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EADS astrium

EXECUTIVE SUMMARY

ARMADA

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1. INTRODUCTION

1.1. PURPOSE

The Entry, Descent and Landing System (EDLS) is one of the main system drivers for an interplanetary mission aiming at landing a payload on a planetary surface. The ultimate goal is to land safely the payload onto the planet surface. Towards that end, different constraints must be fulfilled in order to achieve a successful landing. In this sense, autorotation appears to be a promising conceptual approach to decelerating a probe in the atmosphere of a planetary body. A rotor would offer greater control over the descent of a probe than a traditional disk-gap-band parachute. Future missions that require visiting a specific location in a small area, or landing at a location with relatively hazardous terrain may well require such early and precise control capabilities. The ARMADA project investigates the feasibility of using an autorotation system to replace all elements of a traditional EDL system for application at Mars, except for the aeroshell (that is, parachutes, retrorockets and/or airbags).

The ARMADA study approach consists of three main themes that are mutually interrelated:

- General design of the layout of the autorotation system within the lander, including the deployment of the rotor
- Evaluation of the performance of the autorotation system by means of a software program designed for this task,
- Windtunnel testing of a model of the autorotation system, with a focus on the deployment system.

In general a strategy is followed that aims to keep the final design as close to current technological capabilities as possible, for the purpose of reducing to a minimum the amount of research and development needed to further mature the concept.

1.2. SCOPE

This document contains the executive summary of the project ARMADA. ARMADA has been an 18-month duration project, carried out by GMV, in collaboration with the University of Bologna and EADS-Astrium, under the frame General Study Program (GSP). The main objective of the project is to assess and demonstrate the feasibility of and autorotation based Entry, Descent and Landing system on Mars, although considerations for other planetary environments has been also analyzed.

This document gives an overview of the whole project, starting by the system concepts trade-off, followed by the description of the Software (SW) tools developed, demonstrators test campaign and summarizing the optimum configurations according to the Technology Readiness Level (TRL) timeline, together with the steps that led to them.

1.3. STRUCTURE OF THE DOCUMENT

This document follows the same general approach as the study itself. The different topics are dealt with in the respective chapters:

- Chapter 1 provides an introduction to the document, including definitions and acronyms
- Chapter 2 lists the applicable and the reference documents
- Chapter 3 provides a detailed description of the requirements considered in the study. The chapter lists the different configurations that may be derived from the requirement followed by a trade-off carried out on the layout and deployment concepts. Finally the selected concept and the back-up option are discussed in more detail.
- Chapter 4 gives a description of the two SW tools developed in order to assess the performance of the selected ARMADA concept, which are the Performance Database (PD) and the Integrated Parametric Design Tool (IPDT).
- Chapter 5 describes the windtunnel test campaign, including a description of the mock-up demonstrator design and construction as well as summary of their results.
- Chapter 6 presents the evaluation of the ARMADA concept for the Reference Scenario considered and for different TRLs.

- Chapter 7 outlines the conclusions extracted from the present study and describes briefly the near future roadmap for the ARMADA concept.

1.4. DEFINITIONS AND ACRONYMS

1.4.1. DEFINITIONS

Concepts and terms used in this document and needing a definition are included in the following table:

Table 1-1: Definitions

Concept / Term	Definition
M	Mach Number
L	Lift force
D	Drag force

1.4.2. ACRONYMS

Acronyms used in this document and needing a definition are included in the following table:

Table 1-2: Acronyms

Acronym	Definition
AD	Applicable Document
AGL	Above Ground Level
AoA	Angle of Attack
CDF	Common Data Format
DM	Descent Module
EDL	Entry Descent and Landing
EDLS	Entry Descent and Landing System
EMCD	European Mars Climate Database
GSP	General Study Program
HW	Hardware
IPDT	Integrated Parametric Design Tool
NASA	National Aeronautics and Space Administration
PD	Performance Database
RD	Reference Document
SW	Software
TPS	Thermal Protection System
TRL	Technology Readiness Level

2. REFERENCES

2.1. APPLICABLE DOCUMENTS

The following documents, of the exact issue shown, form part of this document to the extent specified herein. Applicable documents are those referenced in the Contract or approved by the Approval Authority. They are referenced in this document in the form [AD.X]:

Table 2-1: Applicable documents

Ref.	Title	Code	Ver.	Date
[AD.1]	ESA SOW: Autorotation in Martian Descent and Landing. Appendix 1 to AO/1-5422/07/NL/HE	TEC-MMA/2007/209	Issue 01 Rev. 00	July. 6th, 2007
[AD.2]	System Requirement Document For An EDLS Based On Autorotation	GMV-ARMADA-TN1100		2006
[AD.3]	Autorotation System Concepts	GMV-ARMADA-TN1200		2009
[AD.4]	Modelling Approach For The Performance Database Assembly	GMV-ARMADA-TN2100		2006
[AD.5]	Simulation and experimental test plan	GMV-ARMADA-TN2200		2006
[AD.6]	Development Of A Performance Database In Atmospheric Entry, Descent, And Landing	GMV-ARMADA-TN3100		2007
[AD.7]	Integrated Parametric Design Tool Development	GMV-ARMADA-TN4100		2007
[AD.8]	Armada IPDT User's Manual	GMV-ARMADA-TN4110		2007
[AD.9]	Autorotation Deployment System Design	GMV-ARMADA-TN4200		2007
[AD.10]	Simulation of Entry, Descent and Landing with Autorotation	GMV-ARMADA-TN5100	2.0	Sep 2009
[AD.11]	Rotor Deployment Proof-Of-Concept Manufacturing And Testing	GMV-ARMADA-TN6100		2008
[AD.12]	Armada System Performance Comparison With Competitive Technologies	GMV-ARMADA-TN5200		2008
[AD.13]	Conclusions and Recommendations	GMV-ARMADA-TN7100		Jul 2009

2.2. REFERENCE DOCUMENTS

The following documents, although not part of this document, amplify or clarify its contents. Reference documents are those not applicable and referenced within this document. They are referenced in this document in the form [RD.X]:

Table 2-2: Reference documents

Ref.	Title	Code	Version	Date
[RD.1]	Braun R.D., Manning R.M., Mars Exploration Entry, Descent, and Landing Challenges, Journal of Spacecraft and Rockets, vol. 44 (2).			April 2007
[RD.2]	Levin, A.D., Smith R.C., An Analytical Investigation of the Aerodynamic and Performance Characteristics of an Unpowered Rotor Entry Vehicle, Ames Research Centre.	NASA TN-D 4537		1968
[RD.3]	Levin, A. D. and Smith, R. C., Experimental aerodynamic performance characteristics of a rotor entry vehicle configuration. 1 – Subsonic.	NASA-TN-D-7046		1971.
[RD.4]	Levin, A. D. and Smith, R. C., Experimental aerodynamic performance characteristics of a rotor entry vehicle configuration. 2 – Transonic.	NASA-TN-D-7047		1971.

