DOI: 10.13009/EUCASS2017-221

ALTAIR Orbital Module Preliminary Mission and System Design

Andrea Tromba, Cédric Dupont*, Giulio Molinari**, Christoph Karl** and Eduard Diez****

** Bertin Technologies, France - andrea.tromba@bertin.fr, cedric.dupont@bertin.fr*

*** ETH Zürich, Switzerland - molinari@mavt.ethz.ch, karlc@ethz.ch*

**** GTD Sistemas de Información S.A, Spain - eduard.diez@gtd.eu*

Abstract

This paper presents the preliminary system and mission design of the Altair Orbital Module, a cost-effective and versatile multi-mission platform able to provide a dedicated and reliable orbit injection service for micro-payloads. An innovative approach relaying on a sustained use of design to cost and multidisciplinary design optimization techniques has been applied from the very beginning of the design phase to meet both performances and cost requirements. The system has then been conceived to assure high mission flexibility, allowing accommodation of either single or multiple payloads and providing accurate and safe deployment in a range of LEO target orbits. The final reference design consists in a lightweight composite structure, an innovative avionics and a dedicated monopropellant system based on environment friendly hydrogen peroxide, which provides multiple burst capability for orbital transfer manoeuvring as well as attitude control and de-orbiting capabilities.

1. Introduction

The market of small satellites under 200 kg is expected to increase dramatically in the next decades due to several factors such as miniaturization, availability of Commercial Off-the-Shelf (COTS) components and constellation projects. However, currently no launch system adequately addresses this market without the constraints of existing solutions such as piggyback launch. A dedicated system providing an available, reliable and affordable launch service without these constraints would enable the development of small satellites applications.

The ALTAIR project (Air Launch space Transportation using an Automated aircraft and an Innovative Rocket), which has been selected in the frame of the European Union's Horizon 2020 research innovation program (grant agreement No 685963), is aimed at preparing the development of such a launch system. ALTAIR's strategic objective is to demonstrate the economic and technical viability of a future available, reliable and competitive European launch service for the access to space (Low-Earth Orbit) of nano and micro satellites [1].

ALTAIR is an innovative semi-reusable air-launch system consisting of a reusable unmanned aircraft carrier, an expendable rocket launch vehicle and a cost-effective ground segment. Its reference mission is to carry 150 kg of payload(s) to a sun-synchronous orbit at 600 km. The mission profile expects the carrier takes-off from the ground bringing the rocket vehicle at the right altitude, drops it and comes back to the ground for further reuse. After releasing, the launch vehicle boosts the payload over its operational orbit (Figure 1).

Figure 1: Altair System and Mission Overview

DOI: 10.13009/EUCASS2017-221

Risk mitigation, cost savings, reliability and performances are major drivers in the development of the final system, requiring the ability to handle the flimsy balance between technological innovation, low cost solutions and maturity level. This led to a space launch vehicle concept (Figure 2) built on the integration of a low-cost and green hybrid propulsion (solid inert fuel – HTPB – and high-density green storable oxidizer – 87.5% Concentration H_2O_2 –), lightweight composite structure, innovative modular avionics and a smart multi-mission upper-stage [2].

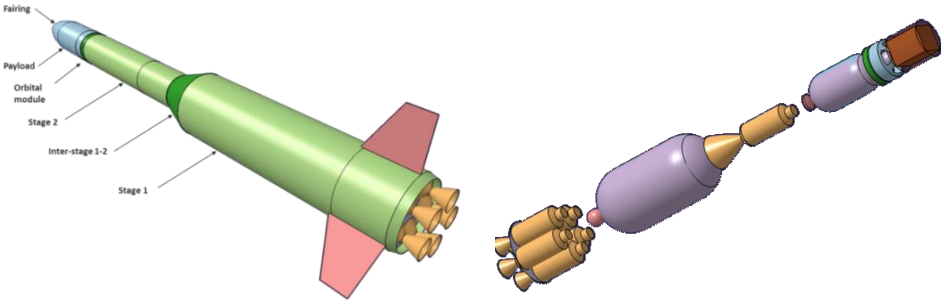


Figure 2: General architecture of the ALTAIR launch vehicle

The most of existing space launch systems provide final payload injection into the target orbit through their upper stages. Such systems have always been designed for a specific launch vehicle and for a defined payload category. For this reason, deep and expensive modifications are usually required to adapt their interfaces and performances in order to meet specific mission needs or to accommodate multiple payloads. So, the payloads shall, usually, yield to the launch vehicle specifications to the detriments of their own needs. About propulsion, existing small upper stage, as for example the North American Pegasus's 3rd stage or the European VEGA's AVUM module, are usually equipped with system relying on solid fuel or on hydrazine liquid monopropellants. In both cases, these solutions are dangerous both for humans and environment: solid propellant is explosive and can produce space debris while hydrazine is, at the same time, explosive, toxic, carcinogenic and polluting. Finally, existing upper stages, usually, do not provide end-of-life disposal capability contributing to space debris population increasing and rising risks for space assets and humans on ground.

