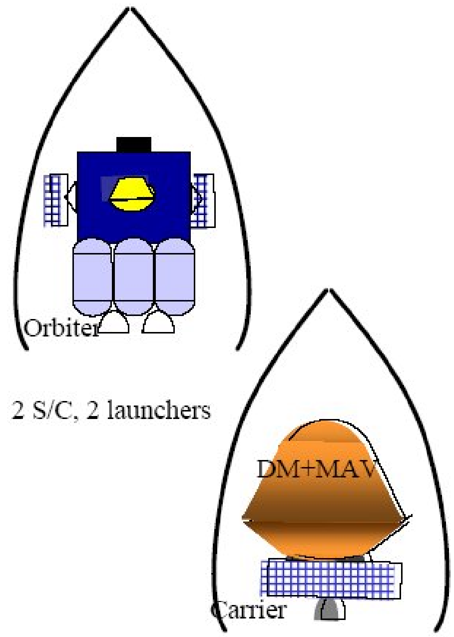

Figure 1. Orbiter in Ar V fairing, DM/MAV in fairing

II. Aurora MSR Mission elements

The Orbiter and ERV are launched in June 2011 on an Ariane V ESC-A, perform a deep space manoeuvre of 120-140m/s ΔV en route to Mars and then capture into a 500km circular Mars orbit, requiring a further 750-800m/s ΔV. The Orbiter then awaits the arrival of the sample carrying MAV from the Mars surface. The Orbiter comprises two stages with a total dry mass of around 1100kg including margin, and requires a propellant mass of 2913kg. A new platform is likely to be required, although maximum use of existing hardware is preferred. The first stage propulsion comprises 6 x EuroStar tanks with propellant management devices and two Astrium 550N bipropellant engines, storing 2200kg of propellant. A second stage is required to control attitude and initiate the return from Mars orbit. The first stage performs trajectory corrections en route to Mars and injects the vehicle into a 500km orbit, then is discarded. The upper stage, containing the Earth Re-entry capsule has a propulsion unit comprising 2 further Astrium 550N engines, and 28 x 10N Astrium engines for attitude control. 4 spherical tanks contain 800kg of propellant. The CDF has estimated the ERV with 5kg of sample containment will have a mass of 80kg. Studies carried out elsewhere[1] suggest that this payload mass can be significantly reduced, to around 45kg, which would enable considerable reduction of the entire Orbiter mass.

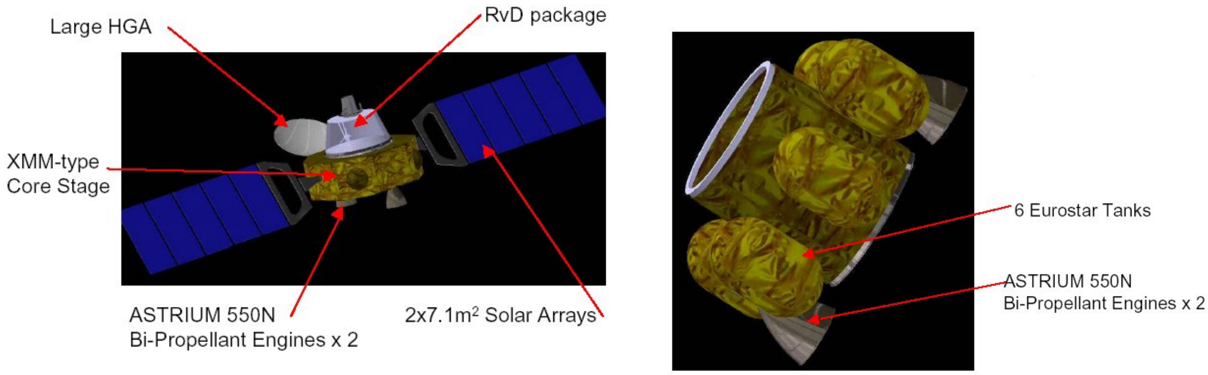

Figure 2. Orbiter showing upper Earth return stage (L) and lower trans-Mars (propulsion) stage (R)

The Descent Module requirements have been derived in part from the Viking mission, since ESA has no experience of carrying out planetary soft landing. The preceding Beagle2 and planned Aurora ExoMars rover missions are designed to use heatshield / parachute / airbag methods due to a relatively low landing mass. The DM platform carrying a fully fuelled MAV will be decelerated from an entry velocity of around 6.1km/s by an ablative heatshield, as per Viking. The entry mass after release of the heatshield is around 1400kg, more than twice that of the Viking lander. A parachute is required to slow the lander composite from the 230m/s velocity after heatshield release to around 60m/s at 3000m altitude. The propulsion system must reduce the velocity and angular rates at touchdown to less than 5m/s horizontal and 5m/s vertical, assuming engine cut off at 3m. Crushable legs will then absorb any residual kinetic energy. 4 Viking type terminal descent engines have been assumed, with an ability to throttle between 0.25 and 2.5kN during the final 60s of descent, and carrying out pitch and yaw stabilisation by differential throttling. Viking's main engine system performed roll control using four secondary 45N engines fed from the main tanks, however roll control for ESA's MSR design is to be performed by engines located on the MAV upper stage.