Ref.	Title	Code	Version	Date
[RD.5]	Levin, A. D. and Smith, R. C., Experimental aerodynamic performance characteristics of a rotor entry vehicle configuration. 3 – Supersonic.	NASA-TN-D-7048		1971
[RD.6]	Hagen, J., Rotor Landing Technology for CEV Earth-to-orbit Crew Transport, AIAA (Houston Section) 2005 Annual Technical Symposium.			May 6th, 2005
[RD.7]	MacNeal , R. H., and Kyser, A.C., ARC-R-219, The Spinning Filamentary Disk As A Hypersonic Decelerator, NAS 7-427, ASTRO Research Corporation, Santa Barbara, CA.	NASA CR-89684		August 25, 1966
[RD.8]	Young L.A. and Aiken, E. W., Vertical Lift Planetary Aerial Vehicles: Three Planetary Bodies and Four Conceptual Design Cases, 27th European Rotorcraft Forum, Moscow, Russia.			September 2001.
[RD.9]	Young, L. A., Vertical Lift – Not Just For Terrestrial Flight, AHS/AIAA/RaeS/SAE International Powered Lift Conference, Arlington, VA.			November 1, 2000.
[RD.10]	10Datta, A., et al., The Martian Autonomous Rotary-wing Vehicle (MARV), Alfred Gessow Rotorcraft Center, Department of Aerospace Engineering, University of Maryland, College Park, Maryland.			1 June 2000
[RD.11]	11Young, L. A., et al., Rotary-Wing Decelerators For Probe Descent Through The Atmosphere Of Venus, 2nd International Planetary Probe Workshop, NASA Ames Research Center, Moffett Field, CA.			August 23 - 27, 2004
[RD.12]	12Hall, J. R., et al., Deployable Aerodynamic Deceleration Systems, NASA SP-8066, Langley Research Centre, NASA.			June 1971
[RD.13]	Cagle, C.M. & Schlecht R.W., Composite Elastic Skins for Shape-Changing Structures, http://www.techbriefs.com/content/view/1113/34/	LAR-16599-1		2007
[RD.14]	Young L. A. et. al, Mars Rotorcraft: Possibilities, Limitations, and Implications For Human/Robotic Exploration.	IEEEAC paper #1274	3	December 16, 2004.
[RD.15]	Schuerch, H. U. and MacNeal, R., Deployable Centrifugally Stabilized Structures For Atmospheric Entry From Space, NAS w-652, ASTRO Research Corporation, Santa Barbara, CA.	NASA CR-69		July 1964
[RD.16]	MacNeal , R. H., and Kyser, A.C., ARC-R-219, The Spinning Filamentary Disk As A Hypersonic Decelerator, NAS 7-427, , ASTRO Research Corporation, Santa Barbara, CA.	NASA CR-89684		August 25, 1966
[RD.17]	Leishman, J. G., “Development of the Autogiro: A Technical Perspective”, in AIAA Journal Of Aircraft Vol. 41, No. 4.			July–August 2004
[RD.18]	Ewing, E. G.; Bixby, H. W. ; Knacke, T. W., Recovery Systems Design Guide, Irvin Industries Inc, Gardena CA.			1978

3. REQUIREMENTS AND CONCEPTS

3.1. ENTRY, DESCENT AND LANDING SCENARIO OVERVIEW AND SYSTEM REQUIREMENTS

The first objective of the Entry, Descent and Landing System (EDLS) is to safely land a payload with a mass between 20 and 200 kg on the planet's surface. The EDLS must decelerate the Descent Module (DM) from interplanetary velocities (typically a few km/s) to a velocity at landing between 10 and 20 m/s and land the payload within close proximity of a pre-defined landing site (ideally a few hundreds of meters).

Although the ARMADA project is aimed primarily at designing an EDL system for Mars, the implications on the system design of selecting a different target body are taken into account. Table 3-1 lists the properties of the planetary bodies of interest. The parameters that directly influence the design of the aerodynamic decelerator are the surface gravity and especially the atmospheric density. Mars has the lowest atmospheric density, and can therefore be considered the critical case.

Planet	R(km)	$g(m/s^2)$	T(K)	p(Pa)	$\rho(kg/m^3)$
Earth	6371	9.82	288.15	101325	1.23
Mars	3390	3.71	214	6.6	$1.55\cdot10^{-3}$
Titan	2575	1.354	94	149.52	5.55
Venus	6052	8.87	735.3	$9.21\cdot10^{6}$	64.79

Table 3-1: Properties of planetary bodies of interest

The Entry, Descent and Landing System (EDLS) is one of the main system drivers for an interplanetary mission that involves landing a payload on a planet's ground. It contains three subsystems with distinct functions designed for the Entry phase, the Descent phase and the Landing. The ARMADA system is mainly concerned with the descent and landing phases.

The entry phase is the phase during which the most stringent loads occur; the DM has to decelerate from hypersonic velocity (between 5 and 9 km/s) to supersonic velocity (about Mach 2) along a limited path length (on the order of 600 km, descending from an altitude of roughly 120 to 5-4 km).

During this phase, the deceleration is provided to the DM by its aeroshell. The DM's kinetic energy is converted into thermal energy. The enormous amount of heat is then dissipated away either by ablation or radiation (e.g. shuttles tiles get red). The aeroshell for the ARMADA system is identical to the aeroshell used for most other missions to Mars, namely, the Viking 70° conical heat shield. The requirements on the ARMADA system during this phase are identical to the requirements on other missions that have gone before. The main requirements are briefly summarized below:

- The entry probe must decelerate from the entry velocity (5.7 km/s from 120 km Above the Ground Level (AGL)) to supersonic velocity (Mach 2 at 8-9 km AGL) using only the aeroshell drag.
- The DM entry shall be ballistic.
- Based on Norcoat-Liege technology for Thermal Protection System (TPS) heat flux at stagnation point shall not be greater than 2000 kW/m^2.
- The peak stagnation pressure shall be less than 25 kPa.
- The peak heat load shall be less than 3815 J/cm^2
- The peak deceleration shall not exceed 8 g.
- The entry angle shall be smaller than the skip-out angle to avoid rebound onto the atmosphere.

The second deceleration stage is the descent phase. Since the success of the missions Viking, the descent stage is "traditionally" characterized by the use of one or several parachutes. Traditional parachutes (such as Disk-Gap-Band or hemisflo) can only be deployed within a given Mach-dynamic pressure envelope ([RD.1]). Similar constraints apply to the ARMADA system; these are identified through the windtunnel tests. The main concern for ARMADA is the rotor speed. High rotor speed leads to high centrifugal loads, and to high vibration loads. By way of contrast, terrestrial helicopters are designed to operate at a near-constant rotor speed.

The last phase of the EDL sequence is the landing phase. The vertical velocity is reduced to be within the horizontal and vertical landing velocity envelope, and the touchdown devices are deployed. The classical landing systems are based on the use of retrorockets, either with liquid or solid fuel, and more recently combined with the use of airbags. For the case of small landers, the vented or non-vented airbags are enough to absorb the impact energy due to the terminal velocity from the descent phase without the use of any other propulsion system. The uncertainties on the surface will affect the performance of such systems in a different way, for instance the airbags are very sensitive to the presence of wind gusts and the crushed-legs to the rock clearance. A detailed selection of the landing site in the mission design will minimize these risks. Nevertheless, the propulsive capability attained in this phase with the use of the retrorockets (Rocket Assisted Deceleration System – or RAD -) and steering rockets, as well as increasing the braking capability of the EDL system in this last phase, allows for re-targeting manoeuvres, so certain hazard avoidance can be done if this is combined with terminal guidance systems.

The ARMADA system is required to replace all elements of the landing system, so no retro-rockets or airbags can be used. Instead, the ARMADA system should use a flare manoeuvre to reduce the vertical velocity to between 10 and 20 m/s. Like terrestrial helicopters, the ARMADA rotor system can be used to perform hazard avoidance and retargeting manoeuvres. For this reason, the navigation system is expected to be highly similar to the one that would be used in combination with the traditional EDL system.

3.2. LAYOUT TRADE STUDY

The first part of the overall design concerns the layout of the system. As is the case for helicopters in general, many configurations are possible. For ARMADA, the only real issues are that the stowed system does not enter into the space reserved for the payload, that it leaves as much space for the payload as possible, and that the stowed system does not cause the vehicle to become unstable by disturbing the flow around the aeroshell, or by causing an unfavourable mass distribution. The following layout concepts have been considered.