This paper is focused on ALTAIR Orbital Module (OM), which is conceived to provide a cost effective and reliable space access solution for multiple small payloads offering a valuable answer to all the state-of-the-art drawbacks mentioned above.

2. Mission Profile & Functional Requirements

The Orbital Module is part of the upper section of the ALTAIR Space Launch Vehicle, named Orbiter. It consists in the Fairing, the Payload Module (PLM) and the OM itself. OM functional requirements (Table 1) are directly derived from a breakdown of ALTAIR System requirements and constraints. Orbital Module mission profile is showed in Figure 3.

Table 1: ALTAIR OM Functional Requirements

Payload accommodation and injection on the target orbit	Orbital Transfer	LV Booster provided initial orbit	-50 km x 600 km i = 97.8°
		Nominal Target Orbit	SSO 600 km
	Payload	Nominal Mass	> 150 kg
		Available Envelop	(d)1.2m x (h_{max})1.5m
		Payload number	Single and multiple payload
Mission Profile Execution	Orbital control	Circularisation	ΔV > 240 m/s
		Orbital Correction	ΔV > 10 m/s
		Collision Avoidance Manoeuver	1 km separation
		EoL Disposal Manoeuver	According to LOS
		Restarting Capability	Multiple boosts
	Attitude determination & control	Attitude determination	Position & orientation function of time
		Attitude control	3-axis control Spin up/down
			Perturbation compensation
Launch and operational environment	Quasi Static Load	Lateral	2 g / 5 g
		Axial	8 g / 10 g
	Thermal-vacuum environment	Temperature	-30 °C / +40 °C
		Pressure	< 10^{-4} Pa
Interfaces	Payload	Mechanical I/F	Payload accommodation & separation
		Electrical I/F	Payload data link
	ALTAIR System	Mechanical & Electrical I/F	Fairing connection and separation
			2nd stage connection and separation
			Ground Segment handling and umbilical

DOI: 10.13009/EUCASS2017-221

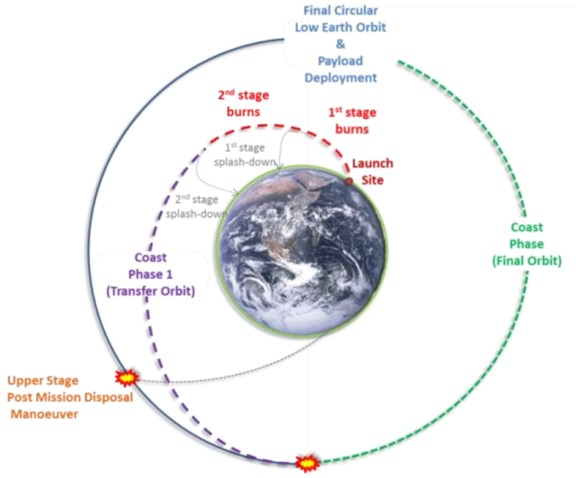


Figure 3: Altair System & OM Mission Profile

The Orbital module main function is to accommodate the payload (single or multiple satellites) and safely complete its transfer on the targeted orbit starting from the initial conditions provided by the launch vehicle main booster stages. Multiple boosts capability will ensure flexibility in final orbital condition provided according to payloads needs and will ensure accurate positioning as well as collision avoidance capability in case of space debris impact risk. Finally, The OM shall be compliant with the French Space Operations Act (LOS), so it shall provide an End of Life decommissioning through an appropriate disposal manoeuver.

3. Design methodology

ALTAIR OM concept is largely based on heritage coming from previous studies carried out by Bertin Technologies, leader of the Altair Launch Vehicle and Orbital Module design, on Micro Launch Vehicles. In particular, ALTAIR OM is based on works carried out in 2014 and 2015 in the frame of the preliminary design of ROXANE MLV (a two-stage- to-orbit LOX/CH4 expandable micro-launch vehicle project co-funded by Bertin Technologies and the French Space Agency, CNES) and of its versatile upper stage, named VENUS [3].
In the frame of ALTAIR project, four consortium members are mainly involved in the Orbital Module design activities:

- Bertin Technologies (France) is the mission and system design authority. It is in charge of the coordination the technical team, of the system engineering activities and of the definition of the propulsion architecture.
- ETH Zurich (Switzerland) is in charge of the mechanical architecture design and optimization.
- GTD (Spain) is in charge of the avionic design for the all launch vehicle and, particularly, for the Orbital Module.
- NAMMO (Norway) is in charge of the hybrid rocket engines design for ALTAIR Launch Vehicle and it is involved as propulsion expert and potential technological provider for the OM propulsion system.

