LUNAR LANDING SYSTEMS STUDY

esa Contract 9558/91/NL/JG/(W0 N°24)

EXECUTIVE SUMMARY

Ref. LULA/MMSF/TN /048/96

MATRA MARCONI SPACE

August 1997

MATRA MARCONI SPACE

ESA STUDY CONTRACT REPORT			
ESA Contract N° 9558/91/NL/JG /(W0 N° 24)	**LUNAR LANDING SYSTEMS STUDY**		NAME OF CONTRACTOR **MMS - F**
ESA CR ( ) N°	STAR CODE	N° of Volumes 1 This is Volume n° 1/1	CONTRACTOR'S REFERENCE LULA/MMSF/TN/048.96

The *Lunar Landing Systems* Study has been performed by MATRA MARCONI SPACE France prime (Toulouse centre) and MATRA MARCONI SPACE Space Systems (Stevenage centre) with the support from GMV (Spain) acting as consultant. This study lies within the framework of the Lunar European Demonstration Approach (LEDA).

The main objectives of the *Lunar Landing Systems* study was to define and analyse GNC requirements and technologies permitting to achieve a safe and accurate landing near the South pole of the Moon, and to evaluate the impact of these requirements on the preliminary definition of the mission scenario and vehicle design previously achieved within the ESA *LEDA Assessment* study and *Lunar Lander Technology* study.

Due to the high criticity of the descent phase, from the de-boost manoeuvre on the preparatory lunar orbit up to the touch down, most of the GNC analyses were focussed on this phase. Risk assessment was performed at system level and included all the mission phases.

The first step of the GNC analyses was to define GNC constraints applying to the descent trajectory and to refine its definition under optimisation of fuel consumption. Then, GNC analyses were carried out for each subphases of the descent phase: de-orbit boost, coast phase, Inertial Guidance Phase, Homing Phase, Approach Phase, Final Descent, Touch Down. The key design drivers were pointed out, and a set of requirements applying to the GNC hardware, propulsion system and vision system were worked out. Verification of the GNC performance and landing accuracy was verified by covariance analyses and Monte Carlo time simulation using respectively CAPTAINS-X and 2D-LANDSIM software tools. Criticity of these requirements were assessed at programmatics level.

An iteration was then performed on the current design of the Reaction Control System and the Landing Gear system, taking into account the obtained GNC requirements and performance. The overall system was then optimised so as to increase as much as possible the payload mass.

Names of authors : B. FRAPARD, C. CHAMPETIER, S. KEMBLE, R. PARKINSON

NAME OF ESA STUDY MANAGER	ESA BUDGET HEADING:	
Martin LANG DIV.: Propulsion & Aerothermodynamics DIRECTORATE: Technical & Operational Support	N5101/060.512/95N10	

TABLE OF CONTENTS

PAGE INTENTIONALLY LEFT BLANK

REFERENCE DOCUMENTS

[1] LEDA Assessment Study, Final Report
ESA/ESTEC Report LEDA-RP-95-02, June 1995

[2] LEDA Mission Analysis
MAS Working Paper No. 376, Jan. 96

[3] Lunar Lander Technology
SENER Final Report

[4] GNC System Design Synthesis
MMS Technical Note LULA/MMSF/TN/022/96, October 1996

[5] Optimal Trajectories for Lunar Landing
MMS Technical Note TN/LLS/STV/1-2, November 1996

[6] Ground Tracking Orbit Determination for Lunar Lander
GMV Technical Note, GMVSA 2067/96, April 1996

[7] GNC Analyses from the De-Boost to the Visual Guidance Phase Entry Point
MMS Technical Note LULA/MMSF/TN/027/96, July 1996

[8] Design and Performances of Trajectory Control in Approach Phase
MMS Technical Note LULA/MMSF/TN/023/96, July 1996

[9] Descent Trajectories Description for use in GNC Analyses
MMS Technical Note LULA/MMSF/TN/031/96, July 1996

[10] Relative Navigation Concepts for the Approach Phase
MMS Technical Note LULA/MMSF/TN/033/96, Aug. 96

[11] Guidance and Control Analyses for the Final Descent
MMS Technical Note LULA/MMSF/TN/041/96, Oct. 96

[12] Risk Assessment Report
MMS Technical Note TN/LLS/STV/3-0, April 1996

[13] Reaction Control System Integration
MMS Technical Note TN/LLS/STV/3-1, November 1996

[14] Touch Down Analysis
MMS Technical Note TN/LLS/STV/3-2, November 1996

[15] System Optimisation
MMS Technical Note TN/LLS/STV/3-3, November 1996

[16] LEDA 3.0 - System Data Sheet
Version 3.0, Dec. 95, Excel Format

PAGE INTENTIONALLY LEFT BLANK

ACRONYMS

CCD .... Charge Coupled Device
DOB .... De-Orbit Boost
FOG .... Fiber Optical Gyro
FOV .... Field of View
G/C .... Guidance/Control
GNC .... Guidance, Navigation, Control
GT .... Ground Tracking
GTO .... Geostationary Transfer Orbit
HG .... High Gate
HGA .... High Gain Antenna
HRG .... Hemispheric Resonant Gyro
IGP .... Inertial Guidance Phase
IMU .... Inertial Measurement Unit
INS .... Inertial Navigation System
LEDA .... Lunar European Demonstration Approach
LG .... Low Gate
LGF .... Lunar Gravity Field
LGM .... Lunar Gravity Model
LOS .... Line of Sight
LM .... Landmark
LS .... Landing Site
MECO .... Main Engine Cut-off
MIB .... Minimum Impulse Bit
PDI .... Powered Descent Initiation
RCS .... Reaction Control System
RLG .... Ring Laser Gyro
S/C .... Spacecraft
ST .... Star Tracker
TBC .... To Be Confirmed
TOF .... Trajectory Optimisation Facility
VGP .... Visual Guidance Phase
VGPEP .... Visual Guidance Phase Entry Point
WFOV .... Wield FOV
WP .... Work Package

PAGE INTENTIONALLY LEFT BLANK

1. INTRODUCTION

This note has been produced in the framework of the « *Lunar Landing Systems* » study, WP 600, ESA Contract 9558/NL/JG/91, Work Order N° 24. It constitutes the Final Report of the work performed within this study.

The development of soft landing capacities is one of the key features for the success of future Moon missions. The Lunar Landing system study was initiated by ESA to analyse in depth the GNC design of an autonomous lander aiming at a safe and accurate landing in the chaotic region of the South Pole. This study follows the ESA Assessment study (ref. 1-2) and the « *Lunar Landing Technology* » study (ref. 3) in the framework of which a preliminary design of the lunar landing mission, landing trajectories and vehicle were performed.

This note presents the results of the GNC analyses carried out during the « *Lunar Landing Systems* » study, the consolidated GNC system requirements issued from these analyses, and the subsequent lander system optimisation. The work focuses on the most critical phase of the mission, the descent from de-orbit boost to landing.

The Executive Summary architecture is as described in the following table. References to the technical notes containing the detailed presentation of the work performed within the study are given.

N°	Sections	Contents	Technical Notes
2	System Assumptions	Review of vehicle design & Descent Scenario	ref. 4
3	Trajectory Design	Detailed trajectory design including GNC constraints	ref. 5
4	GNC Analyses & Performance Assessment	Phase by phase GNC analyses leading to specification of the GNC equipment, camera system and propulsion system	ref. 4, 6, 7, 8, 9, 10, 11
5	GNC Hardware Synthesis	Synthesis of the targetted GNC hardware and system budgets	ref. 4
6	System Optimisation	Impact of GNC requirements at system level	refs. 13, 14, 15

The final report of the study is composed of this Executive Summary, together with the synthesis technical notes attached in annex:

Annex A: GNC System Design Synthesis (ref. 4)

Annex B: System Optimisation (ref. 15)

2. SYSTEM ASSUMPTIONS

2.1 The LEDA Mission

A reference mission scenario for a shared Ariane 5 launch has been assembled on the basis of the results from the ESA LEDA Assessment Study for a mission lift-off on 2002-11-01 at 23:05:00. In this scenario, the vehicle makes a direct transfer from GTO to the Moon, initially entering a 200km altitude polar parking orbit, and then descending to 50 km for site reconnaissance before final descent (see Figure 1).

A baseline landing site has been selected near the South pole (84.5°S latitude, 0°W-10°W longitude) on the basis of scientific interest, Earth-Sun visibility constraints, safety at landing and rover constraints. This landing site has the features of highland regions of the Moon, in particular a very rough surface is expected at landing. Due to that, autonomous detection of obstacles in the final part of the descent trajectory is mandatory together with a landing accuracy of typically some meters.

2.2 Vehicle Description

The overall Lunar Lander Vehicle design (see Figure 2) is that defined in the « *Lunar Lander Technology* » study (ref. 3). It has four 0.95 m diameter propellant tanks and eight LEROS 2B engines of 431 N each, arranged around the 1.67 m core structural cylinder, with the payload installed on the upper face of the cylinder. The all up mass amounts to 2890 kg, including 2036 kg of propellant and a payload mass of 200 kg. During descent, the lander is submitted to large variations of mass and inertia characteristics, from 1630 kg down to 893 kg, from [1752 , 1368, 3370] kgm2 down to [621, 621, 1000] kgm2. Control torques are provided by a set of 12 LEROS 20 engines, of 22N each. Pitch and yaw thrusters are all aligned in the direction of the main engines (« downward »), so that they can contribute to the main thrust.

Obstacle detection, landing site retargetting, accurate relative navigation are achieved by means of a mono-vision system using a fixed wide field-of-view camera and advanced image processing coupled to an Inertial Navigation System.

The landing gear is constituted of four legs with dampers designed to avoid rebound and tip over at touch down despites residual velocities and angular rates, local surface slope and possible obstacles.