- **Top Rotor**: the rotor at the top configuration is the design concept used in most studies; most notably the studies conducted during the Apollo program ([RD.2], [RD.3], [RD.4] and [RD.5]) a study performed by Jeff Hagen ([RD.6]).
- **Bottom Rotor**: the rotor at the bottom is comparable to the rotornet hypersonic decelerator ([RD.7]). The rotornet is implemented as a flexible, filamentous net that is wound around the rotor hub. Conventional blades can be used as well, although rotor tilting will be more difficult, and the rotor hub is likely to be heavier than the top rotor.
- **Side-by-side Rotors**: two rotors mounted side by side could enhance the controllability of the autorotation system. More deployable elements are needed for this concept. The number of rotors can also be extended to three, increasing the controllability, but also complexity and mass.
- **Coaxial Counter-Rotating Rotors**: two counter-rotating rotors mounted on the same axis. Both rotors can be controlled independently. Studies for powered, extra-terrestrial vertical lift vehicles often use this layout ([RD.8], [RD.9], [RD.10]). Counter-rotation offers the advantage that the torques of the powered rotors ideally cancel each other out. Control around the spin axis can be achieved by changing the collective pitch angle of each rotor independently.
- **Synchropter** (meshing rotor blades): the rotor blades could be deployed in such a way that the rotors mesh. For a powered rotor, the compactness can be an advantage. The configuration creates a smaller rotor area, so interference effects of the lander body are likely to be greater. Also, the mechanical complexity is increased, since gearing is required to make the rotors mesh.
- **Side-mounted Rotors**: this configuration uses small rotors with single-piece blades, mounted on the side of the body. This concept has been explored in the study of an autorotation system for descending into the atmosphere of Venus [RD.11].

The top rotor is mechanically the simplest solution, and it is also the lightest solution. The bottom rotor will be somewhat heavier, since the rotor is mounted on a ring structure instead of a central hub, and multiple rotors will most certainly be heavier.

The top and bottom rotors can both conceivably be stored within the space allotted. Multiple rotors will require more space.

The top rotor is least complex mechanically, while it is also closest to traditional helicopter designs, making it the most reliable concept. The bottom rotor is a more complex system, although it would still allow a reasonable technical design. Dual and multiple rotors require additional complexity because all of the rotors need to be deployed and controlled in some way. The worst solution is the meshing rotor: since the rotor is not driven by power applied to the shaft, there is no reason for the additional complexity of a system to make the rotors mesh.

The bottom rotor faces some stability problems, since the vertical distance between the point of thrust and the centre of gravity is small. The other rotors are mounted on the top, so that they have a higher inherent stability.

Multiple rotors allow an easy method of lateral control by means of differential braking of the rotors. The other concepts require other, more complex forms of control mechanisms. The co-axial counter-rotating rotor can control all attitude angles simultaneously, so this concept also receives a score of three.

The bottom rotor can deploy the largest effective rotor area. The top rotor can deploy the second largest rotor area. For these two concepts, most of the stowage space can be used to stow deployable blades. For the other concepts, stowage space is required for deployable support structures supporting the rotors. The dual rotor can probably still deploy an appreciable rotor area. The synchropter has a reduced effective area, because two rotors area meshing, which reduces the effective area. The co-axial rotor also has a reduced effective area, since both rotors are co-axial. The multiple-rotor concept will have the smallest rotor area.

Considering all these advantages and disadvantages in a trade study leads to the top-rotor as the preferred concept.

3.3. DEPLOYMENT TRADE STUDY

Another important aspect of the autorotation landing system is the rotor deployment. The deployment of an autorotation system is complex compared to other types of aerodynamic decelerators such as parachutes or parafoils, and requires a relatively bulky system [RD.12]. A trade-off on the deployment system needs to focus on the relative complexity and mass of the system. Different types of rotor blades deployment systems can be envisaged in order to achieve blade deployment during a Martian atmosphere EDL:

- **Single-piece blade rotor**: single piece blade rotors are by necessity relatively small. Multiple rotors are required for an application at Mars, seeing that a large rotor area will be required, and stowage space is limited. This can be a good option for thicker atmospheres, where the length of the blades may be much smaller.
- **Telescopic blade rotor**: The rotor is composed of several (up to 3-4) tubular sections, which can be extended. The deployment is quite easy, and can be controlled by means of cables running inside the blades [RD.6].
- **Inflatable blade rotor**: The rotor is composed of a gas-pressurized coated fabric, which can be rolled up for stowage. A major advantage is the low weight and the high expansion ratio of the structure, although the low weight also leads to a low kinetic energy storage capacity, diminishing the effectiveness of the flare manoeuvre.
- **Foldable blade rotor**: the rotor is folded in multiple sections. Special care needs to be taken to lock the structure in place to ensure the stiffness of the overall structure. Deployment requires an actuator, either active (engine) or passive (spring, aerodynamic forces).
- **Flexible blade rotor**: the rotor is made out of a flexible material that can be folded or rolled up. The structure is stabilised by means of reefing lines, extendable stiffeners, and / or centrifugal forces (by means of a mass placed at the tip) in the deployed configuration.

Any deployment system that can modify the blade length in-flight is highly desirable, since this allows an easy method of reefing. Telescopic blades, (mixed construction) flexible blades and inflatable blades are all capable of supporting such a partial deployment, although telescopic and mixed construction flexible blades seem more suitable for this purpose than inflatable blades, which probably perform discontinuous modifications to the blade length.

Conflicting needs exist between the rotor mass and the ability to store kinetic energy that requires a large polar moment of inertia. Deployment concepts that score high one of the criteria typically score low on the other, although tip masses could be added to increase the inertia. Considerations of the stability of the structure during deployment, and the rigidity of the structure speak out against the flexible-blade and the inflatable blade rotor, although smart materials with built in actuators currently under development may alleviate this problem ([RD.13]). Such materials, and other kinds of actuators built into the blades may also substantially reduce the mass of the control system ([RD.14]).

The most promising concept is a telescopic blade rotor. The runner-up, which would be a combination of flexible blades with a form of rigidization, is functionally very similar to the telescopic blade. It should be noted that flexible blades without rigidization have a number of problems that require an engineering approach leading away from traditional helicopter design [RD.15], [RD.16].

3.4. ARMADA CONCEPT

The basic concept for ARMADA consists of a telescopic rotor mounted at the top of a Viking-like aeroshell. Figure 3-1 shows the general configuration of the ARMADA system.

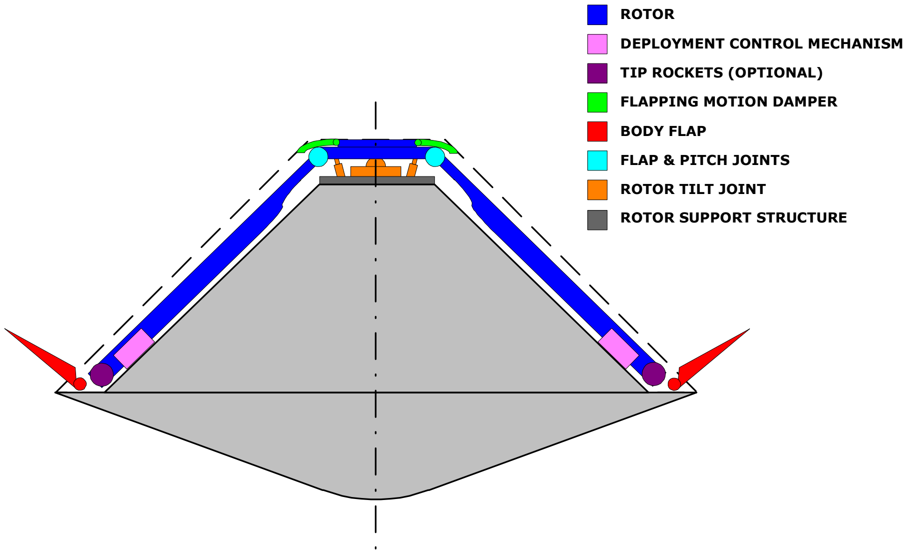

Figure 3-1: ARMADA Autorotation system concept

The autorotation system is installed below the back shell of the lander. The rotor is attached to a conventional helicopter control system, which is itself mounted to a structure that carries the payload.

The deployment occurs in a number of steps. The deployment starts by extending the body flaps, which are used to stabilize the vehicle during the deployment process. Next, the blades are deployed using a deployment cable to limit the forces on the joints. During this process, the blades are set in a zero-lift configuration to prevent rotation. Rotation is further prevented by a braking system on the rotor shaft. When the blades are fully deployed, the retention cables are released, the rotor brake is disengaged, and the collective pitch of the blades is set to an angle that allows the rotor to spin up. As the lander crosses a certain predetermined Mach number, the telescopic blades are extended by means of the centrifugal force (the blades are initially held in the retracted position by means of a cable). From this point onward, the craft descends in its fully deployed and fully extended configuration, and descends at its equilibrium descent velocity. Just before landing a flare manoeuvre is performed, which further reduces the vertical and horizontal velocity.