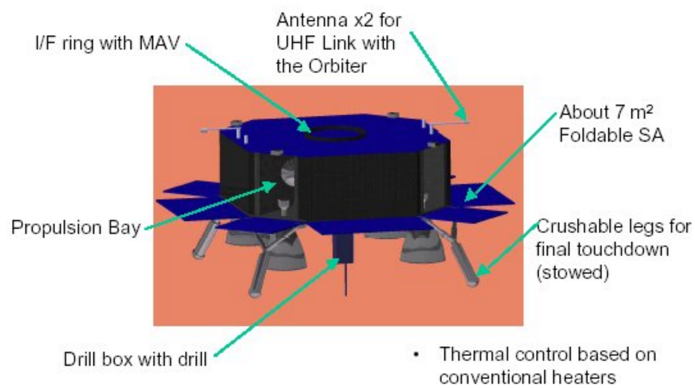

Figure 3. Descent Module

The Mars Ascent Vehicle is intended to be landed fully loaded with propellant onto the Mars surface and to remain attached to the lander (DM) for around 4 months. With the exception of the upper stage thrusters planned to be used to roll stabilise the composite on descent, the MAV is completely self contained. The landing orientation of the MAV will be the launch orientation, no subsequent erection is planned. In October 2013 an automated launch sequence will be initiated. The two stages of the ascent vehicle will be required to generate a minimum ΔV of 3800m/s for launch to a circular 500km altitude. A single stage configuration has been investigated although the CDF suggested that a maximum of 150kg dry mass was required, which did not seem to be feasible for the first iteration. The current MAV wet mass is 900kg, of which 672kg is propellant. The available ΔV from this configuration, assuming certain engine performances (see later) is 4389m/s. Drag losses are assumed to be minimal due to the low atmospheric density of Mars (equivalent to 11-15km on Earth), and the margin is more than ample to compensate for gravity losses due to trajectory shaping. The first stage, using single, steerable engine has a structural mass fraction (structure mass ÷ propellant mass) of 0.32. This relatively high value compared to 0.1 or less for many terrestrial launch vehicles is to be expected due to the small size of the MAV. The upper stage uses a non-steerable engine and separate attitude control thrusters, and has a structure mass fraction of 1.12. This very high value suggests significant further optimisation is possible. The MAV target orbit is 500km with a 15° inclination, trajectory analyses using the ASTOS code have shown that only a weak correlation exists between the MAV final orbit and the ascent vehicle mass. The final orbit must be precise in inclination (0° inclination), within 10km altitude and 0.001° eccentricity to 3σ accuracy. The payload container with an estimated mass of 5kg must be stabilised in 3 axes by the upper stage propulsion unit, which may need to loiter for up to 4 weeks before capture by the Orbiter and transfer to the ERV.

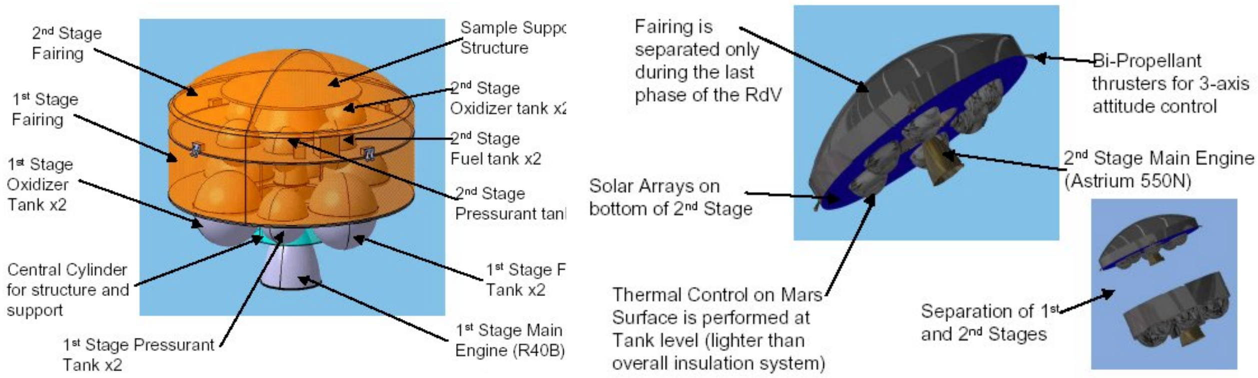

Figure 4. Mars Ascent Vehicle as landed (L) and after upper stage separation (R)

III. Propulsion for MSR

The ESA mission concept requires three separate propulsion systems. The Orbiter propulsion unit must correct trajectory deviations en-route to Mars, brake into Mars orbit, chase the MAV and sample container and inject the ERV onto a hyperbolic return Earth return trajectory. The DM propulsion unit must soft land the descent module and attached science / sample collection package and fully fuelled MAV on the Mars surface at a carefully selected site. The MAV, after a 14 week surface stay must reliably launch the Mars sample and container into a Mars orbit accurately matching that of the Orbiter / ERV, for sample transfer. The ESA Concurrent Design Facility summarised requirements for the 3 propulsion systems in the table below. Note that engine mass requirements are nominal and are subject to confirmation by European industry:

Mission element	Nominal thrust level (kN)	Minimum Isp (s)	Total firing time (s)	Number of restarts	Throttleability range	Maximum dry mass (kg)
Orbiter, each stage	*0.8*	*316*	*1800*	*50*	*N/A*	*20*
Descent module engine	*8-10*	*Not yet known*	*60*	*0*	*TBD but 100-10% preferred*	*20*
MAV	*4-6 lower 0.5-0.8 upper*	*310 lower 315 upper*	*600 each stage*	*2 each stage*	*N/A*	*20*

Table 1 – Summary of mission element propulsion requirements

A. ORBITER

The Orbiter main engine requires a moderate thrust (800N), high Isp (316s) engine to minimise losses associated with braking into Mars orbit and providing the mission ΔV, and to minimise propellant mass. The long burn time and multiple restarts, not required for the other mission elements stem from the need for course corrections en-route to Mars. Although there are no European space engines which generate a thrust of 800N, this is approximately twice that of many industry standard engines, for example, the ARCUK LEROS-1b (using MON and hydrazine propellants, and widely used for Mars missions by both NASA and ESA), the EADS-Astrium S400/N and newer engines being developed for the Ariane Transfer Vehicle programme. A twin engine solution would seem appropriate, although the reliability of using a pair of such engines has yet to be established. A single engine solution would be possible using the 800N Aerojet R42-SR, qualified in 1988, although the current ESA preference is not to use a US engine due to ITAR constraints and the desire for an all-European solution.

