AUTO-ROTATION AND ITS APPLICATION TO DESCENT AND LANDING ON MARS

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INTRODUCTION

Within the next years several missions aim on landing scientific payloads on Mars. Most of the upcoming missions are rover missions, but also other scientific payloads are to be delivered to the surface. In the long term some missions aim at paving the way towards the first man-rated mission.

In this context one critical technology to master is the entry, descent and landing (EDL). While previous missions targeted to reach easy-to-access landing sites with minor precision requirements, the more ambitious scientific missions will have to cope with new issues. Main technology drivers in this context are the requirements to be able to land at higher altitudes, to perform precision landing and finally to perform hazard avoidance manoeuvres. The typical approach for such an EDL scenario is to use supersonic parachutes in combination with powered soft-landing. Clearly, the performance of a powered lander is restricted by its amount of propellant, while all of its initial kinetic energy is dissipated during entry and parachute descent. This disadvantage might be avoided by the implementation of an auto-rotative descent and landing system.

Auto-rotation is a state of motion in which the air stream around a free-falling vehicle propels a rotor such that the rotor produces thrust. Helicopters use this principal to land safely in the event of turbine failure. With an auto-rotating vehicle it is possible to perform precision landing while maintaining complete controllability. A Martian EDL system based on auto-rotation can, therefore, decelerate after entry like a parachute equipped system and glide to a dedicated landing site reaching down-range capability as high as 25 kilometres. This paper describes such a system concept, its applicability and the need for technology demonstration.

The work described here is based on the following requirements:
– the probe to be landed shall follow a ballistic entry trajectory,
– the probe shall have a Viking-like aeroshell,
– the probe shall be landed at altitudes below 2000 m,
– the probe shall be landed at a descent velocity of less than 20 m/s,
– the landing vehicle shall be capable of manoeuvring close to the surface prior to touchdown, and
– the landed mass shall be between 20 and 200 kg.

MARS ENTRY, DESCENT AND LANDING CHALLENGES AND SOLUTIONS

For scientific missions on Mars it is necessary to decelerate a probe from interplanetary arrival speeds of about 5 km/s to 7 km/s to nearly zero speed at touchdown. Typically most of the kinetic energy is dissipated during entry and to date most of the remaining kinetic and potential energy is dissipated during a parachute descent. A typical entry, descent and landing mission on Mars consists of the following phases:
- Entry
- Parachute deployment (at about Mach 1.5 - 2.0)
- Descent
- Retro-rocket braking (at about 100 m altitude and 60 m/s descent velocity)
- Airbag landing

Alternatively, powered landing missions (e.g. Viking) were performed in which the landing was achieved completely by thrusters and landing legs instead of retro-rockets and airbags. Reference [1] provides an overview of past and planned Mars exploration missions.

As the study is intended mainly to analyse the feasibility of an auto-rotation system for descent and landing on Mars, the focus has not been set on the entry phase. Nevertheless, the entry strongly influences the rotor deployment conditions, as it does also with the parachute deployment conditions in a "traditional" EDL scenario. It is also known that the ballistic coefficient of the entry vehicle significantly influences the rotor deployment conditions. The ballistic coefficient is defined as:

$$\beta = \frac{m}{C_D \cdot A_{ref}} \qquad (1)$$

where m = mass of entry vehicle, C_D = drag coefficient, and A_{ref} = aerodynamic reference area

For a Viking-like entry probe the drag coefficient around zero degree angle-of-attack (hypersonic ballistic entry) is $C_D \approx 1.6$. The reference area in this case is a circular area with a diameter d. For different entry vehicles the mass and the diameter are different thus leading to different ballistic coefficients.

As can be seen from Fig. 1, the entry becomes steeper with increasing ballistic coefficient and the deceleration starts at lower altitudes. The result is a higher velocity at any given altitude (e.g. at 10 km the velocity for β = 50 kg/m² is 340 m/s and for β = 150 kg/m² it is 900 m/s). The blue zone in Fig. 1 marks the zone of typical parachute deployment (dynamic pressures between 250 Pa to 1200 Pa and Mach numbers between 1.1 and 2.1). It can be seen that the entry flight paths with ballistic coefficients above 140 kg/m² do not pass through this region and hence cannot be used with ballistic entry missions. This limit will most probably decrease further when atmosphere variations are taken into account. Also, a ballistic coefficient of 30 kg/m² defines a lower limit.