The sizing of the rotor is performed using a basic equation that links the rotor radius to the total weight of the vehicle, the atmospheric density and the descent velocity:

$$R = \sqrt{\frac{2mg}{C_R \pi \rho V^2}} \qquad (1)$$

This equation can be obtained by equating the lift force generated by the rotor to the weight of the vehicle when the vehicle is performing an equilibrium descent ([RD.17]). The rotor drag coefficient appearing in this equation is equal to about 1.25, although a more conservative estimate of 1 has also been explored. NASA2 even reports a drag coefficient as low as 0.25, but this value is probably the result from the rotor operating under non-ideal, stalled conditions. The drag coefficient of 1 – 1.25 is higher than the drag coefficient for parachutes used in traditional EDL system. Typical values for disk-gap-band parachutes ([RD.1]lie between 0.4 and 0.7, and for other EDL ([RD.18]) parachutes below 0.9. This makes a rotor a more efficient decelerator than traditional parachutes.

An iteration on the design established that a Martian lander using this concept would likely experience problems with the size and mass of the rotor and the deployment and the control system. These systems would by necessity form a substantial fraction of the total vehicle mass. A simple model for the mass of the lander can be established by assuming that the vehicle weight is the sum of the mass of the payload, the rotor and the control system:

$$m = m_{payload} + m_{rotor} + + m_{control} \qquad (2)$$

Dimensional analysis suggests that the mass of the rotor and the mass of the control system are related to the rotor by means of a power law.

$$\begin{cases} m_{rotor} = C_{rotor} \cdot R^{\alpha_{rotoro}} \\ m_{control} = C_{control} \cdot R^{\alpha_{control}} \end{cases} \qquad (3)$$

In this case, a mass exponent of 2 indicates that the mass of the rotor is related to the surface (or the skin) of the rotor blades, while a mass exponent of 3 indicates that the mass of the rotor is more related to the volume of the blade. An analysis of current rotors yields a mass constant of 1.6 and a mass exponent of about 2.6. The mass constant of the control system is 2.75 with a mass exponent of 1.8.

It is then apparent that the scaling law that links the mass of the rotor to its radius, contain the radius of the rotor raised to a power that lies between two and three, which means that larger rotors become proportionally heavier. The mass fraction of the autorotation systems lies between about 20% and 40% for the small lander, and between 30% and 50% for the large lander.

	mass(kg)	fraction	mass(kg)	fraction
Payload	20	0.41	200	0.2
Lander total	48.7	1	1013.6	1
Aeroshell	11.4	0.23	236.9	0.23
Autorotation system	17.3	0.36	576.6	0.57
Rotor	8.54	0.18	441.91	0.44
Landed mass	37.3	0.77	776.6	0.77

Table 3-2: Mass budget for a small (left) and a large (right) autorotation lander

Table 3-2 shows the mass budget of a small and a large lander, for a mid-term scenario. The rotor consists of four blades of 5 sections each. Table 3-3 shows the principal dimensions of both landers. For both landers, the equilibrium descent velocity needed in (1) is set to 36 m/s, which is somewhat too high when compared to the upper limit of the landing velocity requirement of 20 m/s.

	Small	Large
Heat shield radius (m)	0.38	1.73
Rotor radius (m)	1.9	8.7

Table 3-3: Principal dimensions of autorotation landers

These results indicate that attaining a velocity that is within the requirements is difficult using a telescoping rotor. Adding rotor sections will not solve the problem, because in this way the rotor mass increases and the total vehicle mass enters into a vicious cycle. It should be noted, however, that both the rotor mass and the control system mass are determined based on quite conservative mass estimation relationships. Other studies make more optimistic assumptions on the rotor mass ([RD.14]).

The deployment system needs to be able to withstand the forces that occur during deployment and the reefing, while the control system needs to be capable of changing the pitch setting of the rotor blades in all phases of the flight. The centrifugal force on the reefing system turns out to be the largest force within these systems. It will be necessary to avoid loads that are too high for the system to cope with; the most effective way of doing this is to place an operational constraint on the rotor velocity.

Finally, if the baseline design is applied to different planetary bodies, the results are quite striking: for Titan and Venus, a rotor consisting of 4 blades of one section each is capable of reaching a descent velocity well below 10 m/s, suggesting that the autorotation landing system is quite a viable option for these bodies.

4. SOFTWARE TOOLS

4.1. PERFORMANCE DATABASE

A number of software tools have been developed for the ARMADA project. These are a **Performance Database (PD)** and an **Integrated Parametric Design Tool (IPDT).** Taken together, these two tools allow a user to explore the design of an autorotation system for planetary EDL with considerable ease. Figure 4-1 shows the high-level architecture of the PD and the IPDT. The PD is coded in FORTRAN 77, while the IPDT is an MS Excel workbook.

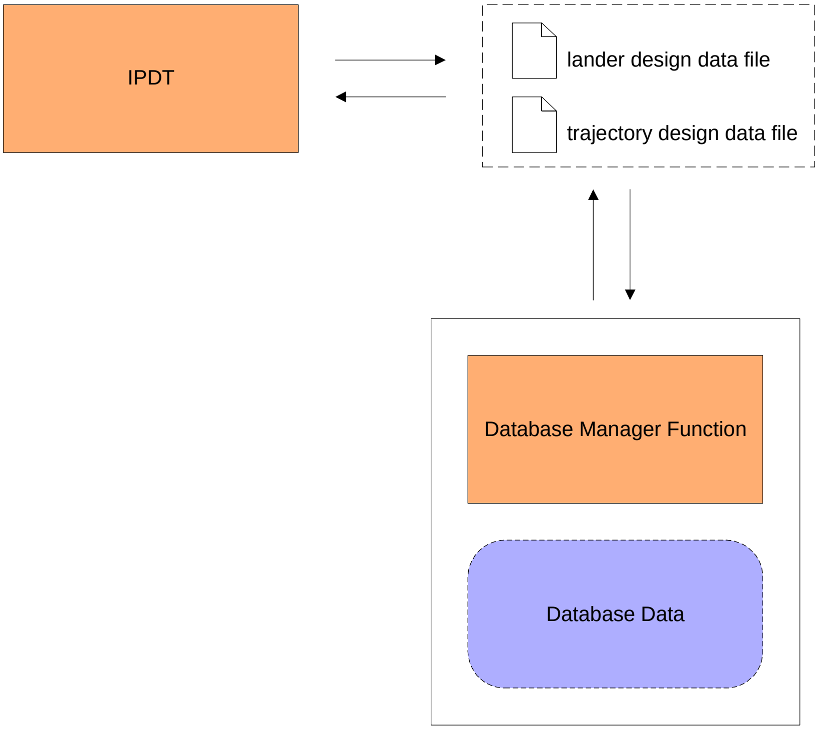

Figure 4-1: PD/IPDT Software Architecture

The PD developed for the project implements the ARMADA design concept baseline (Top rotor with telescopic blades) and its back-up option (Bottom rotor with telescopic blades). The database has been implemented using a computing database architecture, which has been selected due to the requirements of broad applicability to different scenarios and flexibility demanded in terms of mission scenario configuration. Another main characteristic of the PD is its high modularity, presenting significant advantages such as:

- Easy modification/addition of existing/new modules, that is, easy maintenance and reusability
- Clear interfaces between the modules guaranteeing the validation and data management between blocks
- Direct translation of the relevant autorotation system/performance parameters into the modules.
- Capability of operating the modules as stand-alone functions as much as possible.