The Orbiter upper or return stage must capture the MAV sample container and launch it onto an Earth intercept trajectory. The Mars orbit manoeuvring to capture the sample is assumed to be low ΔV (70m/s). Launch onto an

Earth return trajectory requires a further 1.95km/s. The 2nd stage and ERV have a dry mass of ~590kg, coupled with 800kg of propellant, which is comparable to Mars Express at 1100kg, hence a solution similar to that mission may be appropriate here. The Orbiter upper stage also requires an attitude control thruster system for rendezvous with the sample returning from Mars, and for trajectory corrections between mars and Earth. Currently 28 x 10N Astrium bipropellant engines have been determined by ESA. In comparison with the 16 is a more typical for a large spacecraft, and the eight 10N thruster used for Mars Express this requirement is being reviewed.

B. DESCENT MODULE (DM)

The ESA DM engine system design is based on four continuously throttleable engines, nominally similar to those which powered the Viking lander missions in the 1970s. The Viking lander terminal descent engines (TDEs) were specifically designed for the purpose by the Olin-Rocket Research Company, using ultra-pure hydrazine decomposed in a blowdown monopropellant mode, pressurised by nitrogen gas initially at 36Bar and capable of generating a thrust between 2700 and 350N, varied by a motor driven throttle valve. A unique annular catalyst bed spread the decomposed gases between 18 separate small nozzles, due to concerns about disruption of the landing site regolith from a single nozzle. The use of hydrazine monopropellant instead of a higher performance bipropellant stemmed from the requirement to avoid carbon contamination on the surface around the landing site, since a number of Viking experiments were searching for C isotopes as a signature of life. The rationale behind the Viking lander TDE designs is detailed in Ref. 2 and Ref. 3. Ref. 3 notes that the engines had to have 18 nozzles, the capability of 10:l throttling, a new catalyst, be totally sealed until fired, employ no organic unsealed materials, be 100% germ free, utilise a new propellant, and start at a temperature more than 45°F (25°C) below the propellant's freezing point. These were developed in 30 months for a firing time of 45s, but qualified up to 400s, and, it is pointed out were so optimised for the Viking lander application that they were most probably unusable for almost any other application.

The TDE and the suggested ESA CDF schematic for the propulsion system are shown below:

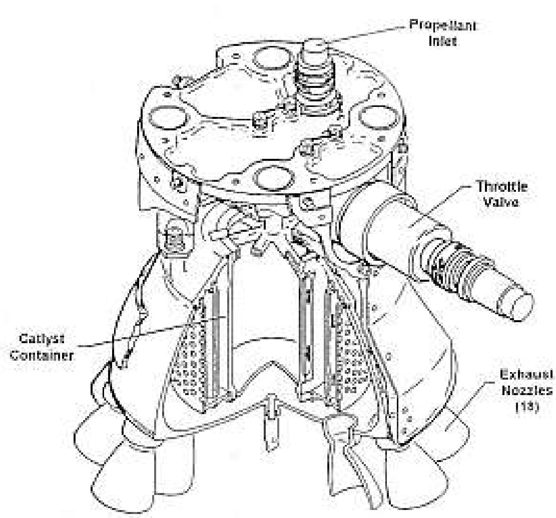

Figure 5. Viking TDE schematic

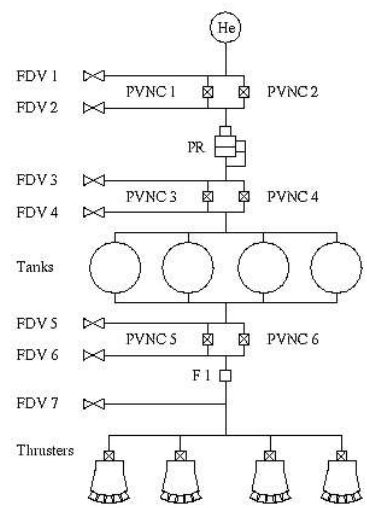

Figure 6. DM propulsion system layout schematic

The DM propulsion system mass breakdown is given in the adjacent Table 2. A total estimated propulsion mass of just under 91kg makes up around 10% of the DM dry mass, and contains 180kg of propellant. This is intended to be stored in 4 spherical metal tanks. An estimate using thin walled pressure vessel theory suggests that a propellant tank mass of at least 4.5kg is required, assuming a proof test safety factor of 1.5x MEOP, and Ti-6Al-4V alloy tank shell material with a 0.25 proof strength 880MPa. The estimate for ACS thrusters of

Name	Quantity	Unit Mass (kg)	Mass Margin (%)	Unit Mass with Margin (kg)
ACS Thrusters	4	8.0	5	8.4
Tanks	4	4.1	10	4.5
Piping, valves & Brackets	1	13.7	20	16.4
Radioaltimeter	1	10.0	20	12.0
Pressurant Tanks	1	4.3	10	4.7
Thrusters Driver	1	5.0	20	6.0
SUBSYSTEM TOTALS		**81.4**	**11.6**	**90.8**

Table 2 – DM propulsion breakdown

8.5kg each including margin is not too dissimilar from the Viking TDE at 7.7kg, although the likely unavailability of such engines meant a cluster of 14 MONARC-445 monopropellant thrusters giving the same thrust level was used for mass estimation, a total mass of 21.6kg.