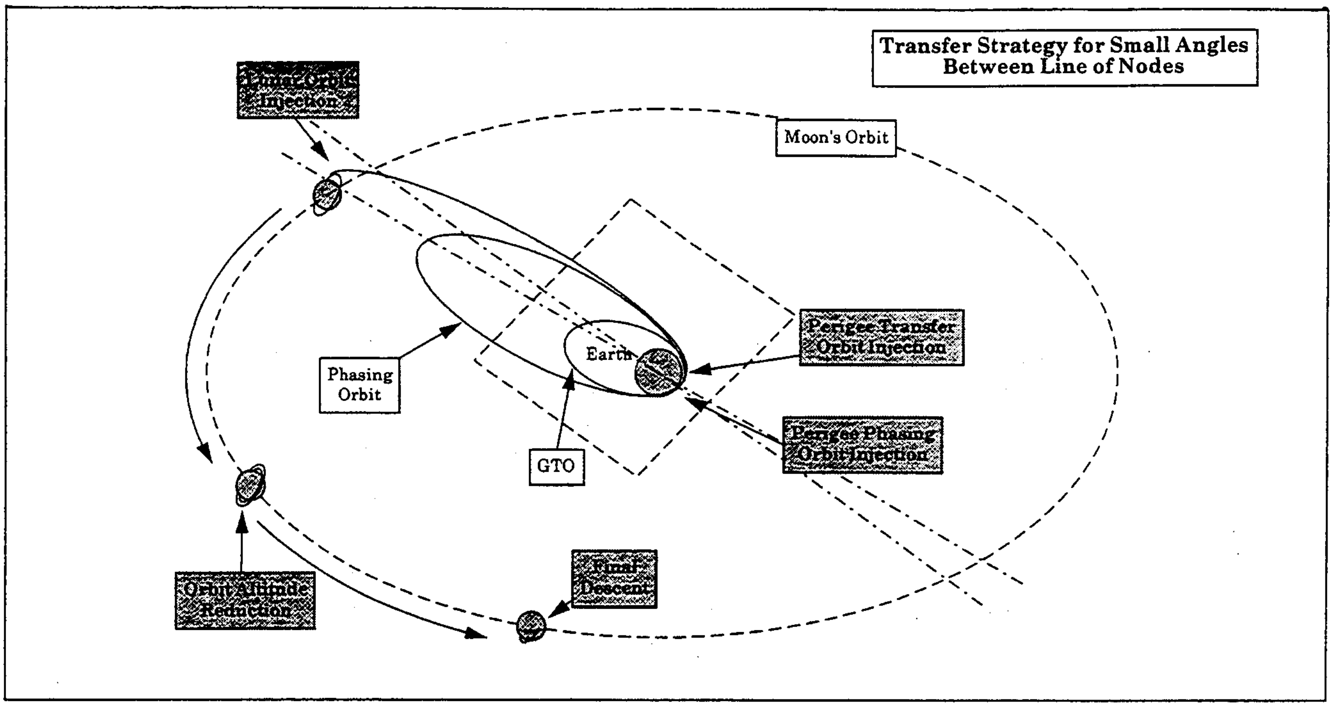


Figure 1: Overall LEDA Mission Scenario

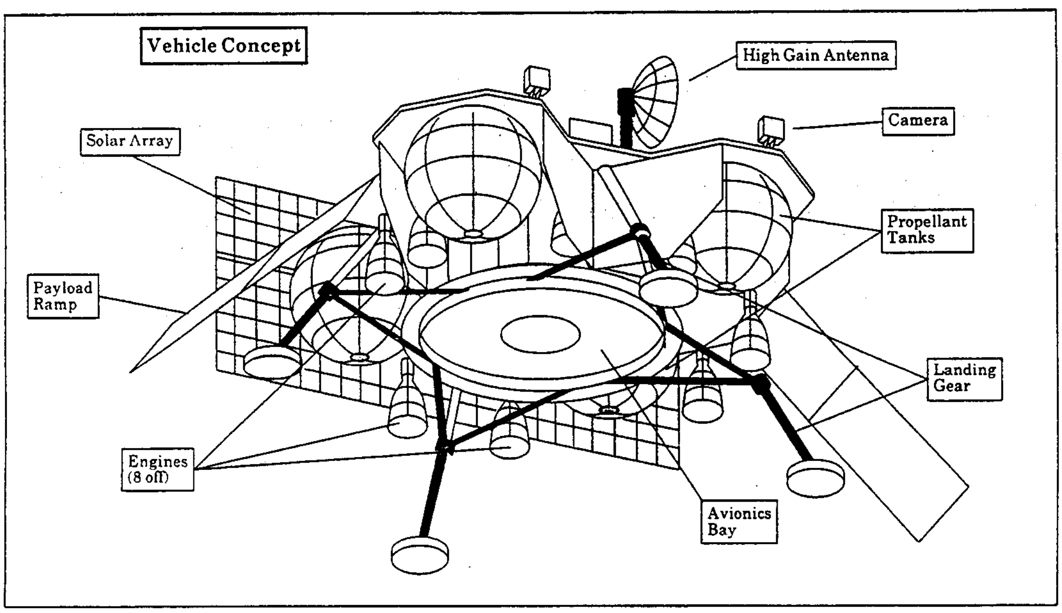


Figure 2: Spacecraft Design for LEDA

2.3 Baseline Descent Scenario

The descent phase drives in a large extent the GNC system design. The baseline descent scenario is that defined in Ref. 1 & 2 and refined in Ref. 4. The main phases of the descent are the following (cf Figure 3):

Descent Preparation Orbit. The spacecraft is placed on a low altitude polar orbit for landing area mapping. The localisation of the spacecraft in a Moon reference frame is performed by ground tracking. An optimal descent trajectory is computed by the ground segment and transmitted to the onboard computer prior to the descent.

De-Orbit Boost (DOB). A small manoeuvre injects the vehicle on an elliptic transfer orbit. A **Coast Phase** results up to the Power Descent Initiation (PDI) occuring near the perilune of this orbit.

Inertial Guidance Phase (IGP). The spacecraft follows a prescribed thrust profile for an optimum constant breaking towards the selected landing site. An Inertial Navigation System (INS) is used during this phase up to landing.

Homing Phase. It corresponds to the first acquisition of images of the lunar surface from which the position of the vehicle relatively to an onboard map of the lunar surface will be updated.

High Gate (HG). At this point, the landing site comes into visibility of the lander vision system.

Approach Phase. Along the nominal trajectory, the landing site is kept into visibility. Obstacles may be detected and retargetting (i.e. change in landing site) is implemented if necessary. During this phase, the actual position/velocity of the vehicle relatively to the selected landing site is estimated by the onboard relative navigation function and the vehicle is steered on a trajectory aiming at this point.

Low Gate (LG). At this point, the landing site visibility conditions are no longer satisfied. The vision system is no longer operable and a « blind » landing follows up to the Main Engine Cut Off (MECO) point.

Touch Down. A free fall follows the MECO up to the contact with the lunar surface. Dynamics conditions at touch down shall be compatible with the landing gear design so that a safe landing occurs.

The homing and approach phases constitute what is called the **Visual Guidance Phase** (VGP).

3. TRAJECTORY DESIGN

3.1 Optimisation Principle

Optimal descent trajectories compatible with the proposed scenario has been designed in the course of the study using the MMS Trajectory Optimisation Facility (TOF). The main objective of this design was to minimise the fuel consumption while keeping the visibility to the landing site as long as possible during the approach phase in order to achieve the critical GNC tasks. Constraints to be satisfied are related to the manoeuvrability of the vehicle (available thrust, available attitude control torque), the visibility of the landing site, the overlap between two successive images of the lunar surface for optimal image processing. Design parameters are basically the altitude of the PDI point and the location of High Gate and Low Gate. The optimisation parameters are the variables defining the pitch profile. Optimisation has been performed phase by phase by Multiple Shooting.

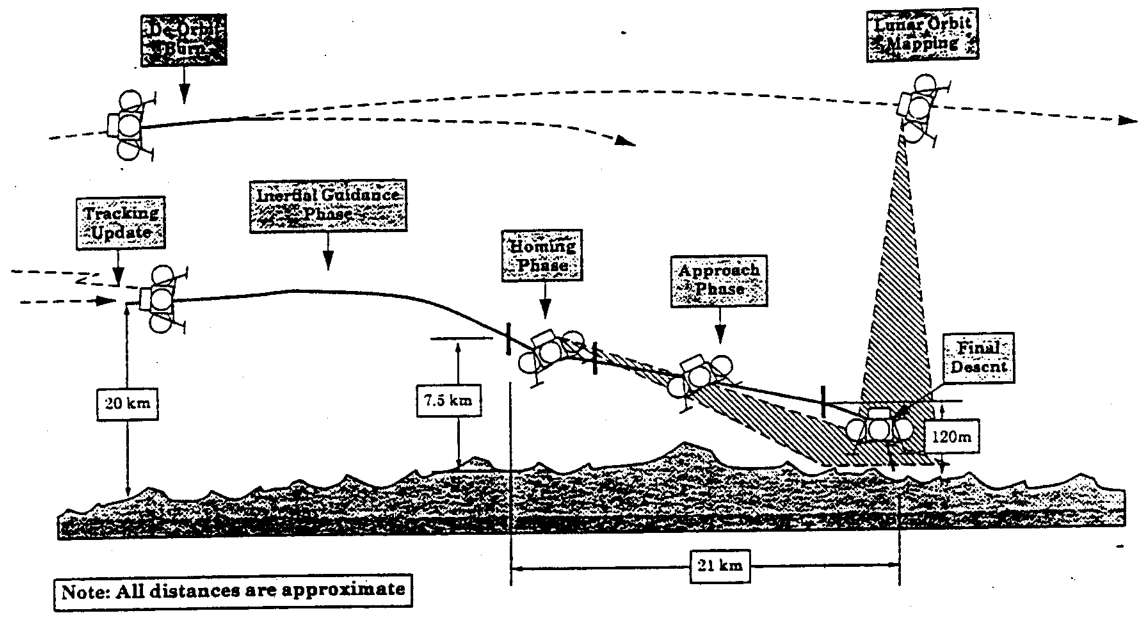


Figure 3: Descent Scenario

3.2 Trajectory Design

Very low PDI altitude is favourable for the fuel consumption: at the limit, a pure Hohmann transfer is the ultimate optimum which can be achieved. However, limitation occurs due to the available finite thrust. PDI altitude above 10 km are achievable with the available main thrust. Sensitivity of the fuel consumption to the PDI altitude has been found negligible from 10km altitude to 20km altitude. The maximum 20 km altitude has been selected so as to optimise the approach phase duration.

In order to simplify the trajectory design, a quasi constant trajectory slope has been selected for the approach phase. Here again, very low slopes are preferable to minimise the gravity loss, while steep slopes are better for the vision system. Sensitivity of both fuel consumption and approach phase duration to the approach trajectory slope has been studied. A gain of about 30m/s and 30% of the Approach Phase duration has been found when decreasing the slope from 40° to 20° (see Figure 4). Finally, a 20° slope has been selected as baseline, leading to the trajectory characteristics presented Table 4.

The approach trajectory has been calculated with a nominal thrust of 3920N (corresponding to an increase in the nominal thrust of the baseline LEROS 2B engines from 431N up to 490N), while the final descent has been computed with a nominal thrust of 3448N.