EDL flight phases are modelled and implemented separately, thus the response of the autorotation system can be analyzed independently, although all of them are properly connected in order to obtain the characterization of the entire trajectory. The transitions between flight phases can be triggered by user-selected parameters that depend on the environment (for example pressure and density) and state conditions (position and velocity). The independent variables on which the control of the autorotation system depends can be selected from the same list of parameters with which the flight-phase transitions are triggered, since usually those transitions are directly related to control operation modes (deployment, reefing, flare manoeuvre).

Additionally the PD includes auxiliary functions to help the user when defining or analyzing the autorotation system. The definition of the environment conditions allows the user amongst others to set the atmospheric profile. System engineering of the autorotation concept allows the user to perform a bottom-up calculation of the masses of various components of the autorotation lander. The auxiliary parameters estimation can be used for example to compute the thermal/mechanical response.

The following list describes briefly the modules present in the PD:

- PD driver in charge of controlling/managing the execution of the PD. The PD includes different execution modes:
- Trajectory propagation that allows evaluating a single scenario configuration for one/all of the flight phases.
- Auxiliary computations mode to trigger the support functions
- Sensitivity mode that enables running up to 10 trajectories to study the influence of a parameter in the current scenario, so the output data may be compared for the different values and thus the response of the system against the parameter may be estimated.
- System engineering is used to derive all the lander mass/inertia properties, as well as for retrieving all aerodynamic data.
- Montecarlo, which is envisaged to study variations in the density profile and initial entry conditions.
- Environmental model that computes the atmospheric properties such as density, pressure, temperature and velocity of the wings, and the gravity acceleration. Atmospheric properties are stored in binary CDF files and are accessed using the netCDF libraries of the User-Defined and EMCD models. The MARS-GRAM model is also available. The winds are only considered during the landing.
- Trajectory phases modules that models the dynamics and kinematics and perform the propagation in each one of the flight phases. This module is divided into the four phases identified in the beginning of the project as part of the autorotation system operation: entry, deployment, descent and landing.
- Trajectory transition and check module in charge of switching between the different phases when a trajectory propagation involving more than one phase is requested. The transition, defined by the user, may be function of the environment of autorotation state variables.
- Lander module representing the autorotation system model. This module computes the mass and inertia properties for a given configuration as well as the forces and moments acting upon the vehicle for that configuration in a given flight condition. The aerodynamic actions are computed by means of integrating the flow-field over the rotor using the bladed element theory. The turbulence region of the rotor is modelled as a lower effective rotor section. The aerodynamic coefficients are function of the local Mach number and angle of attack. They are stored using the binary CDF format and are accessed using the netCDF libraries.
- Control module that implements simple open loop controls laws. The control devices available depends on the flight phase,
  - Entry: there is no control
  - Deployment: deployment angle, supersonic reefing law, and collective pitch angle setting
  - Descent: Subsonic reefing law, and up to 5 different laws for the collective and cyclic pitch angle setting
  - Landing: Flare manoeuvre initial time, cyclic and collective pitch setting angles and tip-rockets ignition time.
- Support design functions module that includes the above mentioned auxiliary functions.

As a support for the development of the PD, a CFD analysis has been performed, using commercial dedicated software, Fluent®. The aim is to provide aerodynamic data for a vehicle equipped with an autorotation system, to complement other available sources, i.e. literature data, experimental campaign on the hardware demonstrators, and the results from the analytical tool for ARMADA trajectory analysis.

The CFD tests have been planned following a modular approach from low level components (blade airfoil sections) up to high level capsule plus rotor configuration, in order to get physical insight in some not-trivial aspects of the study, such as supersonic rotor vehicle flow field structure. In particular the simulations are organized in the following four steps:

1. 2D airfoils tests: determination of lift and drag coefficients as a function of Mach num-ber and angle of attack.
2. Capsule (Viking-shaped aeroshell) analysis: lift, drag and pitching moment coefficients a s a function of Mach number and angle of attack.
3. Rotor in a supersonic flow field and axial flow (rotor angle of attack equal to 90°).

4. Capsule with rotor in supersonic (M=2) and high subsonic (M=0.8) regime and axial flow (rotor angle of attack equal to 90°).

The results of the first two analyses can be directly input into the trajectory analysis tool together with the corresponding experimental data, while the last two are mainly aimed to get an evaluation of the interference between rotor and capsule.

The PD was coded in FORTRAN 77, since this is a very well suited programming language for engineering applications and because there is a large previous experience of the team with language. The availability of the astrodynamic libraries in F77, such as the EMCD and thermal models was also a determining factor, because it means a significant reduction of the implementation time. The PD was compiled using a standard F77 compiler so it can be used on any PC without the need of additional SW. The input/output interface with the PD is done through plain text files, which can be modified by the user with a simple editor in a straightforward manner. These files may be also written from a simple GUI generated with any language program or application (MATLAB, MSEXCEL, Java,…). This is in essence the function of the IPDT. The IPDT is the user interface that provides the user input to the database and receives the output, and presents the output in a more understandable and manageable way.

4.2. INTEGRATED PARAMETRIC DESIGN TOOL

The IPDT for the most part consists of interface sheets that interact with the database, except for the lander preliminary analysis worksheets that contain the preliminary analysis outlined above.

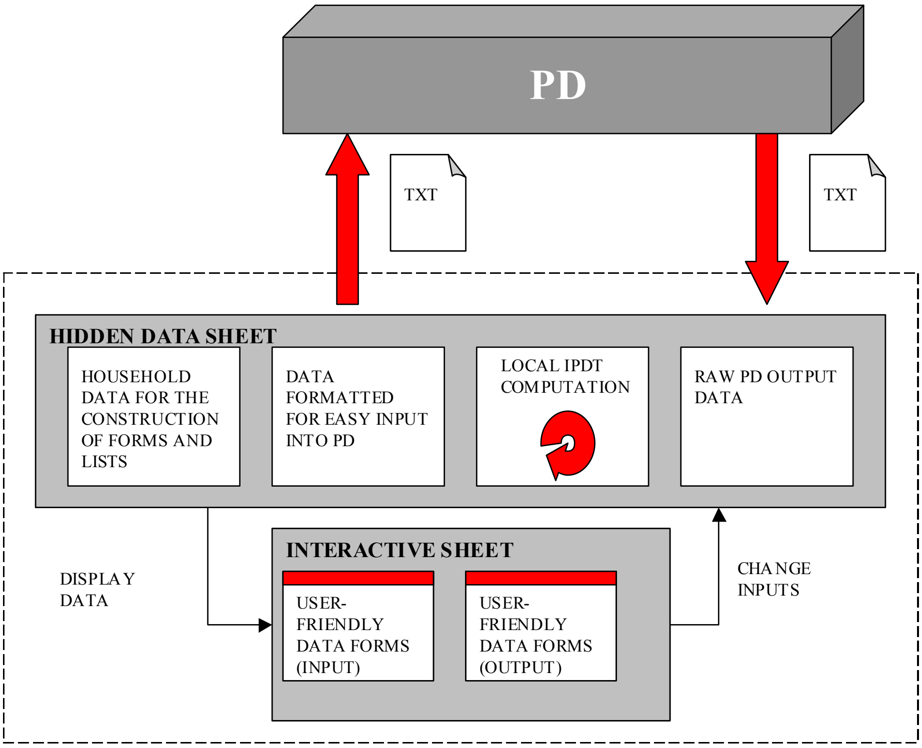

Figure 4-2: IPDT Architecture

Figure 4-2 shows the general architecture of these worksheets. The user only sees the interactive worksheets at the bottom left. This sheet contains the input / output from the database presented in a user-friendly format. Most of the sheet will be locked to prevent inadvertent modification of the layout or data descriptions. The user will only be able to modify the cells requiring input and the interactive control buttons. Note that the hidden data sheets do not correspond one-to-one to each of the interactive sheets; instead they correspond to the general layout of the input and output text files.