C. MARS ASCENT VEHICLE (MAV)

The first stage of the MAV requires a high thrust to lift the wet mass of the vehicle from the Mars surface and generate altitude prior to shaping the trajectory towards the local horizontal at orbital insertion. The original MAV design calculated a liftoff mass of 891kg, which at the Mars surface gravity of around 3.7m/s^2 requires a minimum thrust of 3400N for a single engine. ESA suggested the pressure fed Kaiser-Marquardt (now Aerojet) R-40B2 engine, based on the R40A engine developed for the US Space Shuttle Orbital Manoeuvring System. The R-40 engines have been suggested for NASA Mars Sample Return studies. However results obtained from European prime contractors studying the Mars Sample Return mission found that, depending on the vehicle architecture preferred, a total thrust of either 2750N or 5500N would be required. At the lowest thrust level the R-40B2 can generate 2700N thrust using MON-3 and MMH propellants at a 1.6:1 mixture ratio, with an Isp of around 304 and 306s, and an engine mass of 10-11kg with an optional thrust vector control system adding another 10kg. The R40 engine was used as a baseline engine for this study.

The upper stage of the MAV does not require the same high thrust due to the much reduced wet mass of the vehicle by that point (~250kg v. 891kg at launch), although high Isp performance and restartability are requirements. A flight profile for the MAV is not yet available, although burn times of 266s and 680s for lower and upper stages respectively have been estimated, against a requirement of 600s each. The EADS-Astrium 550N engine using a Pt-Rh thrust chamber and under development has been selected, this weighs approximately 5kg and is expected to generate an Isp of 325s. In addition a set of 8 attitude control thrusters are required, which are assumed to be 10-22N mono- or bi-propellant thrusters. Such thruster are available off-the-shelf from several European suppliers at a mass of between 0.22 and 0.35kg. These are also employed to control roll during the descent of the combined DM/MAV composite.

A total propulsion dry mass of 87.5kg has been estimated by ESA for the lower stage. Of this 30.8kg is propellant tankage, 17.2kg is pressurant tankage and 21.8kg is the main engine. Valves, regulator and fittings make up the remainder. This stage contains 487kg propellant.

A similar analysis of the propellant tankage to the DM has been carried out. Ti-6Al-4V is the optimum material. Breaking down the assumed MON and MMH propellants by a 1.6:1 mixture ratio (which, accounting for the differing densities allows for equal volumes of propellants and hence identical fuel and oxidiser tanks) reveals that four 110litre tanks, including 5% ullage volume for expansion effects are required. ESA has assumed that a mass budget of 7.0kg is required for each of these tanks. Assuming these tanks are required to operate at 1.5x MEOP of the propulsion feed pressure, and that MEOP is around 30Bar (pre-injector pressure of the R-40B2 at maximum thrust plus a factor estimated at 3Bar for pipework ΔP), then application of the thin wall pressure vessel equation using the tensile strength of Ti-6Al-4V alloy at 980MPa shows that a spherical shell of outside diameter 597mm and thickness 1.37mm will give the required burst pressure, for a mass excluding fixtures and supports of 6.77kg. Including margin this closely approximates the ESA estimate of 7.4kg. Options for further reducing this mass include optimising the shell thickness for spherical tanks (where the maximum wall stress is only 50% of that in a cylindrical shell, assumed for the above calculations), and using advanced materials such as carbon fibre composites. Note that an off-the-shelf tank which almost meets the MAV lower stage requirements is the EADS-Astrium OST 31/0 designed for Globalstar. Removal of the diaphragm in this spherical titanium shell would enable a volume of 104litres, with a maximum diameter of 600mm, and a burst pressure of 49.2Bar, at a total mass of 6.4kg. Carbon fibre tanks are unlikely to be available for the relatively low pressures required by pressure fed space engines.

Analysis of the effect of launch vibrations and the manufacturability of thin wall large area shells of titanium alloys will be required in any actual design.

A similar analysis for the stated requirement of 3kg of He gas to pressurise the main engine propellants has shown that each tank (two are required) has a mass of 19.0kg, regardless of whether the He is pressurised to 200 or 300Bar. 19kg is well in excess of the ESA estimate of 8.6kg including margin. This suggests firstly that the CDF methodology for calculating pressurisation component mass needs to be examined further (for example verifying the mass of He gas required). Composite pressurant tanks may enable the CDF mass estimate to be met, alternatively a pump fed system, assuming that a lightweight pump can be procured, may offer significant mass savings.

Off-the-shelf components including a dual series redundant gas pressure regulator, gas and liquid pyrovalves to initiate flow, non-return valves, filters, fill/drain valves, and pressure transducers have been identified. A first iteration mass breakdown estimate for the lower stage propulsion system, assuming CDF pressurant tank mass estimates is given below, adjacent to a schematic of the upper stage propulsion layout, the lower stage is identical, excepting the attitude control thrusters:

Component	Mass (kg)	Notes
4 propellant tanks for 487kg MON/MMH	4 x 6.4	EADS-Astrium OST31/0 104litre
2 pressurant tanks for 3kg He	2 x 8.6 inc. 10% margin	CDF estimate, not verified
Main engine	23.4 inc. margin	Inc. TVC gimbal system
Regulator, valves, pressure transducers, filters	4.43	Polyflex, Stanford Mu, Kulite, OEA Aerospace
Piping, fixings	7.8	
TOTAL	**78.4kg**	