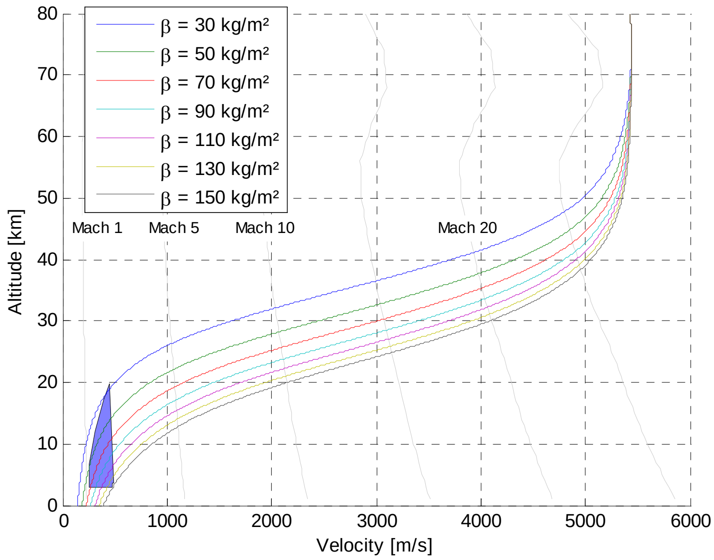

Fig. 1. Flight path velocity - altitude diagram for entries with various ballistic coefficients

Probes are either controlled during entry in order to achieve lift at a specific angle-of-attack or enter on a ballistic trajectory. However, just before entry uncontrolled ballistic capsules are rotated with a small spin rate in order to provide spin-stabilization (Mars Pathfinder's roll-rate is 12 deg/s [2]). The design of the deceleration system has to take into account spinning motion which might tangle parachute lines or induce other undesirable effects.

Besides mission specific constraints arising from the entry, the selection of a target landing site is also of importance for EDL missions. Figure 2 shows the global topography of the surface of Mars as a function of surface elevation. While the northern hemisphere is well below the reference zero altitude, the southern hemisphere is mostly above. The southern hemisphere is on average six kilometres higher than the northern and contains older areological formations. Figure 3 depicts some of the past landed science missions as well as the envisaged landing altitude of the Mars Science Laboratory. From this figure it can be seen that most of the past missions reached landing sites in modest and easy-to-reach altitudes. Regions of interest are, nevertheless, located on the ancient highlands of the south. However, landing sites above 4 km altitude are relatively small parts of the whole planet. Landing missions should, therefore, aim for landing altitudes of up to 4 km. The altitudes were measured by the Mars Orbiter Laser Altimeter (MOLA). Future missions are, therefore, likely to aim for landings below this height.

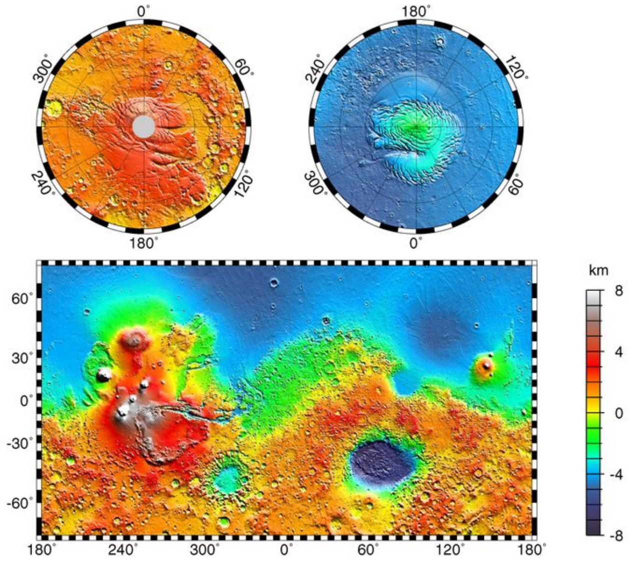

Fig. 2. Maps of Mars global topography (Courtesy NASA/JPL-Caltech)