5. WINDTUNNEL TESTING

5.1. MODEL DESIGN AND CONSTRUCTION

The feasibility of the ARMADA rotor deployment system is demonstrated by means of a windtunnel testing campaign. The two most critical aspects of the deployment of the autorotation system are the initial deployment and the reefing of the rotor. Testing these two aspects is the most important objective of the windtunnel testing. The secondary objective of these tests is to obtain measurements of the aerodynamic coefficients and the rotation speed of the rotor, for the purpose of validating the analytical models. In developing the mission timeline it has been determined that these two events occur at different flow conditions: the deployment will occur at a supersonic speed, while the reefing occurs in the subsonic flow regime.

The small size of the windtunnel models, and the fact that the tests are performed in two different windtunnels, has lead to the decision to separate the two functional aspects in two distinct models. Figure 5-1 shows the supersonic deployment model. This model features four deployable blades, that deploy to a predetermined, fixed blade pitch setting. The tests are performed in the S-1 supersonic/transonic wind tunnel at Von Karman Institute of fluid dynamics, which is capable of performing tests up to M = 2.

Figure 5-1: Supersonic deployment model

Figure 5-2 shows the subsonic reefing model. This model features four telescoping blades of three sections each. This model is tested in the wind tunnel of the Applied Aerodynamic Laboratory of University of Bologna, at Mach numbers ranging up to about 0.11.

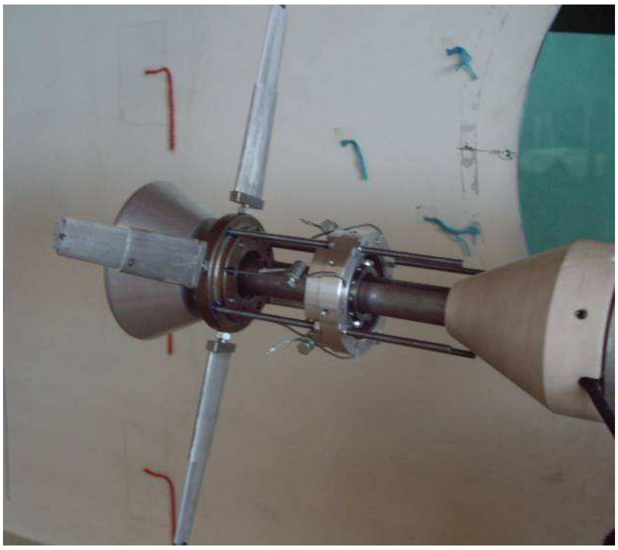

Figure 5-2: Subsonic reefing model

Table 5-1 summarizes the geometrical properties of the windtunnel models.

The sizing of the models is constrained on the one hand by the size of the windtunnel test section and on the other hand by the need to include mechanisms. However, during the design of the models, care has been taken to ensure a good degree of flow similarity between the tests and the actual flow as would be present on Mars. Scaling parameters that play a role during the experimental tests are:

- Mach number, Reynolds number and specific heat ratio for transonic/supersonic tests;
- Reynolds number for subsonic tests (incompressible regime).

The only parameter involving model dimensions is the Reynolds number. Flow similarity computations showed that the subsonic model would be about 49 times smaller than the actual vehicle, meaning that the actual vehicle would have a diameter of about 4,4 meters. This turns out to be a little larger than the dimensions of the heat shield of the large, 200 kg-payload lander reported The supersonic model would need to be between 10 and 13 times larger than the test model, leading to a heat shield diameter of between 36 and 47 cm. This is somewhat smaller than the heatshield dimension of the small, 20 kg-payload lander reported in Table 3-3. Although differences exist between the flow in the windtunnel on Earth, and the flow that would occur during the landing sequence of Mars, the effect on the aerodynamic coefficients was found to be small, based on the CFD simulation campaign reported in the previous chapter.

Description	Supersonic	Subsonic
Heat shield diameter	36 mm	90 mm
Back shell angle	65º	65º
Body height	24 mm	53 mm
Total height	28 mm	88 mm
Number of blades	4	4
Number of blade segments	1	3
Rotor radius (before reefing)	30 mm	94 mm
Rotor radius (after reefing)	N/A	N/A
Effective blade length (before reefing)	17 mm	60 mm
Effective blade length (after reefing)	N/A	N/A
Joint fraction	N/A	~11%
Blade expansion ratio	N/A	2.47
Rotor solidity	0.21	0.128
Taper ratio	untapered	0.508
Blade section chord	5 mm	24 mm
Blade section thickness	1 mm	5 mm
Blade skin thickness	solid	0.75 mm
Rotor to capsule diameter ratio	1.67	4

Table 5-1: Summary of characteristics of the models

5.2. WINDTUNNEL TEST RESULTS

The supersonic/transonic tests were only partially successful due to repeated failure of the model(s). This has been mainly ascribed to fatigue induced by the extremely high rotational speed of the rotor. It has been established empirically that a rotor operating in the supersonic regime experiences severe load conditions. If this is true in general, the high level of miniaturization reached for the present experiments makes it even worse (rotor speed is proportional to 1/R). In other words, miniaturized rotary entry vehicles undergo higher loads, both static (magnitude of centrifugal force) and dynamic (level of vibrations).

Rotor deployment with blades at a constant pitch attitude has been demonstrated at transonic speed and recorded in the form of a Schlieren movie. The Schlieren system showed that the proposed configuration of ARMADA rotor in supersonic regime is almost fully encompassed inside body wake.

Rotor operation at high speed has also been tested (and recorded), up to Mach 2; variation of AoA around the 0° attitude qualitatively showed that the rotation operation is only marginally affected by a perturbation in attitude.

Vehicle drag has been measured at Mach 2: a comparison between present experiments and IPDT output revealed that the analytical model implemented for rotor analysis provides a fairly accurate drag estimate; rotor angular speed is a bit overestimated by IPDT, but the overall level of accuracy is deemed sufficient for the purposes of a conceptual study. However, since only one valid encoder measurement has been achieved, further investigation of this aspect is recommended.

The measured tip speed ratio is also comparable with the results of NASA experiments at supersonic speed. The tip speed ratio is the speed of the blade tip divided by the free-stream velocity.

It is recognized that, on one side, higher angular speed are beneficial for the performance of the system, meaning greater amount of kinetic energy stored in the rotor during spin-up. On the other side, however, the attending increase in static and dynamic loads should be carefully taken into account and may affect several aspects of the design.

Subsonic tests proved that telescopic blades are a viable concept: reefing has been performed in several tests ranging from 0° to 15° of vehicle incidence, showing good repeatability.

Rotor speed measurements have been recorded throughout, revealing the bivalent behaviour of the rotor, which may operate in stalled (unefficient) or unstalled (efficient) mode. Transition from one mode to another depends on blade length and wind speed. A similar behaviour is also found in the NASA studies [RD.2], [RD.3] and clearly emerges from the trajectory simulations performed with the IPDT (see also the performance evaluation. The rotor drag coefficient is highly affected by stalled or unstalled operation modes, thus having a major impact on the attainable touchdown velocity of the vehicle.

The aerodynamic performance has also been measured at fixed blades lengths; the results indicate the following main conclusions:

- The maximum drag coefficient of the configuration is ≈1 with full-length blades in unstalled operation.
- Maximum (in modulus) L/D ratio is ≈-0.2 at 20° AoA.
- The vehicle is stable in pitch in the tested range of incidence.
- Maximum tip speed ratio measured is ≈6.5.

It has been shown that measurements from subsonic experiments and output from analytical model as implemented in IPDT are in good agreement both in regards of forces/moments and of rotor speed.

6. PERFORMANCE EVALUATION

The performance evaluation serves two main purposes; verification of the models in the performance database, and optimization of the ARMADA design.

Two validation scenarios are proposed in order to have a certain degree of confidence in the results obtained from the optimization. The first scenario is based on previous work, namely the experiments performed by the NASA ([RD.2], [RD.3], [RD.4] and [RD.5]). The second scenario is based on the experiments that are carried out during the ARMADA project in order to test the deployment system. This verification allows not only validating the models but also providing an idea of the usefulness of the IPDT when working with a system that was not the baseline of the project, such as the NASA autorotation vehicle.