Table 3. MAV lower stage propulsion masses

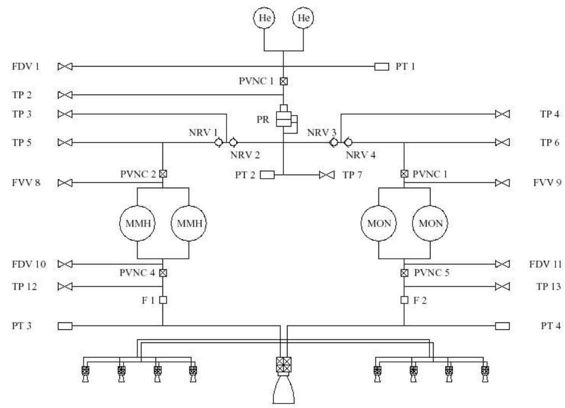

Figure 7. MAV propulsion layout schematic

The propulsion dry mass figure of 78.4kg for the lower stage represents a saving of around 10% on the original ESA estimate of 87.5kg. Further savings could be made by substituting the pressure regulator with a bang-bang regulated system using solenoid valves, saving a further 2.5% and some considerable costs. However uncertainty in the engine and TVC mass needs to be reduced before mass savings in the detailed component breakdown can be made accurately.

Applying a similar method of analysis to the upper stage, which has a propellant mass of 117kg and an estimated propulsion dry mass of 44kg has found that there is potential for a significant mass reduction. Propellant tank capacities of around 25 litres are required, conveniently Ardé in the US manufactures the spherical model D3998 in CRES301 stainless steel, with a mass of only 1.82kg, offering significant mass savings on the ESA CDF estimate of 3.3kg per tank (4 are required). Estimates of He pressurant tank mass again exceed the CDF estimate, around 9.4kg calculated compared to 4.3kg. Composite He pressurant tank technology clearly needs to be examined here. A lower mass main engine, the 4.1kg Leros 1B from ARCUK is also suggested. Note that this has an Isp of 318s against the projected 325s for the Astrium 550N engine, although the thrust of 707N peak is higher. Eight MONARC-22 monopropellant thrusters are required for the upper stage attitude control system. A mass breakdown estimate for the upper stage propulsion system is provided below:

Component	Mass (kg)	Notes
4 propellant tanks for 117kg MON/MMH	4 x 1.82, + 5% margin	Arde D3998 25.6 litre
Pressurant tank for 1.5kg He	4.73 inc. 10% margin	CDF estimate, not verified
Main engine	4.3	ARC Leros-1B
8 AOCS thrusters	8×0.25	MONARC-22
Regulator, valves, pressure transducers, filters	3.35	Polyflex, Stanford Mu, Kulite, OEA Aerospace
Piping, fixings	4.4	
TOTAL	**26.4 kg**	

Table 4. MAV upper stage propulsion masses

The ESA mass estimate for the upper stage propulsion system was 44.2kg, therefore this estimate implies a mass saving of some 40%, assuming the mass estimate for the pressurant tanks can be verified. Examination of alternative options, such as pump feeding of propellant, or bang-bang pressurisation is planned later in this study.

An additional issue which has not affected previous landers to the Mars surface is thermal control of propellants. The low heat capacity of the thin atmosphere results in a wide temperature swing, almost 100K between sunlit and night periods over a Martian sol (day). This diurnal temperature variation is plotted below:

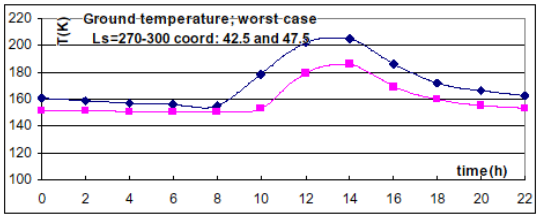

Figure 8. Martian surface temperature variation

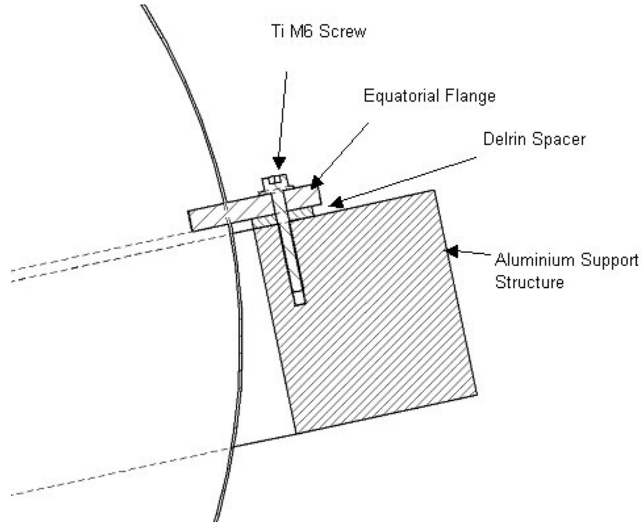

Figure 9. MAV Tank mounting arrangement

Ground temperatures between 150 and 160K at night, or 60-70K below the freezing point of MMH and 90-100K lower than that of MON-3 are expected. The low solar flux on the Mars surface, stemming from the increased distance from the sun compared to Earth, the eccentric Mars orbit and dusty atmosphere combine to provide little solar insolation to warm tanks during the Martian day. Either freezing of the propellant, requiring extensive heat input to thaw before launch and potentially damaging sensitive component or continuously heating the tanks to maintain propellants in liquid form are options.

A simple finite difference computer model of the MAV propellant tankage has been built. This assumes cryogenic foam insulation sprayed on the tanks to reduce convective heat losses to negligible levels, no radiative losses due to the low ambient temperatures, and electrical tank heaters to offset conductive heat losses through an equatorial tank flange which is insulated from the tank support structure by a Delrin (a low conductivity space qualified plastic) standoff. A schematic of the tank mounting arrangement is shown below:

The thermal model has revealed the following issues associated with storage of a liquid propellant launch vehicle on the surface of Mars for long periods:

1. Use of MON-25 (N_2O_4 / 25%NO) / MMH propellants with freezing points of –56 and –52.6°C respectively are preferred over the more commonly used MON-3 and N2H4. This may require engine requalification, although Russian engines are reported to use MON-25 more widely.