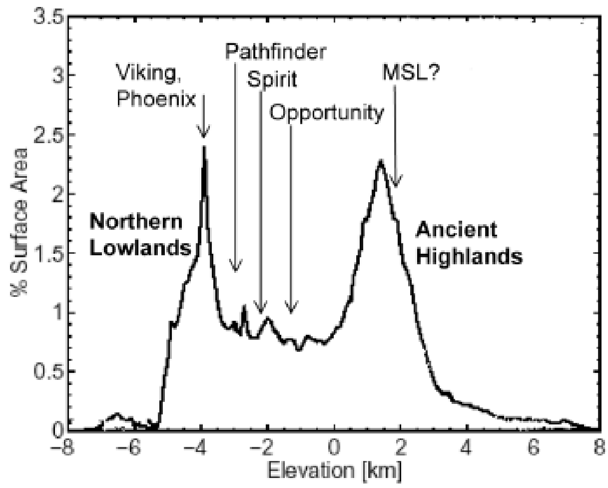

Fig. 3. Mars elevation area distribution [2]

Selection of a specific landing site depends on the goals of each individual mission and imposes constraints on the design of any landing system. An issue of importance is the hazard due to rocks in the landing area. The rock size and distribution defines the landing system as well as the desired method of landing (e.g. landing with a horizontal velocity component and a sled-like landing system).

The above definition of the maximum target landing site altitude defines a lower limit for the deployment altitude. Assuming a parachute deceleration of typical missions, the vertical distance covered during deceleration is about 800 to 1000 m. Accounting for deployment delays due to sensor uncertainties and other sources, a loss of altitude of 1500 m should be assumed for the deceleration. Combining these with the desired landing altitude the deployment must be performed at least at an altitude above 5.5 km.

From Fig. 1 it can be seen that for a parachute deployment at Mach 2.1, the minimum 5.5 km altitude can only be achieved by entry capsules with a ballistic coefficient less than 120 kg/m². The upper deployment altitude is not really limited despite the fact that higher altitudes at given Mach numbers are only reached for lower ballistic profile capsules. Therefore, the upper limit for a 30 kg/m² capsule would be 19 km. Such a capsule would very soon break through the upper dynamic pressure boundary and thus reach Mach 1.1 at a higher than allowed dynamic pressure. An upper limit of 19 km was, nevertheless, selected as the parameter range for further analysis. Table 1 summarizes the selected boundary conditions.

Table 1. Deployment conditions for parachute/auto-rotative EDL missions

Parameter	**Lower limit**	**Upper limit**
Ballistic coefficient (entry) [kg/m²]	30	120
Mach number [-]	1.1	2.1
Dynamic pressure [Pa]	250	1200
Deployment altitude [km]	5.5	19

In contrast to classical EDL missions, the mission sequence of an auto-rotation system is as follows:

- entry
- release of rotor cover
- deployment of rotor, establishment of auto-rotation state and deceleration
- release of heat shield (assuming traditional heat shield technology)
- glide phase
- flare
- touchdown

For a mission capable of meeting future requirements it will almost certainly be mandatory to provide full hazard avoidance capability and, therefore, the complete chain of EDL events should be designed to support guided flight operation. In order to ensure this type of operation, the orientation of the rotor disc of the deployed configuration must be fully controllable. Therefore, an auto-rotation based EDL system will have a high degree of autonomy and, when equipped with a dedicated sensor suite and a modern avionics system, it will have the following features:

- autonomous deployment of the rotor system as commanded by the guidance system
- guided flight to a dedicated target point
- autonomous performance of hazard avoidance manoeuvres (selection of new target point)
- flare manoeuvre to perform a soft and precise landing at the selected location

Transition from Entry to Descent

Transition from entry to descent takes place when the rotor is deployed. This is assumed to happen at Mach 2. Past missions tend to release the parachute at smaller Mach numbers but in order to reach landing sites at a high altitude an earlier deployment at higher Mach numbers is necessary. Rotor deployment is designed to take place at Mach 2 and at an altitude of approximately 10 km. The required target landing velocity is in the range from 10 m/s to 20 m/s. Figure 4 shows the velocity-altitude diagram for the transition to both landing velocities, the solid red lines indicating the velocity profiles for 10 m/s and 20 m/s landing speeds. It can be seen that after entry the capsule encounters the region of parachute/rotor deployment (blue zone). After deployment the velocity decreases until the vehicle achieves steady state descent. Depending on the target landing speed, the vehicle either decelerates faster at higher altitudes or slower at lower altitudes. The dotted line shows the entry trajectory to touchdown if no deceleration device is deployed. The final descent trajectories with 10 m/s and 20 m/s in this figure are assumed to be vertical.