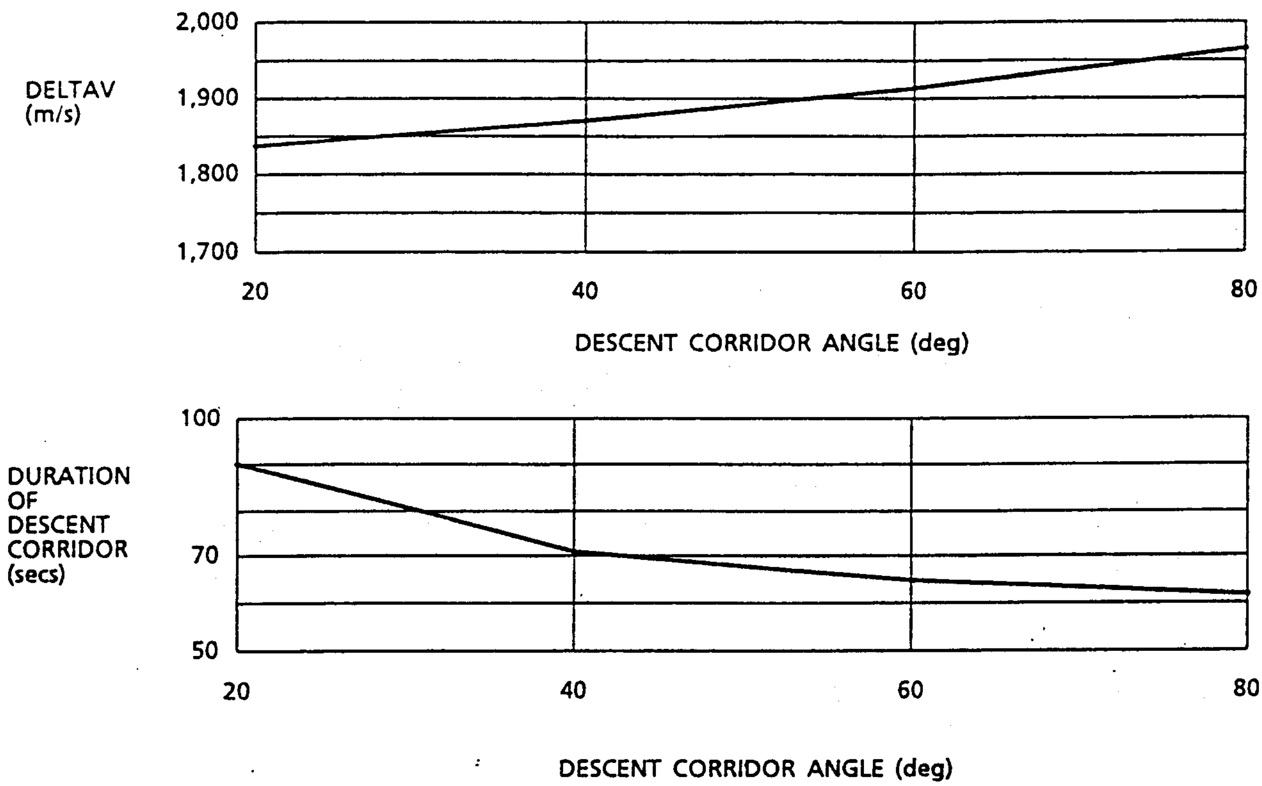


Figure 4: Sensitivity to Trajectory Design Parameters

Trajectory point	time (secs)	position	velocity
prior DOB	0	range : N.A. altitude : 50 km	1655.48 m/s
after DOB	3.19	range : N.A. altitude : 50 km	1648.34 m/s
PDI	2947.83	range : 226.3 km altitude : 20 km	1676.46 m/s
High Gate	3459.96	range : 12.7 km altitude : 5.0 km	287.76 m/s
Low Gate	3540.39	range : 345.9 m altitude : 160 m	46.8 m/s
MECO	3557.9		1 m/s

Trajectory phase	duration (secs)
DOB	3.19
Coast phase	2944.64
IGP	512.3
VGP	80.4.
Final descent	17.5

Table 4: Baseline Trajectory Characteristics

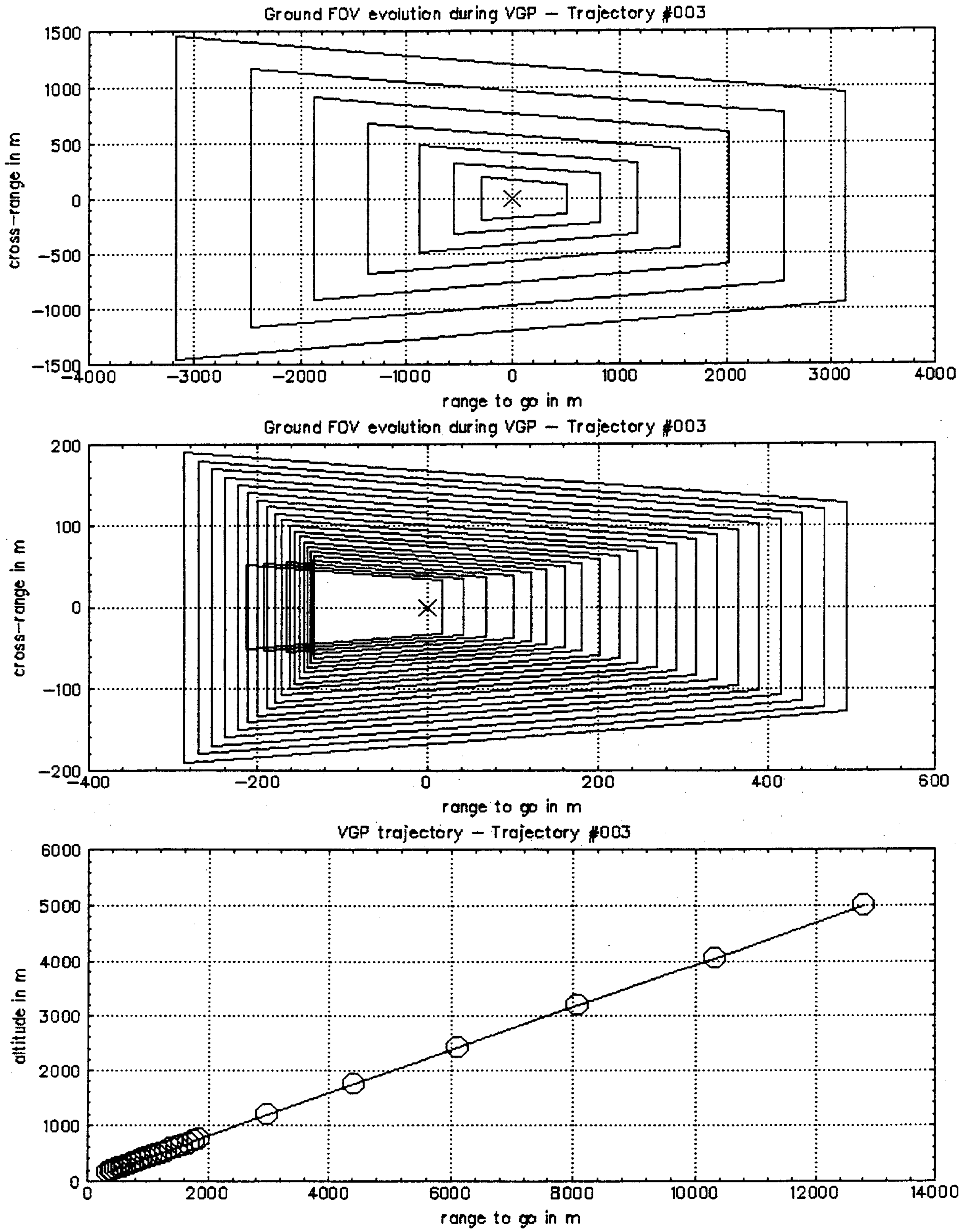


Figure 5 - Landing Site Visibility along the Baseline Trajectory

Plot 1: Traces every 10 secs - Plot 2: Traces every 1 sec - Plot 3: Reference Trajectory

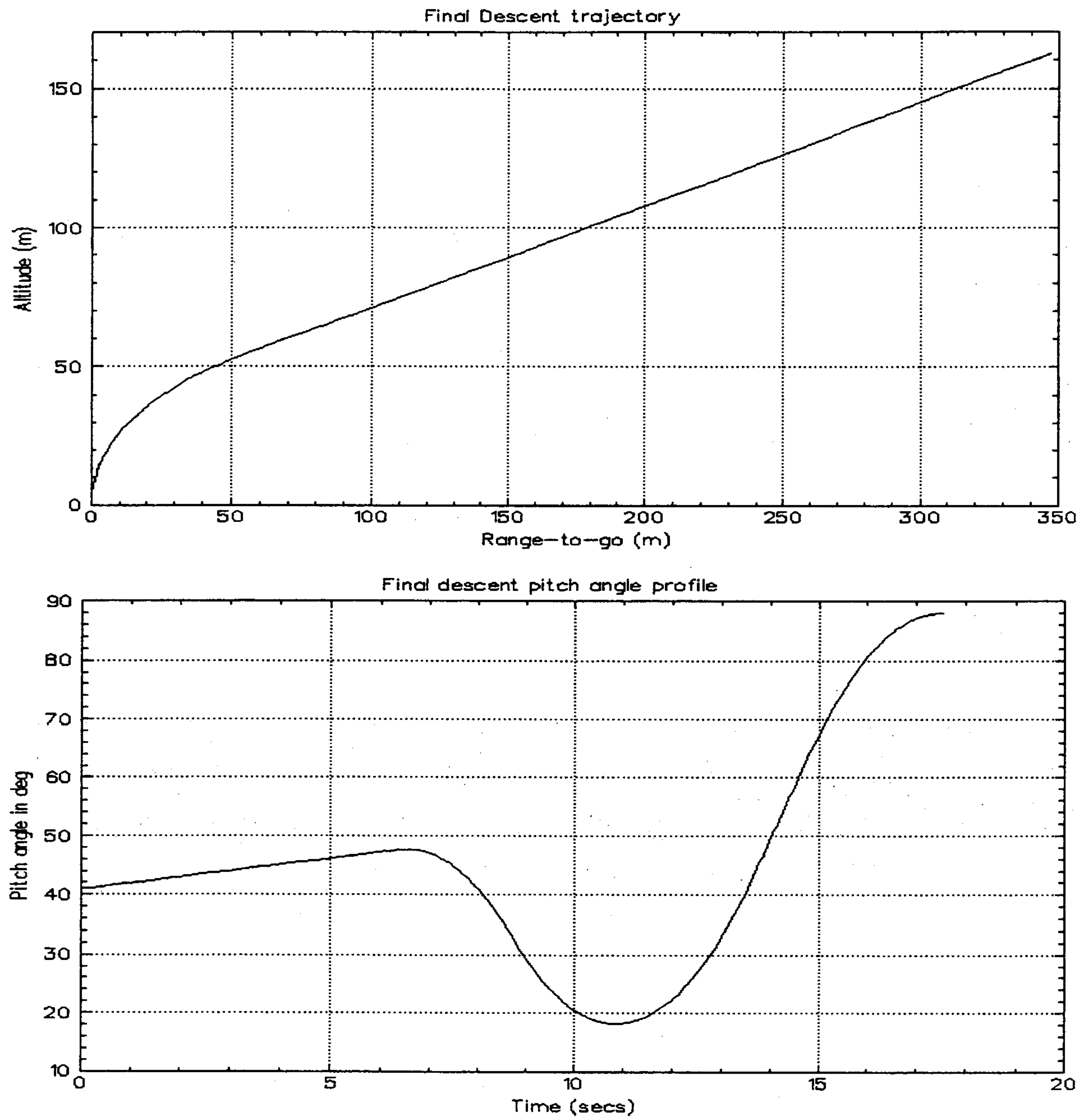


Figure 6: Final Descent Trajectory

4. GNC ANALYSES & PERFORMANCE ASSESSMENT

4.1 De-Orbit Boost and Coast Phase

Ground tracking performance analysis has been carried out on the basis of the descent preparatory scenario presented in Section 1. The 70th order Lunar Gravity Model (LGM) determined by the Goddard Space Centre after the Clementine mission has been used. The characteristics of the Ground Tracking System used in this analysis are given in Table 7. Without Ground Tracking, the localisation errors grow up to (1520m, 360m, 530m) 1-sigma at PDI in the along track, cross track and vertical directions, essentially due to the uncertainties on the LGM (even when the full order model is used). Localisation errors can be maintained at the level of 84m and 7.4cm/s 1-sigma when Ground Tracking is used during the coast segment (full Earth visibility is assumed). This hypothesis is taken as baseline in the subsequent GNC analyses.

Even in the ideal case of no localisation errors at DOB and no DOB realisation errors, actual dispersions with respect to the nominal trajectory are still too important at PDI: (1370m, 323m, 442m) 1-sigma, entirely due to the LGM uncertainties. Different approaches can be implemented to cancel these dispersions :

- trajectory control during the coast and IGP phases;
- update of the PDI timing by the ground segment to cancel the along track dispersions (the most important);
- update of the optimal descent trajectory by the ground segment and uploading to the onboard GNC system prior to PDI ;
- autonomous update of the optimal descent trajectory by the onboard GNC system.

Analyses of trajectory control capability during IGP have shown that all the scenarii can be implemented. In any cases, a radar system is of no help in this phase.