2. Maintaining the propellants above their freezing points throughout the surface stay, as opposed to allowing them to freeze within the first few days after landing and then thawing them by application of electrical power is preferred. The former requires around 30% less power than the latter, which also takes some 6-10weeks.

3. The power required to maintain the 687kg of propellants in both stages liquid throughout the mission is a minimum of 218W, which exceeds considerably the 120W (peak) available form the lander arrays at the start of the mission. Furthermore, science activities will require power, the peak power available is expected to diminish throughout the mission due to dust deposition on the arrays, and the DM power system is not currently designed to generate constant power throughout the Martian day and night.

These issues point to a significant power shortfall for propellant thermal control. Options to alleviate this will be explored later in the study and include use of lower freezing point propellants such as MAF (Mixed Amine Fuels) / IRFNA (Nitric Acid) or gelled liquids, reducing the quantity of propellant, modifying the tank mounting arrangement and redesigning the DM power system to generate more power.

IV. ESA propulsion study team and objectives

ESA in November 2003 initiated a 12 month study contract to examine the crucial propulsion requirements for MSR. The industry team lead by Snecma Moteurs Ltd, Villaroche France includes Surrey Satellite Technology Ltd, Guildford, United Kingdom, Alta S.p.A, Pisa, Italy and Alcatel, Cannes, France. Snecma is coordinating the study and, initially considering the Orbiter propulsion requirements. The study objective is to critically examine MSR propulsion requirements, survey the range of European and non-European engines which might meet said requirements, and develop a set of criteria for confidently ranking and selecting engines.

In the initial stages of the study, SSTL will coordinate the evaluation of ESA's CDF study and the relationship between the systems design and the propulsion requirements. SSTL will also aim to determine a suitable propulsion configuration for the descent module, applying its wide heritage in hybrid, bipropellant and monopropellant thrusters, as well as its experience in low cost approaches to engine development. Two particular engines which it will consider are a low cost bipropellant engine using concentrated hydrogen peroxide / kerosene propellants and a novel decomposed oxidiser cooling mechanism, and a vortex flow geometry hybrid engines designed for packaging in limited volumes and to address some of the limitations of conventional axial hybrids. These engines may be of benefit if new developments are required, for example in Mars ascent vehicles.

Alta will apply its hydrodynamic experience gained in testing liquid turbopumps to the MAV, assessing whether a pump fed engine offers any benefits in reducing system mass. Alta will also examine alternative configurations for the MAV architecture, including pressurisation methods, staging, and mass budget breakdowns. Alta is also involved in MAV assessments for one of the ESA Phase A studies on MSR architecture.

Alcatel will offer an overview of its past experience of Mars sample return mission design, through previous studies carried out for CNES, as well as performing a trade-off between high thrust engines which may induce sloshing effects in tanks and result in poor orbital rendezvous manoeuvre precision, against lower thrust engines which enhance the precision of trajectories but lead to larger gravity losses.

V. Russian engine survey

The consultancy firm Commercial Space Technologies in London have completed a comprehensive survey of Russian and Former Soviet Union engine manufacturers. Data on engines built specifically for space purposes by the Design Bureau for Chemical Machinebuilding (KBKhM Khimmash), the Sojuz Touraevo Machinebuilding Design Bureau (Sojuz TKMB), the Chemical Automatics Design Bureau (CADB) and the Ukrainian Yuzhnoye Design Bureau have been obtained. Additionally engine systems for ballistic missiles have been included, developed by the Moscow Institute of Thermal Technology (MIT) and the Makeyev State Rocket Centre (SRC), which use thrusters developed and manufactured by the R&D Institute of Machinebuilding (NIIMash).

A number of possible engine options have been selected for the MAV and the DM, although the recent change in requirement for the Orbiter engine from a 4kN to an 0.8kN thrust will require a re-examination of the options. The Ukrainian RD-860 engine is the preferred option for the MAV. This is a modern, pump fed design using novel pneumatic pumps. The engine mass is 22kg, slightly higher than the original requirement for 20kg, but is offset by the need for a low (6Bar) tank pressure. A thrust range between 2 and 6kN can be preset, meeting requirements. Isp values of 321s have been obtained, with a total firing time of 1500s. The thrust chamber is an old design, being used in the RD-866 engine forming the post boost upper stage of the SS-18 ICBM, however all other components are new. The engine has not yet completed its ground qualification and has no flight test experience, however is well placed to be qualified, reaching the required TRL in the timeframe of the MSR mission. Export issues related to the use of former or current Russian military technology surrounding many other engines are not an issue with the RD-860. A similar engine, the RD-869 which is being qualified as the upper stage engine of the new European Vega small launch vehicle is also a strong contender for the MAV but in pressure fed form will only meet the lower thrust requirement of the 'capture only' MAV. The RD-860 is shown in the adjacent image:

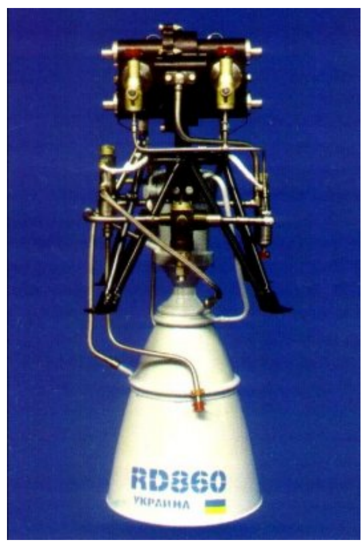

Figure 10. Yuzhnoye DB RD-860 pneumatic pump fed engine

Additional engines being considered are the KRD-79 and KRD-442 manufactured by KBKhM and used on the Zvesda and Zarya modules respectively of the space station. The RD-0237, manufactured by CADB and currently used in the post boost stage of the SS-19 ICBM, and the 17D61 manufactured by KBKhM used in the Soyuz Ikar upper stage are also under consideration.