It was found during first mission design iterations that it might be beneficial to use the lifting capabilities of the rotor even during deployment. However, at present it is not clear how this would interfere with the acquisition of auto-rotation and what the impacts will be on controllability. Therefore, it was assumed for the purposes of analysis, that during rotor deployment, the rotor disc will be perpendicular to the free stream and hence to the general motion of the vehicle. Glide and controlled flight will take place after the vehicle has been decelerated and a steady descent state has been achieved. During deceleration altitude is lost and, since it is an uncontrolled phase in terms of flight direction, down and cross range capability is also lost.

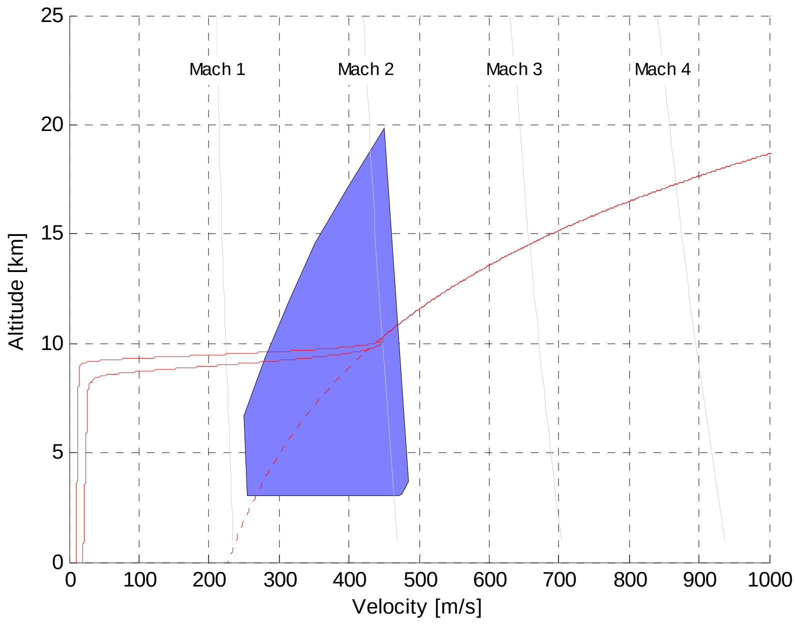

Fig. 4. Altitude vs. Velocity for reference mission ($\beta = 70$ kg/m²)

Glide Phase

During the glide phase it is possible to guide the auto-rotation system in a desired direction. Figure 5 shows the theoretically possible down range capability. For example, the lander could achieve a maximum down range of 28 km from an initial height of 8 km to reach a desired landing site at an elevation of 0 km assuming a lift/drag (L/D) ratio of 3.5.

However, two effects need to be considered. Firstly, if it is intended to land at higher altitudes, say 4 km, the possible achievable down range is reduced to 14 km. Secondly, if winds are encountered (which is most likely), then the down range capability might also be reduced since the wind alters the angle-of-attack and hence the L/D ratio. Furthermore, it might be necessary to perform a hazard avoidance manoeuvre, which, depending on the mission specific needs, imposes a minimum cross and down range requirement