The results obtained by the performance database are in good agreement with both the results of the NASA study, and with the windtunnel test results. One of the most important aspects to be well validated is the fact that the rotor can operate in two modes when the flow is subsonic. These are the stalled and unstalled operation modes. In the case of a stalled rotor, the rotor tip speed ratio is relatively low, namely around 1. In unstalled operation the rotor can reach a tip speed ratio of about 5. It should be noted that all studies, including this one, found that the rotor in supersonic flow displays only one, unique rotor velocity. This implies that a transition occurs at a certain Mach number.

Figure 6-1 and Figure 6-2 show the simulated descent-phase trajectory of a small lander equipped with an autorotation landing system, with characteristics broadly similar to the ones reported Table 3-2 and Table 3-3. The first sharp decrease in velocity in Figure 6-1 is associated with the reefing of the rotor, which starts at about 220 seconds, and ends about 10 seconds later.

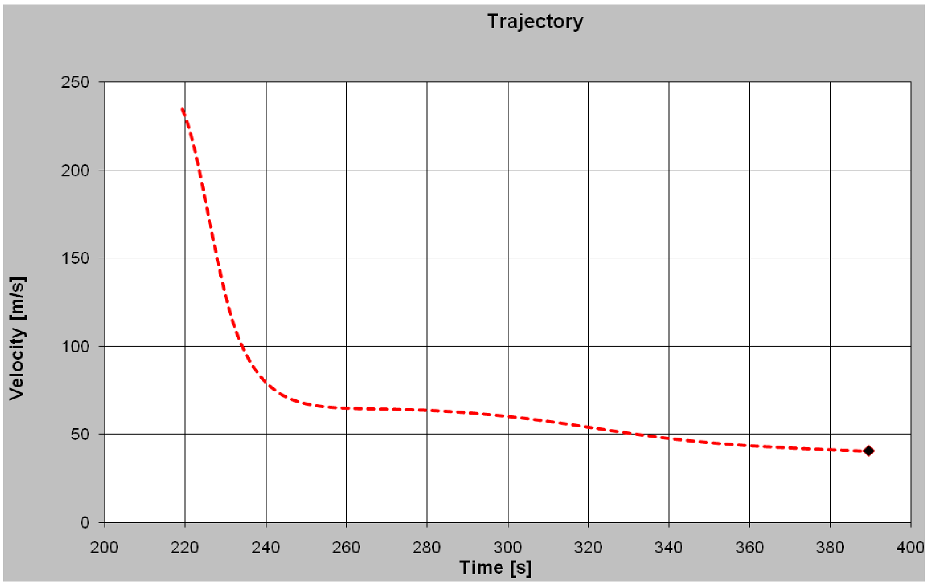

Figure 6-1: Velocity versus time during the descent phase

Figure 6-2 clearly shows the transition between stalled operation and non-stalled operation of the rotor. During the landing phase, the velocity actually decreases even further, to about 32 m/s at touchdown.

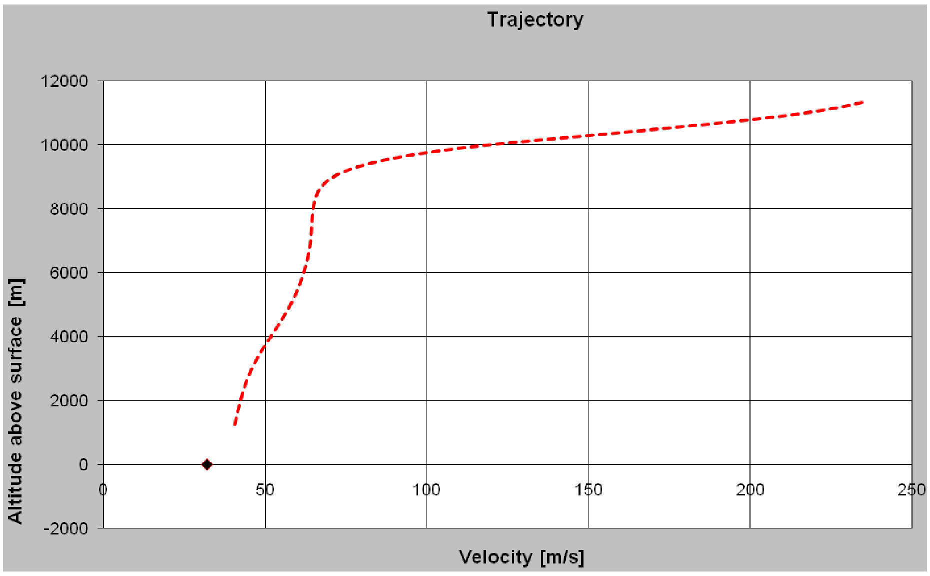

Figure 6-2: ARMADA trajectory during the descent phase

These results provide an indication that the behaviour of the rotor, and the resulting trajectory of the lander are strongly dependent on the collective pitch setting of the rotor blades. Figure 6-3 shows the evolution of the tip speed ratio as a function of time, with the collective pitch setting as a parameter. The collective pitch varies between about -5° and -13°. The collective pitch settings that have the lowest (absolute) value lead to a stalled rotor. However, after an unstalled flow has established itself, the lowest collective pitch setting for an unstalled rotor leads to the highest tip speed ratio, and the most efficient behaviour. This indicates that the collective pitch control should be designed with great care.

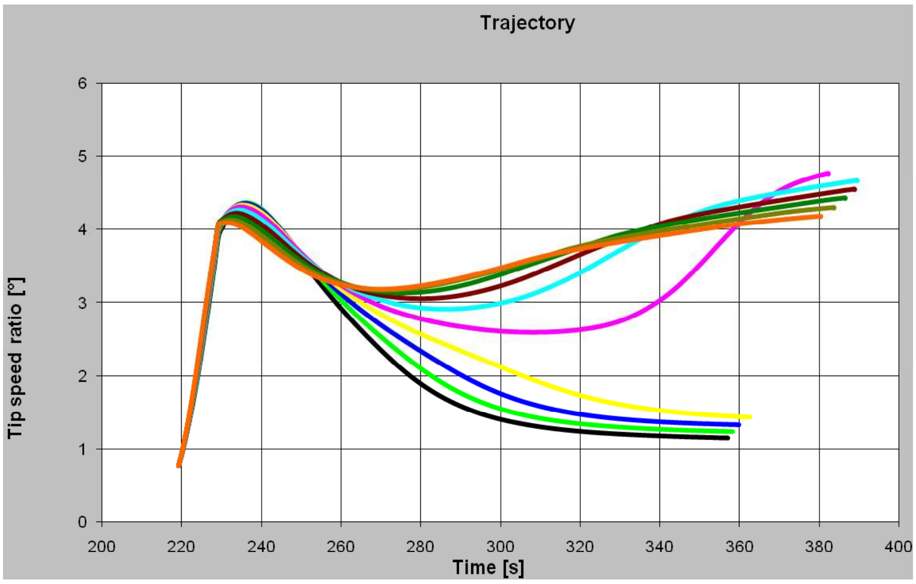

Figure 6-3: Tip speed ratio during the descent phase[1]

Another feature in the evolution of the tip speed ratio is the peak that occurs immediately after reefing. This peak is a direct result of the rotor reefing. Blade extension leads to a rapid deceleration of the vehicle, leading to an ever-lower free-stream velocity. At the same time, the rotor is slightly over-speeding when compared to steady-state conditions. This effect leads to an increase in the tip speed ratio. At the same time, the rotor radius

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[1] The collective pitch angle setting is used as a parameter, ranging from -5º (black graph) to –13º (orange graph) increasing with an equal spacing

is increasing while the angular momentum remains (approximately) constant. All else being equal, this would lead to a decrease in the dimensionless tip speed velocity. Although these two effects oppose each other, the effect of the rapid deceleration wins out and leads to a peak in the tip speed ratio.

Figure 6-4 shows the effect of the stall behaviour of the rotor on the descent velocity. Particularly noticeable is the trajectory of the lander with a collective pitch setting close to the stalled operation. After recovering from the near-stall behaviour, this configuration eventually leads to the lowest vertical velocity. These results indicate that a precise pitch control will be required for the rotor.