For the Descent Module selection of suitable Russian engines has proved more challenging. Although highly throttleable engines have been developed for Lunar soft landing missions such as Luna 16-24 and the manned LK L3 missions, these tend to suffer from obsolescence (1960 vintage) or a high dry mass, being intended for high thrust and deceleration of considerably higher mass missions than the robotic DM/MAV composite envisaged by ESA. Alternative options are being evaluated, based around either clusters of small 200-500N engines, the NIIMash RDMT-200A and RDMT-500A/1 thrusters, operated in on/off pulse modulated mode (similar to Mars Polar lander), or a larger main engine which can be throttled to ~50%, such as the RD-0237, combined with smaller thrusters such as the RDMT-400M. This evaluation is ongoing.

VI. MAV systems analysis

ALTA have evaluated the feasibility of some alternative configurations for the MAV architecture. Different engine choices, with various thrust levels, have been taken in account. For each configuration, the impact of adopting different tank materials and using pump fed systems instead of pressure fed systems was analysed.

Alta proposed two alternative small scale pressure fed alternative system, a compact gas generator cycle powered reciprocating quad piston pump built specifically for hydrazine monopropellant propelled small launch vehicles but also tested with hydrogen peroxide[5, 6] and a plenum pressurisation pump under development by Flometric in the USA (http://www.flometrics.com). The aims of the exercise were to reduce the weight utilising low thrust engines and to investigate the impact of the different feeding system and tanks structures on the MAV configuration. Vehicle mass budget estimates have been derived for some reference configurations. For each configuration the optimum initial mass of the vehicle was calculated in several options: three different tank materials (Al alloy, Ti alloy and composite tanks) with a given feeding system (pressure fed) and three different feeding systems (pressure fed, reciprocating quad piston pump and plenum pressurization pump) with a fixed tank material (Al alloy).

The mass of the reciprocating quad piston pump has been estimated, by scaling the figures used for the Astrid vehicle[6], as 3 kg (plus 3 kg for additional fittings and valves and for the gas generator system, in order to avoid the use of pressurant). The mass of the plenum pressurisation pump has been estimated using the open data given by Flometric Inc on their internet site.

The selected configurations are:

- <u>S1</u>: single stage, 2 x DASA S-3K engines (total thrust = 7000 N)
- <u>S2</u>: single stage, 2 x R-40B engines (total thrust = 10936 N)
- <u>D1</u>: double stage. First stage: 1 x DASA S3-K engine (total thrust = 3500 N). Second stage: 1 x DASA S400 engine (total thrust = 400 N)
- <u>D2</u>: double stage. First stage: 4 x Kaiser Marquardt R4-D engines (total thrust = 1960 N). Second stage: 1 x Kaiser Marquardt R4-D engine (total thrust = 490 N)

The R-40B, as described earlier, meets the MAV and Orbiter requirements as closely as any off-the-shelf engine, but poses technology transfer issues to Europe. An alternative is the DASA engine concept, the S3k, which has been built, tested but not qualified and is meant to represent the upper limit of performance which might be obtainable using MMH and NTO based propellants. The S3K concept generates 3.5kN thrust for a mass of 14.5kg, for MON3/MMH mixture ratios of 1.6-2.1. The Isp performance is in excess of 325s, and in the limit at very high (pump fed) chamber pressures might approach 350s. Operation time and restartability have not been established. The DASA S400 and Kaiser Marquardt R4-D engines are typical 400N class bi-propellants using MON and Hydrazine/MMH propellants, used for orbit transfer and deep space manoeuvring in a wide range of missions. The ARC Leros 1B and Leros 1C are similar but can achieve slightly higher thrust levels.

The analytical model has been evaluated with various MAV configurations, down to lower stage thrust levels of 1000 N, and single stage configurations. Various bipropellant and monopropellant thrusters of this class have been tried, but none seem to match ESA's specifications for the payload mass and orbit. This result suggests that the only way to obtain a suitable MAV configuration with a 1000N class thrust level is to follow the indications given in [Ref. 5], rescaling an existing thruster (if possible, a monopropellant one) in order to obtain a significant reduction of nozzle area and weight, and taking in consideration the omission of some features included on conventional satellite thrusters but not necessary to the MAV, such as heaters and thermal shielding. Rearranging the global architecture of the vehicle, in order to obtain smaller proportionality factors for some subsystems (structure, mechanisms, thermal control, power) has also been considered.

Some of the results obtained for the selected configurations were:

③ Pressure fed single stage to orbit configurations were not found favourable from a wet mass standpoint, results showing around 780kg gross mass assuming pressure fed engines and Ti alloy propellant tanks. However pump feed combined with low mass tanks gave a significant improvement, reducing gross mass by up to 260kg.

③ A two stage configuration with a significant reduction in the vehicle wet mass was identified, even without varying the propellant feed method and the tank material. Values of 300-500kg, comparable with NASA instigated industry studies[4] have been derived.

③ For two stage configurations, the benefits from pump propellant feed and alternative (composite, Al alloy) tank materials does not reduce the gross lift-off mass of the vehicle in a significant fashion.

③ Vehicles able to launch on lower thrust propulsion (about 1000N) monopropellant engines, as indicated in Ref. 7 do not appear feasible in either single stage and double stage configurations.