Selected Ground Stations	Perth, Vancouver, Madrid	
Measurement Characteristics	Range	Doppler
Frequency	1/60 Hz	1/60 Hz
Random error	10 meters	5 mm/sec
Bias error	3 meters	-
Station location error	1 m along X/Y axes (equator), 3 m along Z axis	
Minimum elevation for meast	5°	

Table 7: Ground Tracking Characteristics

4.2 Inertial Guidance Phase

IGP is a powered phase lying between the PDI and the first image acquisition used for navigation update. During this phase, Inertial Navigation is mandatory to propagate the knowledge of position and attitude of the vehicle. IGP is a dimensioning phase for the specification of the Inertial Navigation System. The performances of the Inertial Measurement Unit and external sensors used by the INS shall be sufficient to maintain the position and velocity dispersions at High Gate at a level compatible with the performances of the GNC, vision and propulsion subsystems during the Approach Phase. A typical 1000m (3 sigma) dispersion at High Gate has been assumed admissible at this stage of the study.

It is worth noting that these dispersions together with the uncertainty on the lunar surface elevation at High Gate (mainly due to dispersions on horizontal position) require to define a minimum High Gate altitude to avoid dramatic impact at this point. In the proposed baseline trajectory, the High Gate altitude is assumed to be at least 5 km which seems safe.

It is assumed that position knowledge is provided by the Ground Tracking System prior to PDI, that the attitude knowledge is updated at the end of the coast phase using external sensor(s) (this is possible up to some seconds before PDI, as no attitude constraints exist during this unpowered phase) and that no further attitude update is done during IGP (this is a worst case hypothesis since the near-horizontal thrust orientation during IGP and the star tracker implementation should be compatible with star tracking).

Parametric covariance analyses using the MMS software CAPTAINS-X have been carried out around the reference nominal trajectory to specify the required external sensor and IMU performances. The resulting dispersions at High Gate have been estimated in the following cases: no trajectory/thrust control, closed-loop trajectory control, closed-loop thrust control. In the second case, it has been assumed that the actual dispersions are essentially due to navigation errors. In the third case, thrust control errors have also been taken into account.

These analyses have shown that trajectory control or thrust control are mandatory during IGP. The key performance drivers are the external attitude sensor accuracy and misalignments and the gyro drift as concerns the cross track and vertical errors, the acceleros bias as concerns the along track errors. Weak sensitivity to LGM order reduction, gyros scale factors and acceleros scale factors have been found.

The derived external sensor and IMU specifications are provided on Tables 8. It results that a Wide Field-Of-View Star Tracker class external sensor and a medium class IMU are required. The performances at High Gate are shown on Table 9. In the thrust control option, the performances are given under the assumption that the attitude control residual biases are perfectly cancelled out.

IMU SPECIFICATIONS	(1 sigma Errors)
LGM order	4
Gyros Constant Drift	calibrated
Gyros Random Drift	0.1 °/hr over 600s
Gyros Scale Factor	1000 ppm
Gyros noise PSD	0.02 deg/sqrt(hr)
Acceleros Constant Bias	calibrated
Acceleros Random Bias	50 µg over 600s
Acceleros Scale Factor	300 ppm
Acceleros noise PSD	0.05 ms^{-1}/sqrt(hr)

Table 8a: IMU Specifications

SPECIFICATIONS	1 sigma
Z-axis (pointing axis)	0.1°
X/Y axes	0.02°
Misalignments ST/IMU	0.03°

Table 8b: External Attitude Sensor Specifications

DISPERSIONS at HIGH GATE (1 sigma)	No Control	Trajectory Control	Thrust Control
along track position	1025 m	156 m	201 m
cross track position	469 m	225 m	215 m
radial position	482 m	198 m	184 m
along track velocity	4.03 m/s	0.52 m/s	0.73 m/s
cross track velocity	1.94 m/s	1.03 m/s	1.00 m/s
radial velocity	2.07 m/s	0.70 m/s	0.66 m/s

Table 9: Inertial Navigation Performances at the end of IGP

4.3 The Approach Phase

4.3.1 Guidance & Control Analyses

Guidance and Control (G/C) analyses related to the Approach Phase have been conduced with the following objectives:

- elaborate the requirements on the G/C and propulsion system so that the dispersions at High Gate resulting from INS errors during IGP can be cancelled out at Low Gate;
- verify that the baseline reference approach trajectory is well adapted to these requirements;
- derive the G/C performances at Low Gate in terms of position and velocity dispersions, over-consumption, landing site visibility during the Approach, retargetting capabilities during the Approach;

Due to the fact that the actual dispersions during the Approach Phase (typically 1000m at High Gate down to some tens of meters at Low Gate) are expected to be much larger than the relative navigation errors after update by the vision system (typically from some tens of meters at High Gate down to one meter at Low Gate), the G/C analyses have been done under the assumption that the relative navigation is perfect.

In most of the G/C analyses performed here, the selected guidance strategy was to steer the vehicle along a fixed trajectory (the reference one), just translated in case of retargetting such that the terminal point becomes the desired one. In order to reduce the along track position errors, a re-synchronisation procedure has been set up: the initial point of the guidance trajectory is selected as the point of the reference trajectory the closest from the actual position of the vehicle.

Short term controllability of the trajectory is possible only if the thrust magnitude can be modulated (throttling capability). Controllability of the terminal point without throttling has been shown not robust, so throttling capability of the propulsion is assumed as baseline. The control actions then result from modification of the thrust magnitude by throttling and modification of the thrust direction through attitude control.

In all the analyses, simple proportional-derivate control schemes have been applied. It has been shown that gain scheduling is necessary to optimise the performances of the system under saturation coonstraints. Smooth variations of the settling time such that it is equal to the time-to-go before touch down is certainly the optimal way to perform gain scheduling. In our preliminary design, only two sets of control gains have been designed and implemented.

Only the motion in the vertical plane has been considered here. Indeed, the lateral motion is decoupled at the first order from the vertical one, and very simple adaptive guidance laws exist to efficiently cancel out the lateral dispersions. Furthermore, the landing site visibility is not so much affected by the lateral control since there exists a dgree-of-freedom around the thrust direction to control the azimuthal pointing of the camera towards the landing site if needed.

Due to nonlinearities existing in the system (thrust saturations), G/C analyses were performed by Monte-Carlo time simulations. All the simulations were performed with worst case random High Gate dispersions (corresponding to a worst case IMU selection): 1040m (3-sigma) down range dispersions, 728 m (3-sigma) vertical dispersions. Furthermore, it has been assumed that High Gate is acquired at 12 km down range, 80s time-to-go from Low Gate (actually, landing site visibility is available at a longer range). Under these hypotheses, the results of G/C analyses were the following (see Figure 10).

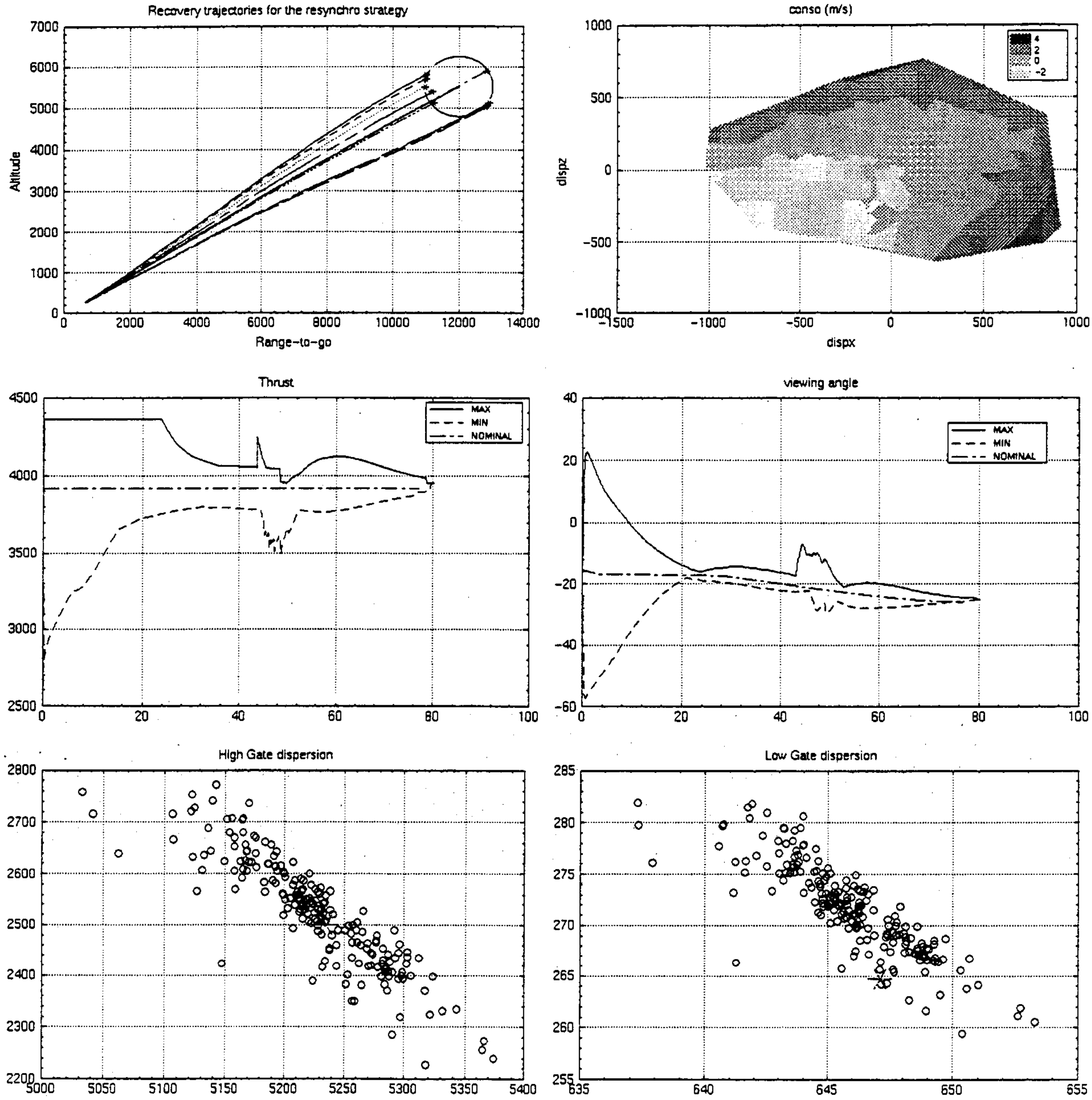


Figure 10: Result of G/C Monte Carlo Analysis

Recovery from Initial Dispersions at High Gate.

(Transient at about 50s from High Gate is due to change in control gains. This transient may disappear if a smooth gain scheduling is implemented).

A throttling capability of [0N, 4320N] (from 0% to 110% of the nominal thrust) has been shown just sufficient to reach good performances at Low Gate. These performances are given in Table 11. Full landing site visibility recovered after a 10s transient in the case where a [-20°,+20°] camera field-of-view is selected. The visibility loss lasts 20s if a [-10°, +10°] FOV is selected. These performances are robust to thrust realisation errors. The thrust error assumptions are given in Table 12. From these analyses, **a recommended maximum torque level is 100Nm.**

Table 13 presents the retargetting capability in terms of horizontal change of the landing site as function of the time-to-go up to Low Gate. Retargetting is assumed to be admissible if it induces a dispersions at Low Gate below 30m 3-sigma on each axis (this dispersion has been shown to be recoverable during the final descent path - see Section 4.4).