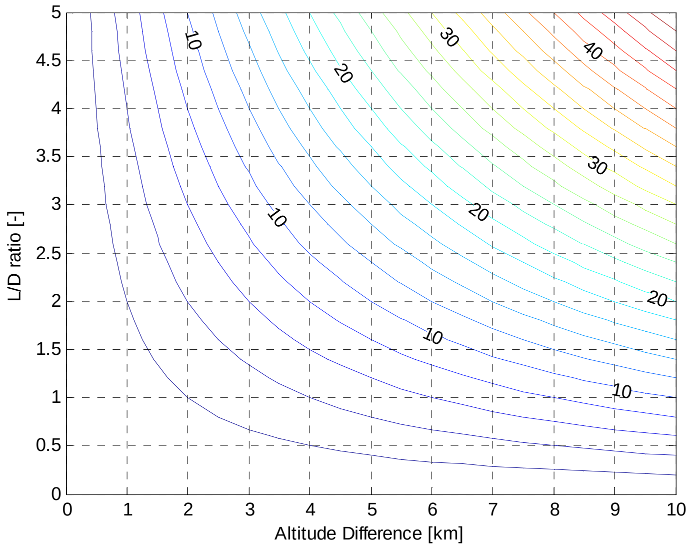

Fig. 5. Down range (km) capability a as function of altitude difference and L/D ratio

Figure 6 shows the down range capability of an auto-rotation system for a ballistic coefficient of 0.8 kg/m² and an 8 km altitude difference. It can be seen that, depending on the L/D ratio at lower wind velocities, flight against the wind is possible. When the wind increases, the down range capability is lost and even becomes negative. Changing the L/D ratio actively during flight by altering the angle-of-attack has a beneficial effect only on the down range if the drag is not increased.

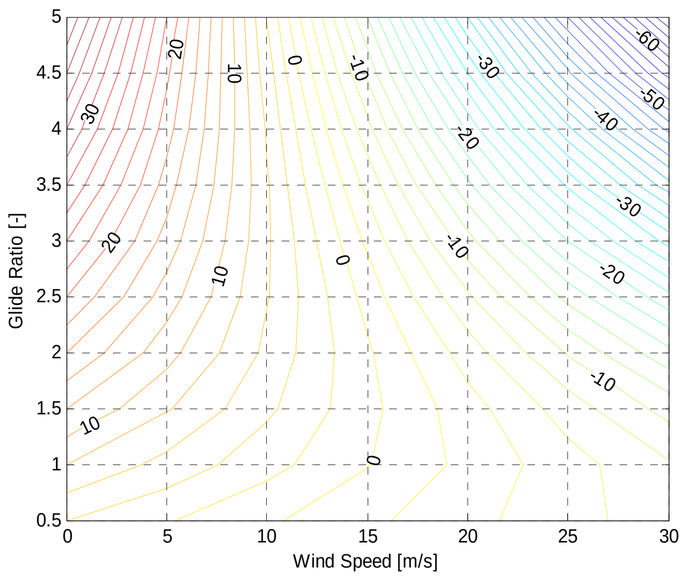

Fig. 6. Down range (km) as a function of L/D ratio and wind speed (ΔH = 8 km, β = 0.8 kg/m²)

Flare and Touchdown

The gliding phase could include a flare manoeuvre prior to the actual landing. This manoeuvre can be performed in two ways:

- by pitching up the whole vehicle and thus reducing vertical speed while simultaneously decreasing lateral speed (maximum horizontal velocity component reduction will be achieved when pitching against the wind direction), or
- by increasing the collective pitch angle of the rotor blades.

In principal, the two options might also be combined and performed simultaneously.

AUTO-ROTATION

The auto-rotation principle is based on the aerodynamic lift generated by freely-rotating (i.e. unpowered) rotor blades in forward and vertically-descending flight. Vehicles using this principle are termed autogyros. Unlike helicopter rotor systems, the autogyro rotor is mechanically simple and the blades do not necessarily require cyclic pitch control. Autogyros have been developed and flown with a moveable axis without cyclic pitch control or a fixed rotor axis with cyclic pitch control.

Auto-rotational landings of various types of vehicle have been conducted in terrestrial free flight trials and wind tunnels tests, and, practically, in emergency situations with helicopters since the 1920s. Interest in auto-rotation landing systems re-emerged in the 1950s with schemes for the recovery of air-launched payloads, rocket boosters and manned lifting bodies and capsules as an alternative to parachutes and parafoils. In the event only parachutes were adopted for the retrieval of re-entry vehicles; rotor systems being considered difficult to develop, especially if deployed during the re-entry phase (aerothermodynamic technology development was required), too complex mechanically and difficult to install.