Principal tasks undertaken by ALTA in future trade-offs will be to evaluate in greater detail the single stage MAV, combined with advanced pump fed engines such as the RD-860 and ultra-lightweight propellant tanks.

VII. Descent module analysis

A comparison has been carried out of the Viking and more recent Mars Polar Lander propulsion systems, which used two different approaches[2]. Both used monopropellant hydrazine thrusters due to their ability to modulate thrust levels with a short cycle time, without the instability that might be experiences due to injector pressure drop losses in a bipropellant. Viking used prepressurised propellants tanks which were allowed to blow down during firing, with the engines varying the thrust level using a motor driven throttle valve. The Viking TDEs were developed at great cost specifically for that mission, an effort which is unlikely to be repeated.

Mars Polar Lander used off-the-shelf hydrazine monopropellant thrusters (6 pairs of 290N Primex MR-107 monopropellants, arranged in three groups of four thrusters each). The hydrazine was pressurised from a regulated high pressure He tank, and the thrusters were on-off pulse width modulated using a 10Hz control logic. Six of the twelve thrusters were canted off the Z-axis to provide roll control, where Viking used a dedicated set of roll control thrusters. This design methodology is an alternative to the use of highly throttleable engines, which do not appear to be available in Europe, Russia or the US.

The Russian engine survey has also highlighted the fact that Russian monopropellant thruster technology is inferior in level of development and thrust to US efforts (and little work on highly throttleable engines has been carried out recently). Although Russian Lunar and Mars missions used throttleable engines for descent, these were generally designed for much larger (multi-ton) platforms and are now largely obsolete. Options under consideration include use of an array of on/off modulated bipropellant thrusters in the same fashion as MPL, although European monopropellant hydrazine thrusters might prove to have a better response time. Alternatively, a central main engine which can be throttled to ~50% of maximum thrust by varying feed pressure, coupled with a pair of smaller engines which can also be lightly throttled or off-modulated might an approximation of a smooth variable thrust between 2500 and 250N. However the fidelity of this approach and the response time all require further investigation.

VIII. Low cost options

In the event that an off-the-shelf engine cannot be obtained, bespoke developments may be required under a limited budget and to be qualified for the 2006. SSTL has a history of rapid product development which may be applicable to this study. Hybrid engines represent a low cost, low complexity option which SSTL and the Surrey Space Centre have examined in the last several years for small space missions. The vortex flow hybrid engine has been developed at the Surrey Space Centre as a highly packageable means of orbit transfer for small satellites. Its reduced aspect ratio compared to a traditional axial hybrid engine, relative simplicity and ruggedness (using a polymer fuel) compared to a bi-propellant, demonstrated performance (c* >1600m/s, combustion efficiency near 100%) and significant development heritage point towards it being a low cost solution which merits consideration for the demanding ascent vehicle needs of ESA's Mars Sample Return mission. Low cost bipropellant engines using hydrogen peroxide oxidiser developed at the Surrey Space Centre, which offer potential for deep throttling by varying the flow rate of hydrogen peroxide are another low cost alternative which are due to be examined.

IX. Conclusions

ESA has initiated a study contract lead by Snecma Moteurs (France) and partnered by SSTL (UK), Alta (Italy) and Alcatel Space (France) to critically examine Aurora Mars Sample Return propulsion requirements, survey the range of engines which might meet said requirements, and develop a set of criteria for accurately ranking and selecting engines. Following feedback from ESA and a review of ongoing MSR Phase A studies by two major industry consortium, an updated set of requirements for Orbiter, Descent Module and Mars Ascent Vehicle mission elements are being considered in the context of a detailed survey of Russian, low cost and pump fed propulsion systems. Following a mid-term review in July 2004, a final selection of engines for the first Mars Sample Return mission in 2011 will be made, development/qualification activity steps determined and a preliminary set of concept designs produced. The overall goal is to add a new, intermediate thrust (kN range) rocket engine to the European inventory to support the Aurora programme.

References

[1] SSTL pre-Phase A study, Aurora Earth Re-Entry Vehicle Demonstrator, EVD. Engineering data package ENG_01. ESA Statement of Work AO/1-4398/03/NL/AG. March 31st, 2004.

[2] Schmidt E W, Bewster G T, Cain G A; Mars Lander Retro Propulsion; paper IAF-99-S.2.02; presented at the 50th I.A.F. Congress, Amsterdam, the Netherlands, October 1999; pp.15.

[3] Morrisey D C; Historical perspective: Viking Mars lander propulsion; AIAA paper 89-2391; presented at the 25th AIAA Joint Propulsion Conference; Monterey, USA; July 1989; pp. 19.

[4] Stephenson D; Mars ascent vehicle – Concept development; AIAA 2002-4318; presented at the 38th AIAA Joint Propulsion Conference; Indianapolis, USA; July 2002; pp. 9.

[5] Whitehead J C; Hydrogen peroxide gas generator with a reciprocating pump; AIAA 2002-3702; presented at the 38th AIAA Joint Propulsion Conference & Exhibit; July 2002, Indianapolis, USA; pp. 9.

[6] Whitehead J C, Pittenger L C; Collela J C; Astrid Rocket Flight Test; *Lawrence Livermore National Laboratory E&TR publication*; 1994; pp. 7-17; available from http://www.llnl.gov/etr/pdfs/07_94.2.pdf; page accessed 5/7/2002.

[7] Whitehead J, Brewster G T; Mars to orbit with pumped hydrazine; AIAA 1999-2150; presented at the 35th AIAA Joint Propulsion Conference & Exhibit; June 1999, Los Angeles, USA; pp. 11.