Dispersions at Low Gate	**(3 sigma)**
Down Range	7.8 m
Altitude	13.4 m
Down Range Velocity	0.7 m/s
Vertival Velocity	1.3 m/s
Max. Initial Transient Amplitude	36°
Max. Initial Transient Duration	20s
Max. Viewing Angle Variations	[-5°,+6°]
Over-Consumption Variations	[-2m/s,+4m/s]

Table 11: Dispersions at Low Gate (Perfect Navigation assumed)

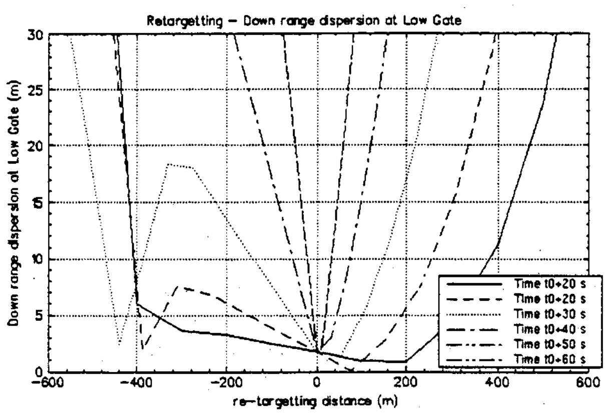
Thrust Realisation Errors	**(3 sigma)**
Attitude Control Bandwidth	1 Hz
Attitude Control Bias	1°
Attitude Control Noise	0.1°
Throttling Bandwidth	10 Hz
Throttling Error	1% scale factor

Table 12: Assumptions on Thrust Error

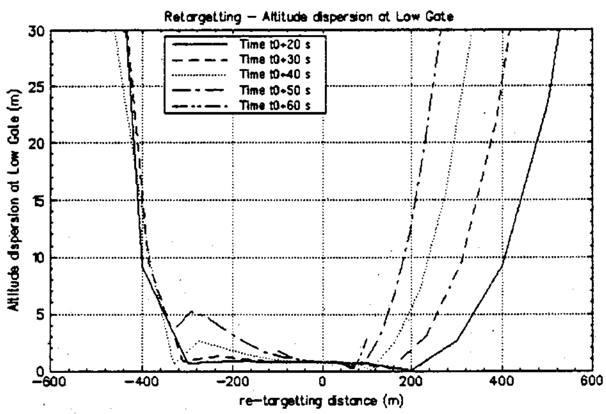
The retargetting capability decreases from typically 400m at 60s time-to-go down to 80m at 20s time-to-go from Low Gate. The retargetting capability has been assessed with the assumption that the vehicle is on its nominal trajectory. So, the presented retargetting capability is with respect to the « current » landing point instead of the targetted one.

Time-to-go from Low Gate	60 s	50 s	40 s	30 s	20 s
Max Allowed Re-targetting (rom)	-400m, +500m	-400m;+400m	-300m;+250m	-200m;+150m	-80m;+80m
Viewing angle departure from nominal	[-10°,+10°] after	[-10°,+10°] after	[-10°,+10°] after	[-5°,+5°] after	[-5°,+5°] after
Time to recover LS Visibility	20s	10s	10s	8s	8s

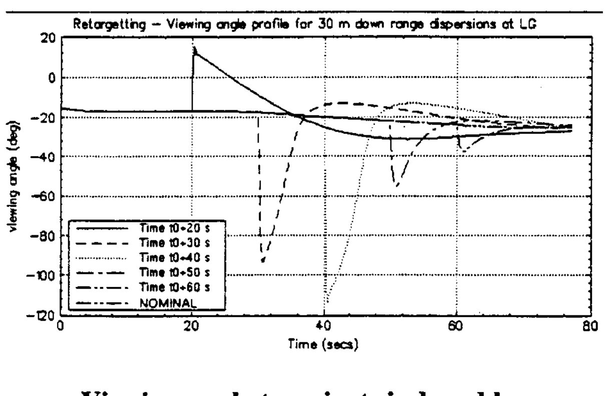
Table 13 - Retargetting Capabilities



Dispersions at Low Gate induced by Retargetting at different time after VGPEP



Dispersions at Low Gate induced by Retargetting at different time after VGPEP



Viewing angle transients induced by Retargetting at different time after VGPEP

4.3.2 Relative Navigation Analyses

Three concepts of relative navigation based on monovision have been investigated for application to the Approach Phase:

- LOS + range to one landmark (LOS = Line-Of-Sight)
- LOS to multiple landmarks
- LOS to one landmark

In the first concept, the LOS to landmark is directly measured by the camera in the instrument frame, and translated into the Moon surface reference frame by the Inertial Navigation System. The range is estimated through image processing thanks to the so-called zoom-effect (see Figure 14a).

In the second concept, the position of the vehicle is estimated by triangulation from the measurement of the LOS to at least two landmarks. The position estimation is « attitude-free » (i.e. does not need knowledge of the absolute attitude) if at least three landmarks are tracked (see Figure 14b). This localisation process needs the knowledge, or the estimation of the position of the landmarks relatively to a reference landmark.

In the third concept, the observability of relative position to the tracked landmark is gained through variations of the LOS to landmark along the descent trajectory (see Figure 14c).

The two first concepts are applicable to any type of descent trajectory (in particular to the straight line approach trajectory as the baseline selected here). They needs heavy image processing due to the tracking of multiple landmarks (with the induced problem of landmark disappearance from the camera FOV as the vehicle gets closer and closer from the target). The third concept needs the tracking of only one landmark (selected near the landing site) which should simplify a lot the image processing, with as a counterpart the necessity to have large variations of the LOS-to-landing site angle along the trajectory.

The three concepts have been investigated first by analytical evaluations. Then, the hybridation of the Inertial Navigation System with the optical measurements has been assessed by covariance analyses for the first and the third concepts. This has permitted to consolidate the analytical evaluations and assess the performances of the relative velocity estimation not tractable by analytical calculus. Good correlation between the analytical evaluation and the covariance analyses has been quoted.

It has been assumed that the Inertial Navigation System performances are those specified in Section 4.2. Then, with the assumption that a 40° x 40° FOV, 1000x1000 pixels camera is used with typical optical errors of 0.06° (3 sigma) and a 0.5Hz image processing frequency, it has been shown that the following relative navigation performances can be achieved:

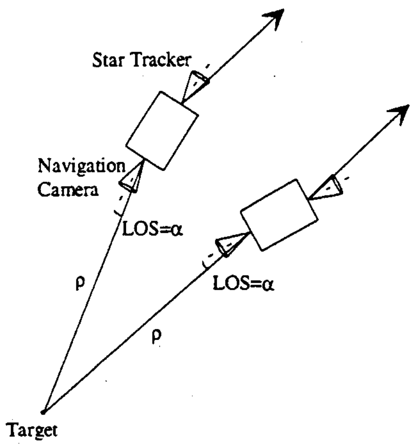
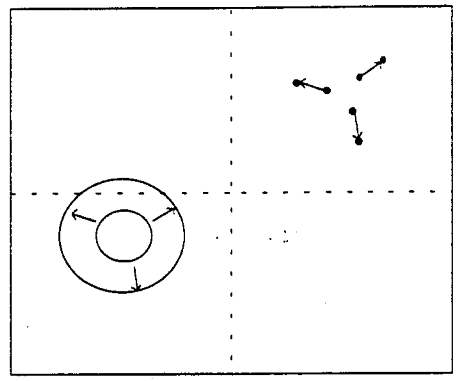


Figure 14a: Range + LOS Concept



Zoom effect in the camera Field of View

In a given time step, the dilatation of the pattern ("zoom" effect) is proportional to the inverse of the range

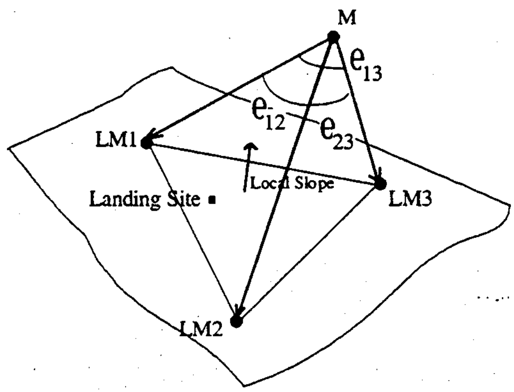


Figure 14b: LOS-to-Three Landmarks Concept

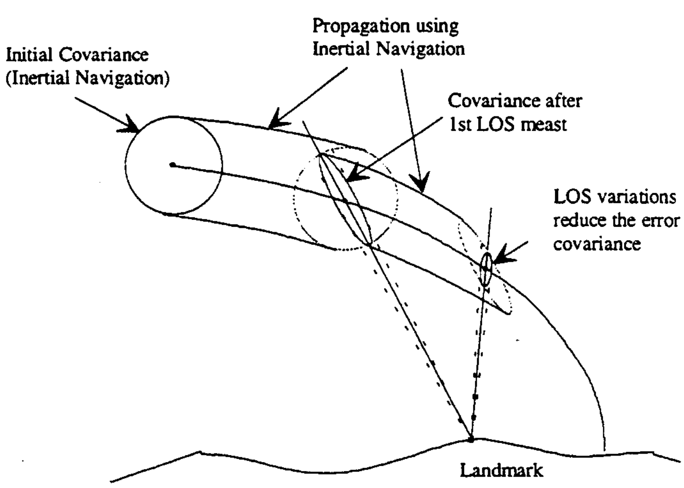


Figure 14c: LOS-to-One Landmark Concept

CONCEPTS	PERFORMANCE AT HIGH GATE (3 sigma)	PERFORMANCE AT LOW GATE (3 sigma)
LOS + range	150 m	3 m
LOS to multiple landmarks	30 m	landmark positioning accuracy
LOS to one landmark	20 m	4 m

The preferred concept at long range are the LOS to multiple landmarks (good performances, no constraint on the trajectory) or the LOS to one landmark (good performances, constraints on the trajectory but simple processing).

The preferred concepts at short range are the LOS + range concept (good performances, no constraint on the trajectory) or the LOS to one landmark (good performances, constraints on the trajectory but simple processing).

The most sensitive parameters which drive the performances are the following:

CONCEPTS	KEY PERFORMANCE DRIVERS
LOS + range	radial relative velocity error
LOS to multiple landmarks	FOV + landmark relative position errors
LOS to one landmark	LOS biases + LOS variations

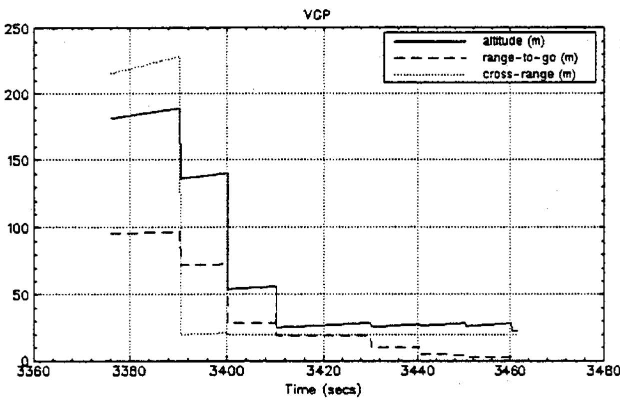
A **1-2 meters accuracy** can be achieved provided that the following conditions are fulfilled:

CONCEPTS	Requirements for a 1m navigation accuracy
LOS + range	Accuracy on relative velocity estimation: 0.2m/s 3 sigma at 200m
LOS to multiple landmarks	Landmark relative positioning errors below 1m
LOS to one landmark	LOS biases estimated with an accuracy of 0.01° 3 sigma. At least 20° LOS variations during the last 1000m of the descent.