The most extensive investigation of auto-rotation landing systems was done by Kaman Aircraft Corporation in the period mid-1957 to mid-1967. At that time Kaman was interested in developing these systems for air-launched payloads but suggested that the system was applicable to the terrestrial recovery of space vehicles such as manned re-entry capsules [3]. Other proposals for auto-rotation EDL systems also considered inflatable rotors [4, 5].

Figure 7 shows the interaction of design parameters with the dynamic and kinematic parameters influencing the design of an auto-rotation EDL system.

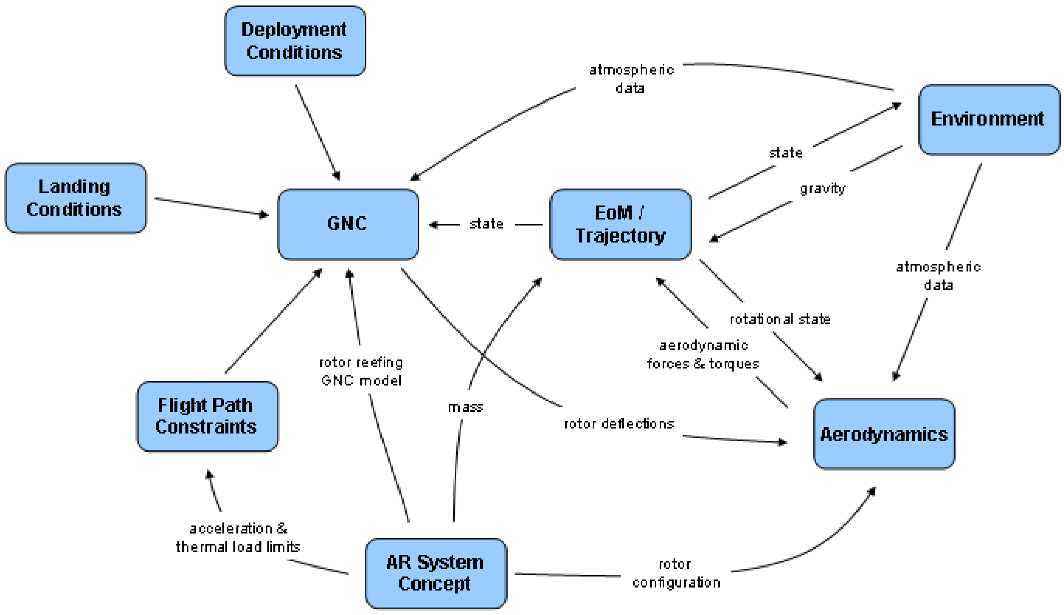

Fig. 7. Design parameters and their interaction for an auto-rotation EDL system

A comparatively simple calculation to determine rotor sizes to meet the requirements specified in the introduction to this paper can be based on estimates of the performance of a rotor system by applying momentum theory for an actuator disc model of a uniformly loaded rotor. The auto-rotation performance may be considered in terms of a drag coefficient based on rotor disc area and vertical descent velocity (leading to a drag coefficient $C_D \approx 1.1$).

Considering a vertical descent, the condition that the auto-rotation system drag should equal weight for the maximum permissible descent velocity at the elevation of the landing surface leads to the following simple formula:

$$d = \frac{1}{\mathrm{v}} \cdot \sqrt{\frac{8 \cdot m \cdot g}{\pi \cdot C_D \cdot \rho}} \tag{2}$$

where d = rotor diameter, v = descent velocity, m = landing mass including rotor system, g = gravitational acceleration, C_D = drag coefficient, and ρ = atmospheric density

Therefore, the rotor diameter required is inversely proportional to the maximum permissible vertical speed at landing elevation and proportional to the square root of the total landed mass (landing mass and mass of descent and landing system).

The main considerations for initial rotor sizing can be summarized as follows:

- The rotor diameter is basically determined by the desired vertical descent velocity.
- The desired vertical descent velocity 10-20 m/s (ESA Specification) is high. However, glide operations with a forward speed component as well as landing flare allow significant reductions of sink rate compared to vertical descent.
- Blade geometry (chord length, aspect ratio) is determined by the number of blades and the desired rotor solidity.
- Maximum permissible rotor RPM is determined by rotor blade tip Mach number limitation (Mach 0.8 … 0.9).