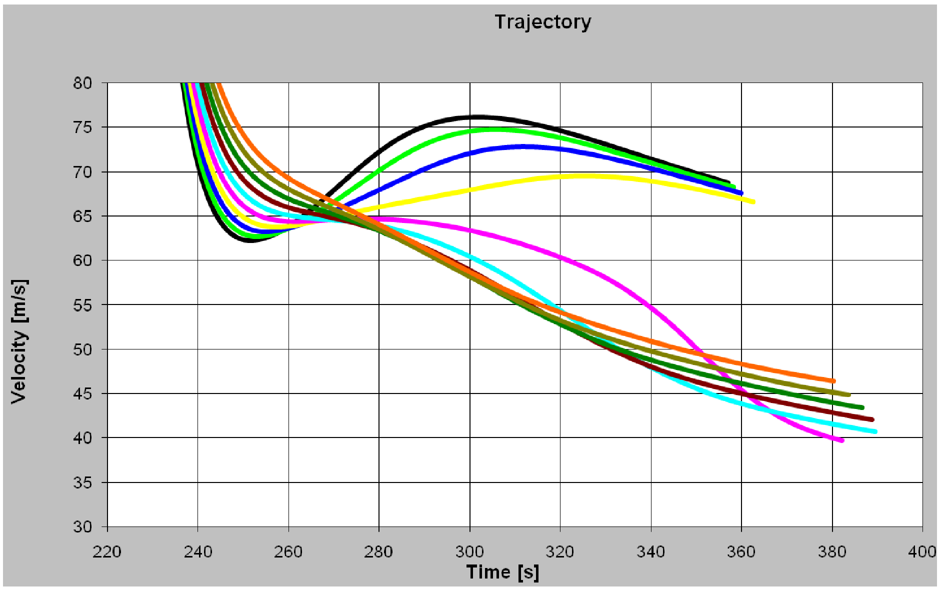

Figure 6-4: Vertical velocity during the descent phase[2]

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[2] The collective pitch angle setting is used as a parameter, ranging from -5º (black graph) to –13º (orange graph) increasing with an equal spacing

7. CONCLUSION

The confidence in the results obtained during the design of the ARMADA system is considered quite high, due to the very different methods used to obtain the results (that is, for example, the equilibrium descent velocity, the rotor drag coefficient, etc.). Some limitations have been exposed that are mainly due to the mechanics of the concepts and the maturity of the required technologies. These problems can be summed up as follows:

- High mass fraction of the rotor,
- High mechanical complexity leading to reliability problems,
- Too high equilibrium descent velocity,
- Potential problems with vibration, and mechanical loads.

As mentioned, most of the problems are of a mechanical nature, and should be solvable in the future. Candidate solutions can be found in various fields.

- Improvements in structures and materials could lead to lighter rotors, while control mechanisms integrated into the blade (possibly combined with 'smart' materials) could substantially reduce the mass of the control systems.
- Improvements in materials may also mitigate the problem with vibration and mechanical loads, for example by means of active damping, and through mass-reduction.
- The problem with the descent velocity can be resolved by relaxing the constraints on the system: although the aim of the system has been to replace all components of the traditional EDL systems, some additional braking systems could still be considered: tip rockets, airbags, or retrorockets. The optimal mix of these landing system ingredients could be investigated in future studies.

In the near-term the problems outweigh the advantages presented by the ARMADA system, namely early trajectory control and high cross-range capability. The situation changes for applications at Titan and Venus; in this case, the required size of the rotor is quite small, so rotor blades can consist of a single piece, and the rotor can be constructed using current technology (made space-qualified).

Compared with traditional EDL systems, the performance of ARMADA is not good enough to justify incorporating such a system in the short term. However, in the mid-term the concept may be a feasible option if an effective flare manoeuvre or the use of tip rockets is taken into account. It must also be remarked that in the mid-term scenario the value of the velocity at about 1-2 km AGL is comparable to the velocity of traditional EDL systems before the retrorockets ignition. The results presented here do not include either a flare manoeuvre or tip rockets:

- A flare manoeuvre, which is inherent to rotary systems, that converts 20 % of the energy stored in the rotor into a change in the velocity would allow an equilibrium descent velocity of up to 35 m/s and reduce the velocity to 20 m/s. This manoeuvre would work better for proportionally heavier rotors, meaning that they would work best for heavy landers.
- Tip rockets could potentially allow the descent velocity to be reduced to 0 m/s. Moreover, tip rockets add mass to the tips, increasing the stored kinetic energy and also increasing the potential effectiveness of a flare.

Neither solution was sufficiently verified due to the limited amount of time available within the present study; both deserve more attention in the future. Further studies should be focused in this direction.

Summarizing, in the mid- to long-term, and for application on Earth, Venus or Titan in the near-term, the concept may be quite an interesting and promising alternative to traditional EDL systems.

7.1. TECHNICAL BACKGROUND

- The equilibrium descent velocity that can be attained with the type of design put forward is about 30-40 m/s. This limit can hardly be improved by increasing the rotor size. The number of telescoping section is limited to four for the near-term scenarios, and five for the mid-term scenarios. Furthermore, the mass of the rotor itself becomes an important contributor to the overall mass of the vehicle, leading to a vicious cycle of increasing rotor size and mass. These two factors put a lower limit on the equilibrium descent velocity that can be attained.
- The nature of the scaling laws governing the mass of the rotor makes that larger landers require a greater fraction of their mass to be allocated to the rotor. This makes larger landers somewhat less effective than smaller landers. This is, in effect, a vicious cycle in the sense that a larger rotor would be required to accommodate the larger rotor mass.

- The rotor blades and the rotor hub (plus control mechanisms) form a very significant fraction of the total lander mass. In addition, the full mass of the rotor system remains attached to the lander, in contrast to traditional systems, where some mass gets ejected (parachutes) or burnt (propellant).
- The rotor speed during the deployment becomes quite high, leading to high centrifugal loads on the rotor hub. This would lead to an especially heavy rotor hub plus associated reefing and control mechanisms. In order to avoid this, operational constraints can be put in place. For example, by setting the collective pitch such that the rotor operates inefficiently, or by reefing earlier. Special attention needs to be paid to the collective pitch and reefing control mechanisms, such that overspeeding does not occur. Fine pitch control would then be required during the deployment phase.
- In addition, vibration may become a problem at very high rotor speeds. This was partly demonstrated in the windtunnel tests. It should also be kept in mind that current helicopters are designed to operate their rotors at a certain angular speed that is more or less kept fixed. ARMADA will span a large range of operating speeds, depending on the extension of the rotor. Care must be taken not to enter into a resonance.
- Special attention needs to be paid to rotor stall, because it affects the performance drastically. The stalled operating conditions of the rotor, as identified by NASA and reproduced in the verification scenarios can actually occur in the PD simulations as well. An unfortunate setting of the collective pitch angle during the transition from the supersonic to the subsonic regime causes this stalled operating condition. Fine pitch angle control, or in general a good rotor control is required to simultaneously optimize rotor performance and avoid rotor stall. The difference between stalled and unstalled operation is visible in both the rotor drag coefficient and the descent velocity; unstalled operation leads to a rotor drag coefficient of about 1 – 1,25 and a descent velocity of 31 m/s, while stalled operation leads to a rotor drag coefficient of about 0,25, and a descent velocity of about 80 m/s.
- Autorotation landers with telescopic blades, using no other means for decelerating than autorotation alone, cannot be recommended for use on Mars in the near-term. Apart from the lower limit on the equilibrium descent velocity of between 30 and 40 m/s as a consequence of the impracticality of increasing the number of sections to over 4-5, and the increasing mass of the rotor with increasing size, a number of other considerations speak out against autorotative landers: mechanical complexity, reliability, limited lander total mass (the upper limit is comparable to the upper limit achievable with traditional systems), high levels of autonomy required, all seem to pose significant obstacles to the particular design. These disadvantages outweigh the major advantages of the control capability very early in the EDL sequence – namely, immediately after deploying the rotor, and the capability to generate lift – leading to a significant cross-range capability.

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