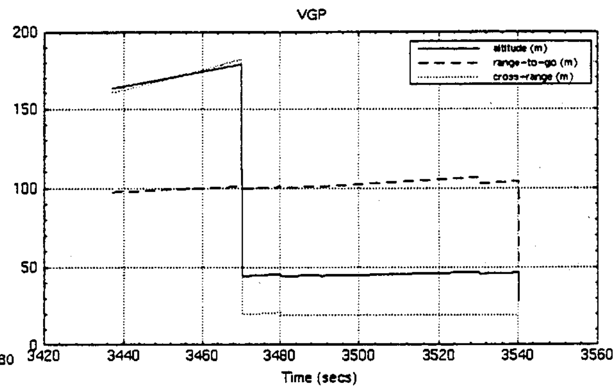
These ultimate performances can be achieved only through a close hybridation between the Inertial Navigation System and the optical measurements, using optimised navigation filters. The performances of such filters shall be verified by simulations.

Preliminary covariance analyses using CAPTAINS-X (see Figure 15) have shown that with such an hybridation, the relative velocity can be estimated with an accuracy of typically **0.2m/s (1 sigma)**, which avoids the use of a RF Doppler instrument as nominal navaid.

Additional features which must be assessed by detailed simulations are the convergence time of the relative navigation process (a typical convergence of 30s has been quoted in covariance analyses) as well as the interaction between the relative navigation bandwidth and the guidance/control system.



Trajectory with LOS variations



Baseline Trajectory (very weak observability)

Figure 15a: Validation of LOS to one Landmark concept by Covariance Analyses

Optical Measurements are hybridated with the IMU measurements

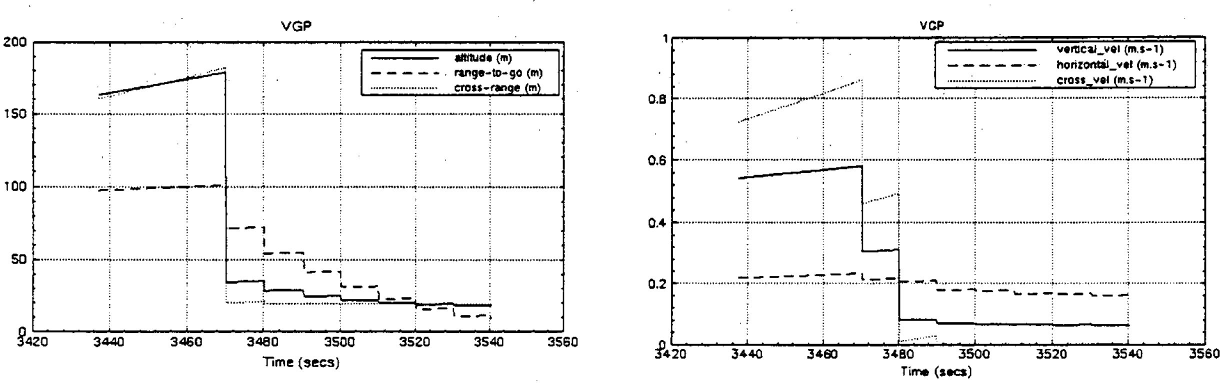


Figure 15b: LOS + Range Concept

Relative Velocity can be accurately updated by combining both optical and inertial measurements

4.4 The Final Descent Phase

Guidance/Control analyses have been carried out for the final descent phase to verify the capability of the GNC system to cancel the residual errors at Low Gate. These errors have been taken as: (30m, 1m/s) (3 sigma) (worst case from retargetting study). A control around a fixed guidance trajectory has been designed by appropriate control gain tuning. Low Gate characteristics are that defined in Table 4 Section 3.2.

A 25% throttling capability has been demonstrated necessary to achieve the desirable final accuracy. The Guidance/Control performances at the Main Engine Cut Off point (MECO) are those given in Table 16a. The plots given in Figure 18 summarizes the achieved Monte Carlo analysis for this phase.

It results from these analyses that the dispersions at landing due to Guidance/Control will be as expected an order of magnitude below the navigation errors.

It is worth noting that the obtained results have been obtained at the cost of an over-designed torque capability (no saturation on the torque level has been simulated). Recommendation is to optimise the final descent so as to avoid extreme torque demand.

The global GNC performances at MECO are presented in Table 16b where all the contributors to the final kinematics dispersions have been taken into account (cumulated inertial navigation errors from Low Gate to MECO, disymmetric engine cut-off, ...).

Dispersions	Mean	3 sigma
Down Range	42.4 cm	2.7 cm
Altitude	-23.7 cm	3 cm
Horizontal Velocity	-7.6 cm/s	1.2 cm/s
Vertical Velocity	5.8 cm/s	1.5 cm/s

Table 16a: G/C Performances at MECO

Dispersions	(3 sigma)
Horizontal Velocity	0.9 m/s
Vertical Velocity	0.9 m/s
Angular Rates	0.8 deg/s
Attitude Angles	negligible

Table 16b: GNC errors at MECO (all contributors)

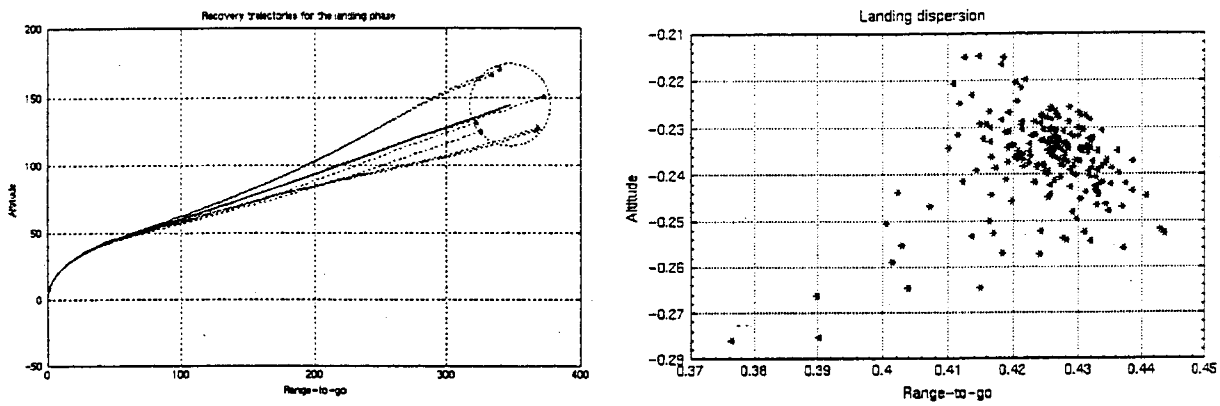


Figure 18: Result of G/C Monte Carlo Analysis

Recovery from Dispersions at Low Gate.

The errors at MECO have been translated into dynamical dispersions at touch down by considering the free falling phase. Table 17 below summarizes these dispersions (the attitude error has been evaluated with the assumptions that RCS is shut down at MECO).

Dispersions on	Dynamical Dispersions (1 sigma)
Horizontal Velocity	0.3 m/s
Vertical Velocity	0.3 m/s
Angular Rate	0.33 deg/s
Attitude	2 deg

Figure 17: Dynamical Dispersions at Touch Down

5. GNC HARDWARE SYNTHESIS

Taking into account the GNC specifications presented in Section 4, the following GNC, vision and propulsion subsystem hardware is recommended (redundancy is not taken into account here).

External Attitude Sensor

Single Wide Field-Of-View star tracker of the class (0.1°, 0.02°, 0.02°) 1 sigma. This star tracker is implemented along the anti-thrust direction, or in the thrust direction with a certain bias if it is used also as camera sensor for navigation. Autonomous star pattern recognition is a desirable requirement to simplify the operations. Such class of sensors exists in Europe.

Objective: 3kg, 10W

IMU

(0.1 °/hr, 50 microg, 300 ppm) IMU class. Numerous off-the-shelf equipment of this class exists in US in RLG and HRG miniaturized technology, soonly in FOG technology, and in Europe in RLG technology, and at medium term in FOG technology. More investigations are needed (dedicated consultations towards IMU equipment European manufacturers) to evaluate the impact of stringent mass and power requirements on a Lunar Lander development programmatics.

Objective (from US equipment): 4kg, 15W

Vision Camera

40°x40° FOV, 1000x1000 pixels CCD. Requirements on resolution at system level may increase the desired number of pixel. WFOV star tracker and CCD matrix under development at ESA presents this type of characteristics. The vision camera could be used as star tracker to minimise the system complexity. The tracking of typically 10-30 landmarks shall be achieved at a 0.5-1 Hz frequency. The image processing shall be done on a DSP computer. Such a device is being developed by ESA in the frame of the DSPDEV programme.

Objective (idem Star Tracker): 3kg, 10W

Range/Range Rate Instruments

Use of ranging and range rate instruments is not mandatory in the case where a vision system is implemented. Such instruments shall be used in the case where a « blind » landing on a smooth landmark-free area is to be achieved. A three-beam radar and a 1-beam Doppler system should be sufficient for a safe landing in this case.

Propulsion System

Eight LEROS 2B engines of 431 N each for main thrust, 12 LEROS 20 engines, of 22N each for attitude control (TBC). A 25% throttling capability is to be implemented.

6. SYSTEM OPTIMISATION

6.1 Introduction

The performance of the guidance, navigation and control systems in bringing the lander to the surface of the Moon affects the requirements on other parts of the spacecraft, in particular:

- the requirements on the propulsion delta-V budget for the descent
- the attitude control requirements on the RCS during the descent
- the ability of the landing gear to absorb the residual velocities on impact
- the mass of the avionics and associated power requirements.

To follow these interactions, an integrated vehicle System Model had been constructed as a part of the previous *Lunar Lander Technology* Study (ref. 3). The Model was reviewed and extended in this study, with modifications made to the delta-V requirements, the RCS and the landing gear models as a consequence of information learned. The Model is in the form of an Excel 5.0 spreadsheet containing models of all the subsystems and the mission and performances in an integrated whole. Geometric and related mass changes in the structure etc are included, so that the System Model represents a « closed » design at any time, and can be used to investigate the sensitivity of the performance to the design requirements and assumptions.

6.2 Delta-V Requirements

The reference mission assumes a shared Arioane 5 launch into GTO from which the spacecraft inserts itself first onto a phasing orbit, and subsequently a lunar transfer orbit. On arrival at the Moon, the spacecraft brakes into a 200 km polar orbit, and later makes a Hohmann transfer to a 50 km « reconnaissance » orbit before beginning its descent to the Lunar surface. The delta-V requirements for transfer to lunar orbit depends on the angle between the GTO plane and that of the Moon's orbit. The reference impulsive delta-V for transfer from GTO to 200km lunar orbit is 1542m/s. Figure 19 shows the effect of increasing the delta-V to provide a wider range of launch opportunities. With a fixed launch mass, increasing the delta-V by 250 m/s reduces the payload by more than 30%. However, this can be recovered by increasing the vehicle all up mass by just 12.6%.