Assuming a single rotor system, a variation of key parameters reveals the dependencies shown in Figs. 8 and 9, the former showing the required rotor diameter versus total landing mass for a vertical descent speed range of 10 to 20 m/s und Mars conditions at 0 m MOLA.

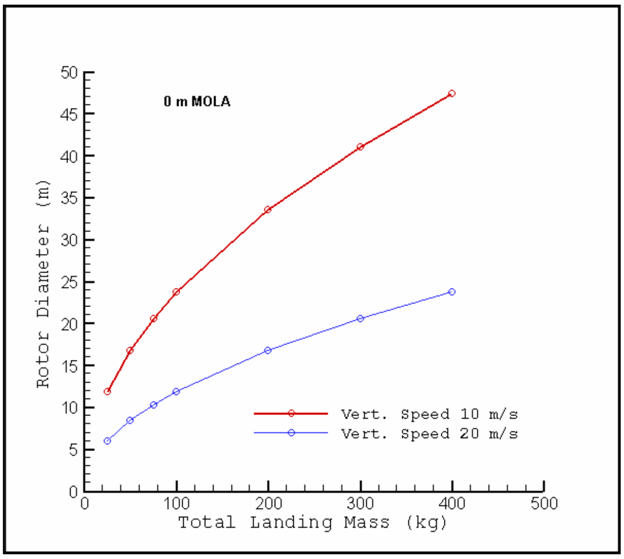

Fig. 8. Required Rotor Diameter vs. Total Landing Mass for a vertical descent speeds of 10 m/s and 20 m/s (Mars conditions 0 m MOLA)

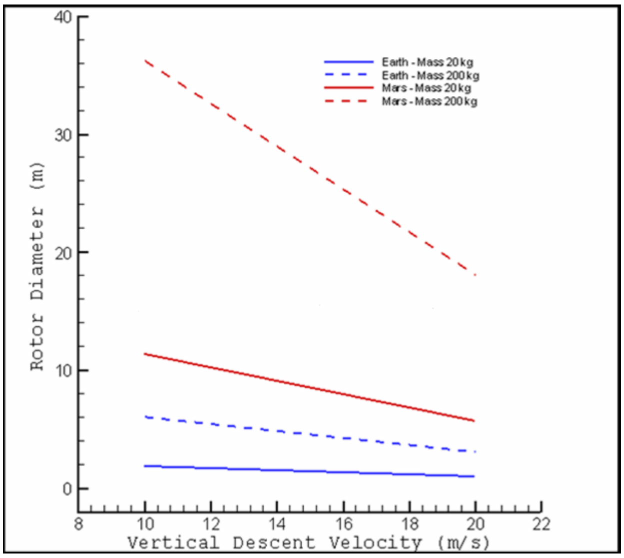

Fig. 9. Comparison of Rotor Diameter vs. Vertical Descent Velocity for Mars and Earth (Mars conditions 0 m MOLA)

Figure 9 shows the non-linear increase of the rotor diameter with increasing EDL system mass. The shape of the curve allows the conclusion that the relative weight and size required for rotor system based EDL system tends to favour such applications towards heavier lander masses.

AN INFLATABLE AUTO-ROTATION SYSTEM CONCEPT FOR EDL ON MARS

Two basic rotor concepts were examined using data derived from Figs. 8 and 9, namely, a “rigid” rotor system with telescopic blades (Fig. 10) and a fully inflatable rotor system (Fig. 11).

The potential advantage of the rigid rotor system might be seen in the fact that there exists a wealth of experience on the actual aerodynamic function of rotor systems made of rigid materials, at least for terrestrial applications. This also includes various rotor folding mechanisms to reduce hangar space required by larger helicopters. Of course, such folding mechanisms can in no way be considered fully comparable to the requirements for a rotor system to be unfolded in flight on Mars. Of relevance to telescopic rotor blade mechanisms, is the experience gained with of a high performance sailplane, the Akaflieg Stuttgart fs 29, which demonstrated the feasibility of varying the span of thin, high aspect ratio lifting surfaces by means of telescopic wings in flight. On the other hand, no actual experience exists regarding the aerodynamic operation and performance of inflatable rotor systems. However, inflatable fixed wing piloted and remotely-piloted aircraft have been demonstrated in actual flight under Earth conditions.