Descent delta-V requirements were provided by the results obtained in the Trajectory Analysis and GNC requirements. The descent delta-V is the sum of:

- the ideal impulsive descent delta-V (1702 m/s)
- gravity loss in the inertial descent phase (106 m/s)
- gravity loss during the visual approach, determined by the thrust-to-mass ratio and the angle of approach (136m/s)
- gravity loss in the final descent afeter visual approach, determined by the time of rotation of the vehicle (4m/s)
- a control tuning error (10m/s)
- a VGPEP location error correction (8m/s)
- a thrust dispersion allowance (4m/s)
- and an allowance for retargetting during the visual approach (7m/s)

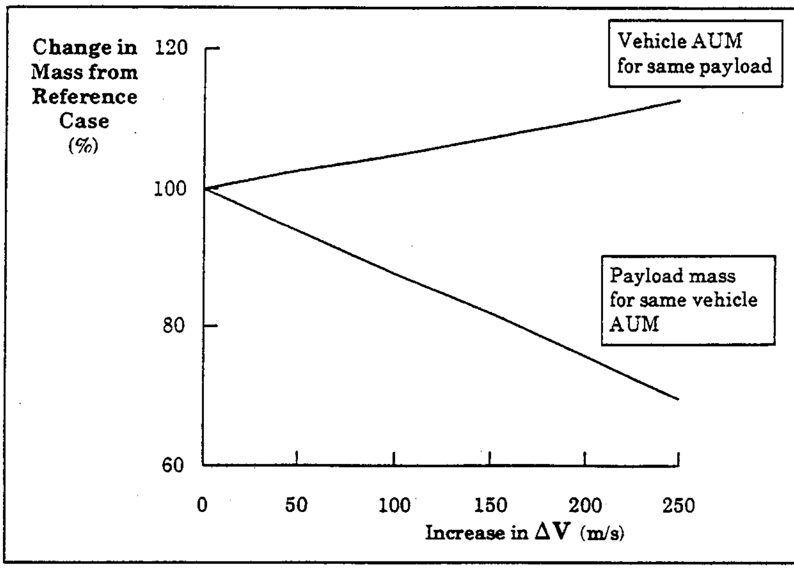


Figure 19: Effect of changing delta-V requirements from GTO to Lunar Orbit

For the « reference case », the descent delta-V is 1976 m/s. This value will change with the angle of approach and the mean throttle setting required to permit thrust magnitude control of the descent trajectory (Figure 20). Increasing the angle of approach to 30° rather than 20° reduces the payload by about 5 kg, about the same as setting the mean throttle to 90% full thrust.

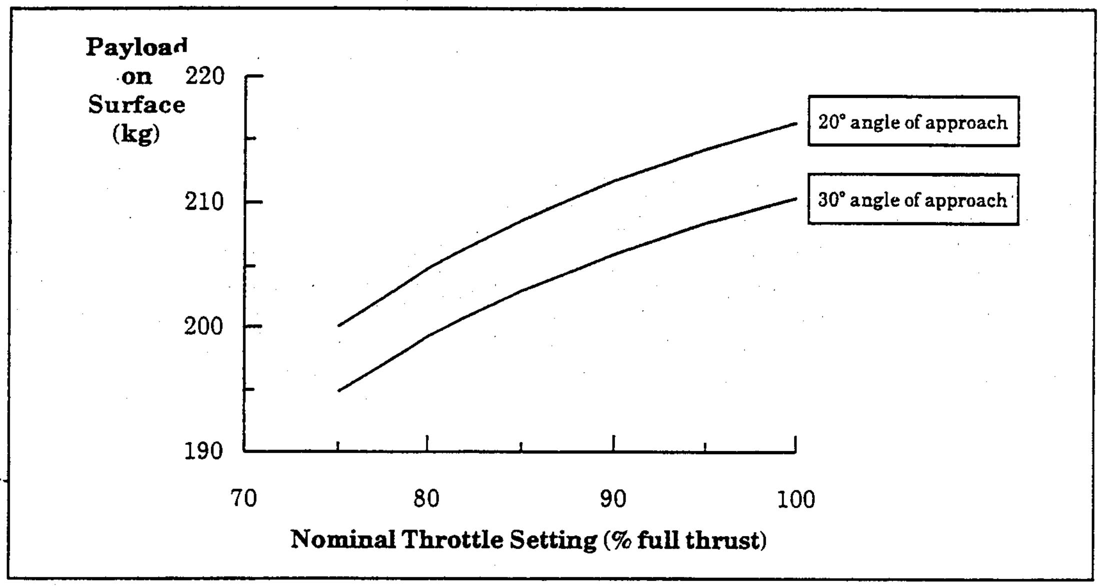


Figure 20: Effect of Descent Throttle Setting and Angle of Approach

6.3 Touch Down Dynamics

The GNC system affects the vehicle touchdown due to the residual errors in position and velocity at the main engine cut off (MECO). MECO dispersions principally affect the forces encountered by the landing gear. The *Lunar Landing Technology* study indicated that where sliding can occur at touch down, stability is unlikely to be a problem. Figure 21 shows that there is an optimum « maximum impact g-level ». Higher impact g-levels generate higher forces on the landing gear, but shorter gear strokes. Conversely, reducing the maximum impact g-level leads to larger dimensions for the landing gear.

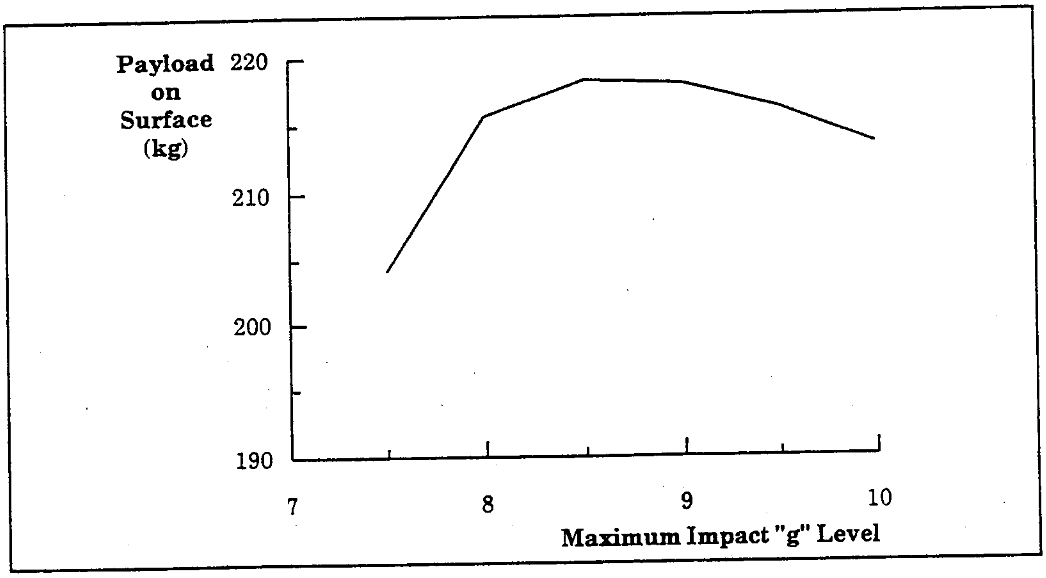


Figure 21: Effect of Impact « g »-level

Velocity and position errors remaining at the end of the final approach can be approximated by impact velocity. The vertical velocity at impact is clearly determined by the residual velocity at MECO together with the altitude at which this happens. Figure 22 shows the effect of impact velocity at touchdown on the payloard performance. The estimated 3-sigma MECO residual velocity is about 0.9m/s. With this terminal velocity, the effect of changing altitude at MECO is surprisingly small. Raising the MECO altitude from 10m to 20m drops the payload by only 5 kg. If this result can be substantiated, it leads to the conclusion that the lunar lander can be designed with a good margin on MECO altitude without much sacrifice of payload.

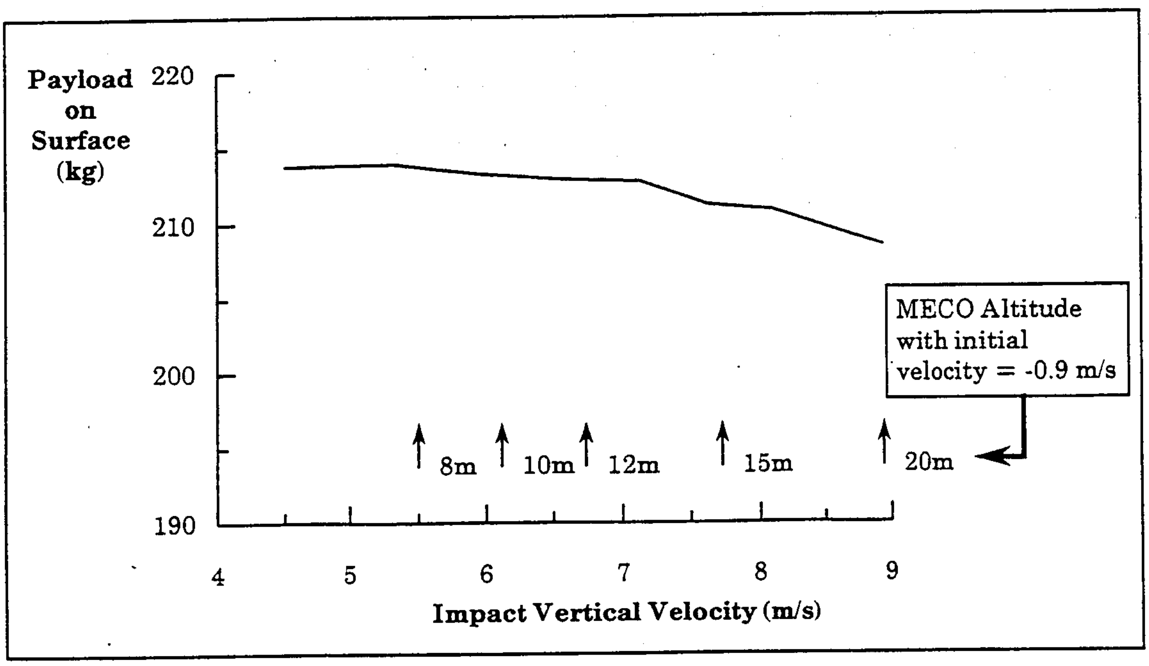


Figure 22: Effect of MECO altitude / residual velocity

6.4 Avionics Assumptions

The vehicle avionics systems were given assumed masses and powers at the start of the study. These mass and power requirements can be treated as block values. What matters is how much overall mass and power demands change, Figure 23 shows how increases in avionics mass and power affects the payload. Both avionics mass and power affect other parts of the system. An increase in avionics mass of 1 kg leads to a 1.24 kg decrease in payload. Similarly, the effect of increasing power demand is an approximate 0.51kg/watt decrease in payload. In fact, the expected increase in avionics power (shown by the arrow) are likely to have more effect than the expected increases in avionics mass.

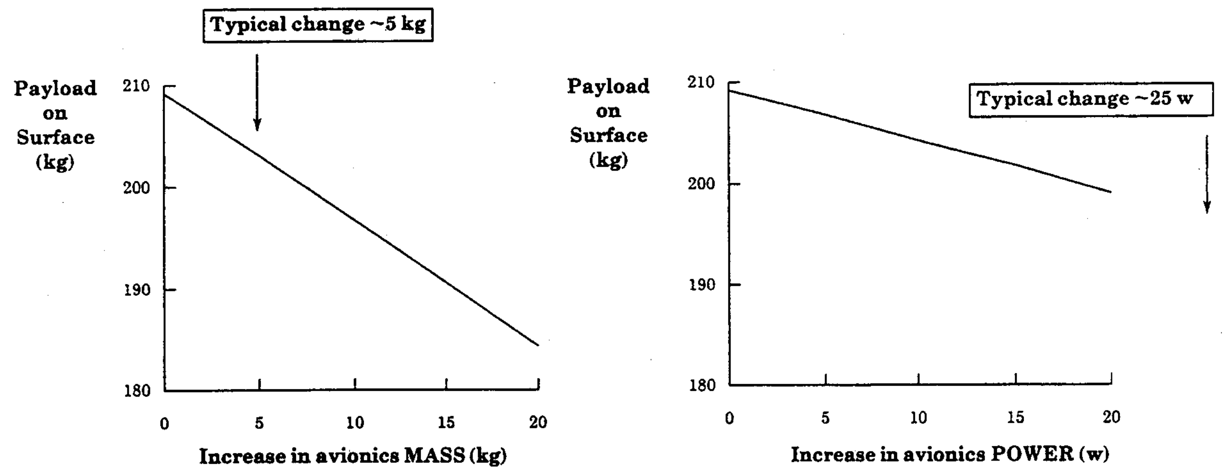


Figure 23: Effect of Increasing Avionics mass and power on Payload

6.5 Propulsion & Structure

With 70% of the vehicle all up mass in the form of propellant, the vehicle performance is sensitive to propulsion assumptions. Current specific impulses for 400N bipropellant engines vary between 3072 m/s (313 sec) and 3129m/s (319sec). Using the higher specific impulse (LEROS 2B) increases the payload by about 20kg over the initial reference design. Projections that the specific impulse of these engines can be raised to 3188m/s (325 sec) are currently being made. Modest reductions in the tank pressurization from 1.42 MPa to 1.2 MPa also adds about 3 kg of payload.

The main propulsion system also affects the demands on the RCS to provide control torques during powered manoeuvres. Pitch and yaw moments generated by the RCS during main engines burns constitute additional thrust to the main engine, so that RCS propellant does not have to be accounted for as additional propellant. Control torque demands are generated by:

- main engine thrust differences
- main engine thrust centre misalignments (assymetries)
- main engine thrust vector angle misalignments
- vehicle centre of gravity offsets
- the need to turn the vehicle.

During the descent, the most important of these turns out to be the centre of gravity offset generated by differential draining of opposing propellant tanks. If there is a 1% difference in propellant consumption from opposing tanks, then close to touchdown, the control moment torque generated by the offset centre of gravity is equal to the whole control moment generated by two 22N RCS thrusters. A number of alleviating measures is possible, including moving the tanks closer to the vehicle axis (see below), but an attractive approach appears to be to use the « slow acting latch valves » on each tank (which primarly provide for engine throttling) to re-balance the draining based on the RCS pitch/yaw demands. This might have the additional advantage of allowing a reduction on the propellant residuals allowance, which would also add to the payload.

Preliminary analyses have shown that pulse modulation of the main engines for pitch and yaw control generates a wobble in the vehicle not compatible with precise control of the vehicle during the final approach phase, but acceptable in the earlier homing phase to give a more rapid pitch transition to the line of site approach trajectory. This point shall be investigated in more details in further analyses.

During the lunar orbit phase of the mission, non-uniformities in the lunar gravity field cause progressive distorsions in the orbit. Maintaining a precise circular orbit at 200 km for one month would required about 11kg of RCS propoellant. However, only the circularity of the orbit is affected. The requirements could be abandoned for limited stay times (about 1 month) in the higher lunar orbit.

Since available control torques would be improved by moving the propellant tanks closer to the axis of the spacecraft, the effect of reducing the core diameter of the vechile was investigated (see Figure 24). The circles on the curve represent standard Ariane interface diameters. There appears to be a substantial mass saving and improvement in payload with reducing the core diameter to as low as 916mm. At this stage, there may be interference problems with the current design between the landing gear side arms and the engines. This could be accomodated either by bringing the landing gear side arm attachment points out from the launch interface ring on a bracket, or by using a core and cylinder centre structure to use a larger diameter interface ring.

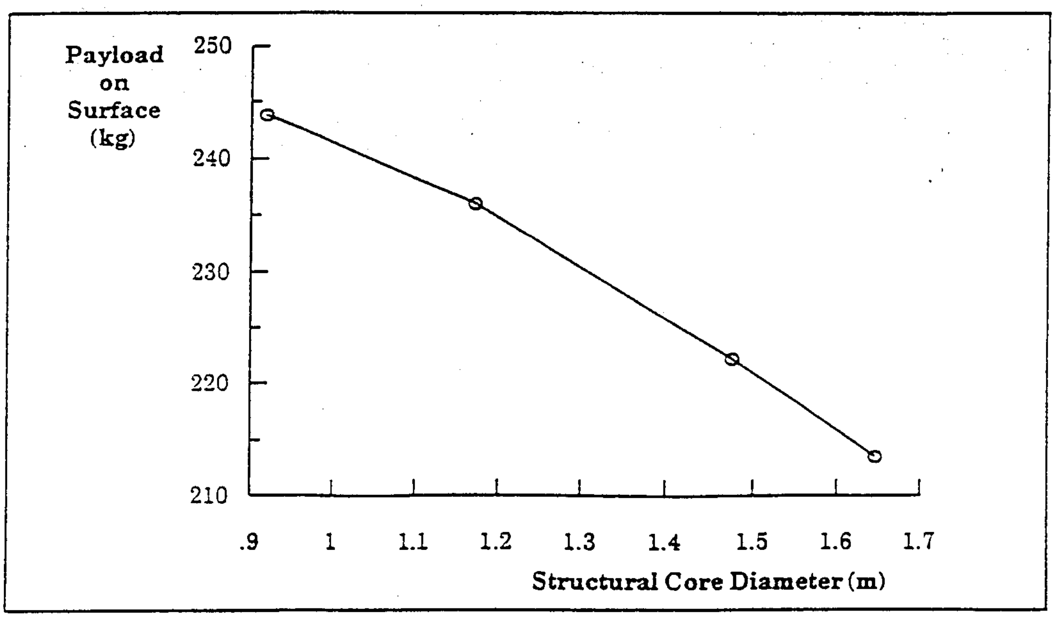


Figure 24: Effect of Structural Core Diameter on Payload Performance

6.6 Vehicle Re-Optimisation

The effects of the various system changes have been incorporated into the System Model. Table 25 constructs an evolution of the payload mass as each of the changes is introduced. By far, the largest mass change has been introduced by reducing the structure core diamter. One interesting result is that the optimum impact « g » level changed significantly with the change in structure geometry.

The effect of changing the all up mass on the delivered payload is shown in Figure 26. Because a significant part of the mass of the vehicle does not change with the all up mass, added mass translates into payload at a rate of about 0.2 kg per kg of added all up mass, compared with a payload fraction at the reference case of 8.8%.

System Changes	Payload Mass (kg)
Initial reference case	206.6
Avionics update	188.1
landing accuracy changed to 0.9m/s 3 sigma at 8m altitude	187.4
no orbital maintenance in 200km orbit	188.8
optimise impact « g » level to 8.3 g maximum	197.1
increase specific impulse from 3070m/s to 3129m/s	215.4
throttle engines to 80% for descent	203.0
change propulsion system pressure to 1.2 MPa	205.4
change core diameter to 937mm	249.1
reoptimise max impact « g » level to 7.4 g optimum	254.3

Table 25: Evolution of Payload with Design Changes

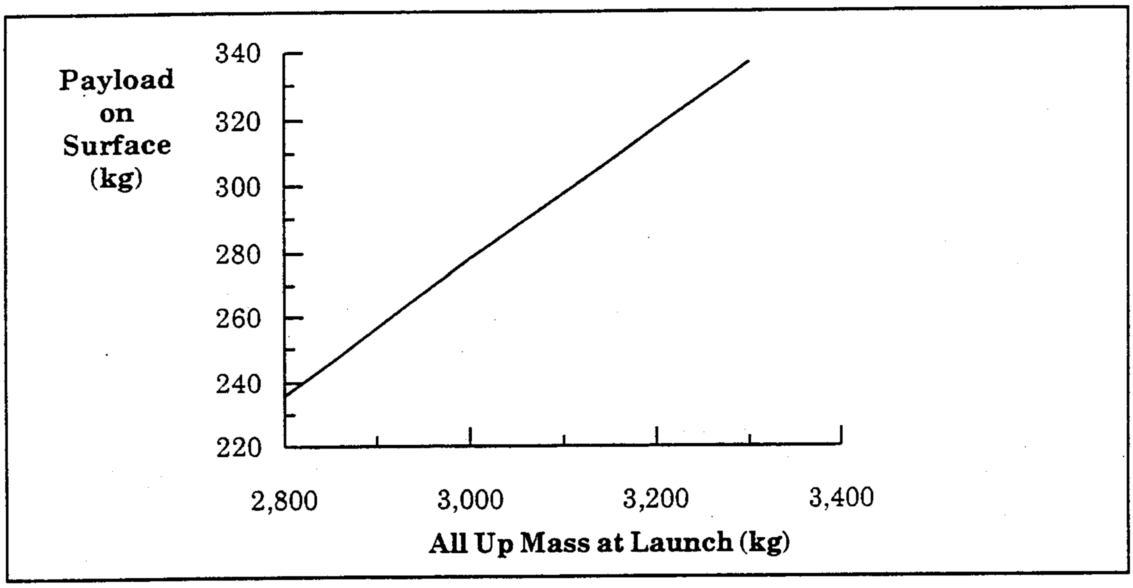


Figure 26: Effect of Changing All Up Mass

PAGE INTENTIONALLY LEFT BLANK

7. STUDY SYNTHESIS & RECOMMENDATION

The feasibility of a vision-based GNC system for safe and accurate lunar landing have been demonstrated by analyses. The specifications induced on the navigation subsystem and other subsystems (propulsion, vision, landing gear) have been derived. They are in a large extent compatible with the state-of-the-art technology.

Star tracker is required for attitude estimation prior to the powered descent phase. A Wide FOV star tracker can be used. The opportunity to have the same equipment used as attitude sensor and navigation camera is to be further investigated.

An Inertial Measurement Unit is needed for the powered phase. The specifications issued from the study are compatible with off-the-shelf equipment (specially in US). Further iterations with European manufacturers are needed to evaluate the impact of the stringent mass & power requirements on a Lunar Lander development programmatics.

Whether an advanced vision system is used for landing, it has been shown by preliminary analyses that the use of a radar/Doppler system is not mandatory to ensure a safe and accurate landing. This point shall be confirmed by more detailed analyses on the coupled vision/navigation aspects.

It has been shown that a thrust throttling capability is mandatory in order to achieve a safe and accurate landing. The impact of this requirement on the development of the propulsion system of a Lunar Lander shall be investigated in more details.

Ground tracking is necessary during the coast phase prior to the powered descent phase, which can induced constraints on the mission operations. Performances are highly sensitive to the Lunar Gravity Field model errors and descent orbit characteristics. Further analyses, including Lunar Gravity Field identification during orbiting phase and sensitivity to descent scenario are required.

Tight timing during the Approach Phase claims for advanced guidance schemes to recover from possible large dispersions at High Gate and increased re-targetting capabilities for obstacle avoidance. Adaptive guidance schemes shall permit to optimise the system performances and possibly to relax GNC requirements.

The process of optimal landing site selection combining both guidance and vision aspects shall be investigated in details to derive robust obstacle avoidance capability.

Relative navigation during the Approach Phase is a key driver of the system performances. Tight hybridation between the Vision System and the Inertial Navigation System shall be further investigated to derive the ultimate performances of the system.

The image processing supporting relative navigation and obstacle avoidance is a quite new development (software and processing hardware) which claims for prototyping and testing activities on dedicated test beds.

The trajectory design shall be refined by taking into account requirements from the relative navigation, adaptive guidance, manoeuvrability of the vehicle.

All these recommended activities shall be followed by a further iteration in order to evaluate their impact at system level.

PAGE INTENTIONALLY LEFT BLANK