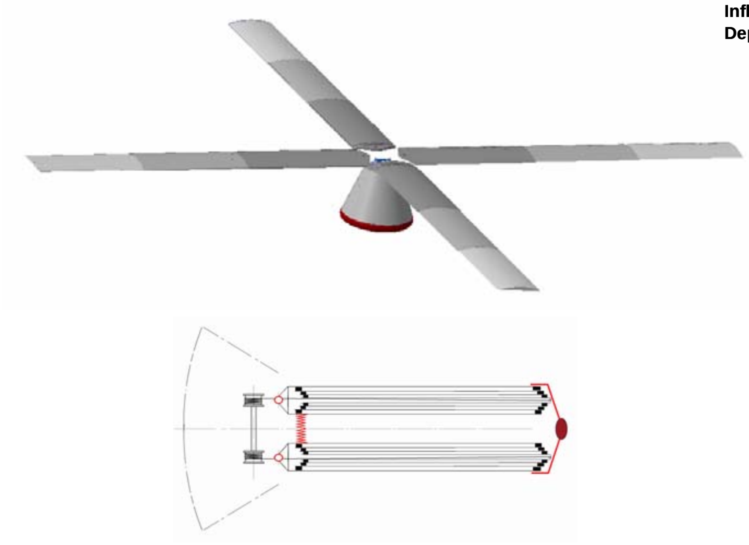

Fig. 10. “Rigid” rotor system with telescopic blades

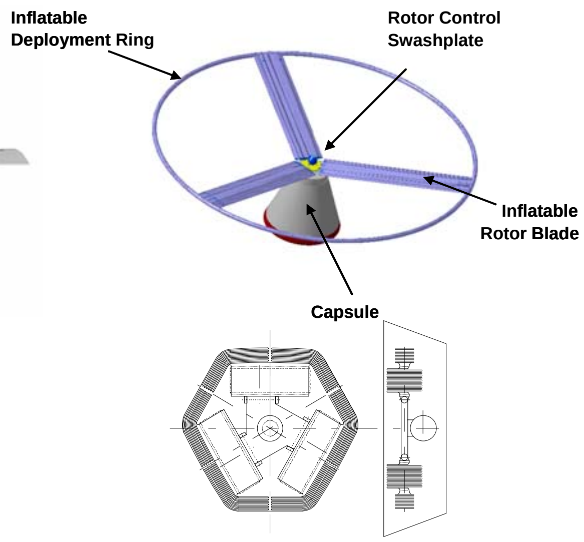

Fig. 11. Inflatable rotor system concept

A detailed trade-off between these two basic options clearly favoured the inflatable system both in terms of mass and packaging. The rigid rotor system proved much heavier than the inflatable system and had a mass almost as much as the capsule it was intended to land. The inflatable rotor mass/landed mass fraction is of the order of 15%. The size of the stowed rigid rotors required them to be housed in a tail-like fairing behind the entry vehicle, whereas the deflated inflatable rotor was stowed in a flat circular compartment within the capsule’s back shell. The inflatable rotor was selected as the more promising concept and a deployment test demonstration is planned to take place in the spring of 2009.

CONCLUSIONS

The present status of the study indicates that auto-rotation EDL systems are feasible within the limits specified by the requirements. Although the preferred concept relies on an “unconventional” approach, the authors consider that the risk associated with the development of appropriate inflatable rotor system technology is acceptable and can be realized.

Compared with “rigid” rotor systems, the proposed inflatable concept offers a low installation mass An auto-rotation EDL systems offers the possibility of performing precision landings at relatively high elevations and low descent speeds when combined with a flared landing.

ACKNOWLEDGEMENTS

The work described here is part of an on-going study being conducted under ESA Contract No. 21233/07/NL/CB. The authors thank ESA/ESTEC’s project manager, Dr. Johan Köhler, for his support.

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