In this context, design-to-cost, Multidisciplinary Design Optimization (MDO) and collaborative engineering techniques (Figure 4) have been applied from the very beginning of ALTAIR OM design in order to orient the partners work and achieve rapidly and effectively a reference concept, able to provide the right balance between performances, mission flexibility, costs and risks.

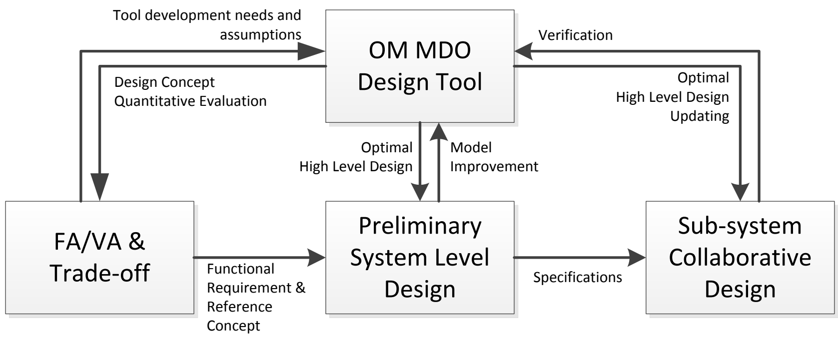


Figure 4: ALTAIR OM Design Strategy

DOI: 10.13009/EUCASS2017-221

From the early phases, mission and accommodation flexibility requirements, safety and environment constraints, as well as recurring and development costs have been addressed in the functional analysis to find the right mix of innovations and mature technologies. This required exploration of several options for the global architecture, the propulsion system or the mission strategy. Dozens of possible concepts were possible so, in order to cover the largest range of design opportunities, a multidisciplinary design approach has been applied. This led to the development of a preliminary design MDO tool able to automatically produce optimum system level designs according to several technical variables and mission scenarios. The MDO tool is used to provide inputs for all the design phase, including the functional and value analysis, the trade-off, the preliminary system design and the collaborative engineering. Being evolutionary and flexible the tool can be easily updated, so it is continuously improved in each step of the design process. Finally, in the latest phase, it will be also used to check the coherence between sub-system level designs defined through a collaborative approach.
Fairing and Orbital Module are strictly interconnected and their design shows strong interdependences. For this reasons in the OM design fairing is also taken into account and addressed in this paper.

3.1 System Level Multidisciplinary Design and Performances Optimization

Astrodynamics, propulsion and structural mechanics disciplines have been coupled together with nonlinear optimization algorithms in the preliminary system optimization loop showed in Figure 5. Main modules and computation approach are briefly described in Table 2.

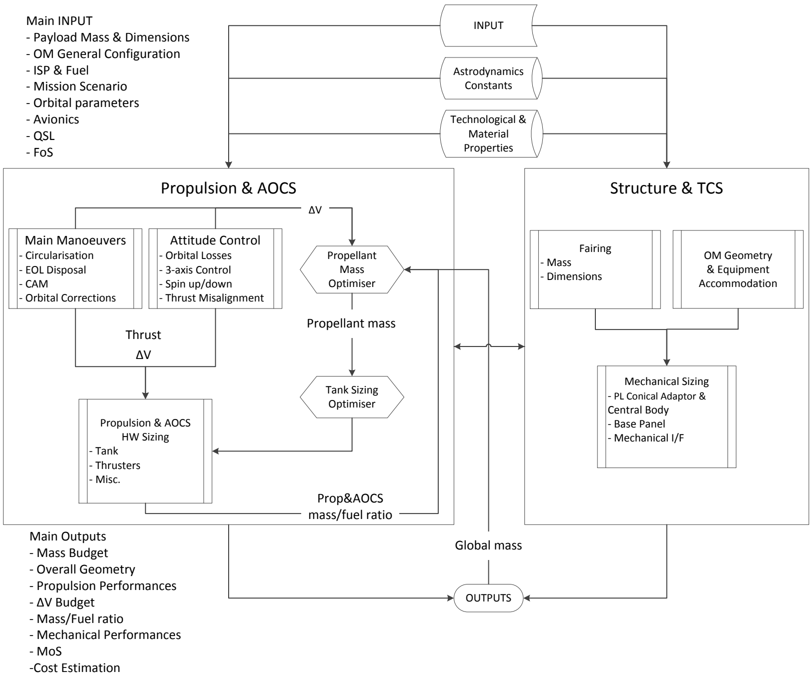


Figure 5: ALTAIR OM System Optimisation Design Loop

3.2 Propulsion Sub-system Design and Performances Optimization

Bertin Technologies has developed a preliminary design and optimisation tool to assess various potential propulsion architecture and technologies for the ALTAIR OM. This tool is based on the design methodologies proposed by Larson and Wertz [4] and it allow to:

- Model performances of various propellant technologies:
  - Hypergolic bi-propellants: nitrogen-tetroxide/monomethyl-hydrazine (N_2O_4 MMH, Isp ~ 280 sec);
  - Monopropellants: hydrazine (N_2H_4, Isp ~ 220 s), hydrogen peroxide (H_2O_2, ~ 160 s), cold nitrogen gas (N_2, Isp ~ 60 sec);
- Model various propulsion system architectures:
  - Monopropellants or bi-propellants architectures;
  - Pressurisation: blow down and pressure fed systems;
- Size thrusters;

DOI: 10.13009/EUCASS2017-221

- Size tanks;
- Preliminary model attitude control configurations;
- Preliminary assess losses due to finite thrust and mounting configurations;
- Preliminary assess the cost of the system;
- Provide a mass budget (including propellant feed line equipment as piping, filters and valves).

Table 2: ALTAIR OM MDO module description

Module	Discipline	Objective	Main model approach and assumptions
Propulsion & attitude control	Astrodynamics	Velocity increment computation	Impulsive manoeuvers and Hohmann transfer
	Attitude control	Performances and propellant mass requirements	Orientation manoeuvers, Perturbations compensation (residual drag, gravity gradient, thrust misalignment)
	Hardware sizing	Tank and thruster design, Pressurisation (blow down or pressure fed) design, mass budget.	Blow down or pressure fed system. State-of-the-art equipment for the feed line.
	Optimisation	Propellant consumption optimisation	GRG Nonlinear solver
Structure and Thermal Control System	Structure	Margin of safety verification, primary structure sizing, geometry definition, mass budget	Monolithic or composite lightweight structure, multiple material properties (CFRP, Ti6Al4V, Al7175, etc.), beam model based on various failure modes (von Mises yield criterion, buckling and stability).
	Thermal control	Mass budget	Simplified mass model for a passive system.

This design module has been integrated in the MDO tool described in Chapter 3.1 and it has been used to analyse various potential option for the OM architecture to feed the trade-off analysis described in Chapter 4 and to provide the preliminary design of the OM propulsion and attitude control system described in Chapter 5.

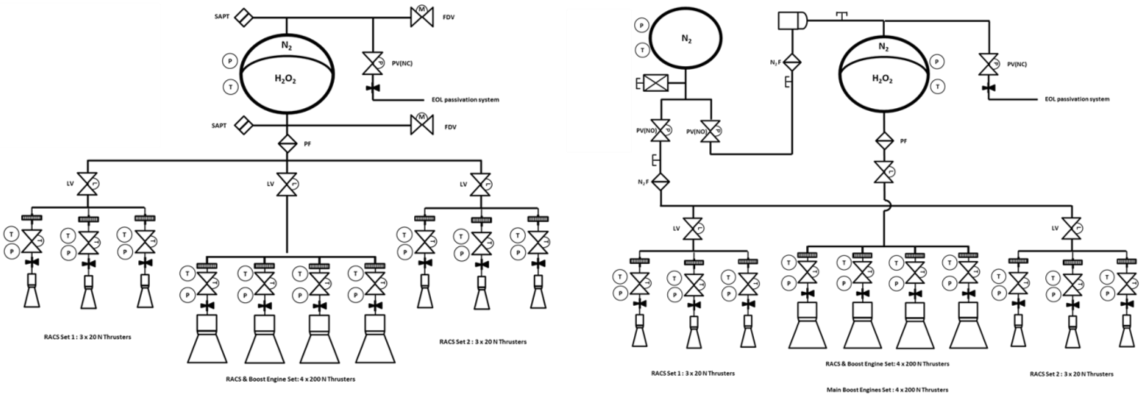


Figure 4: Example of OM potential propulsion system architecture.

3.3 Structural Optimization

Since the Orbital Module (OM) constitutes the uppermost stage of the ALTAIR launch vehicle, lightweight design of its load-carrying structures is of paramount importance to increase the overall performance of the launch vehicle.
The first preliminary mechanical sizing is provided by the MDO tool described before. It is based on beam theory and it takes into account various technologies (metallic or composite) and various failure modes (von Mises yield criterion, buckling and stability) trying to optimise the margin of safety associated to the maximum static load scenario showed in Table 1, by applying a Factor of Safety of 1.25.

DOI: 10.13009/EUCASS2017-221

After this first evaluation a further and more detailed mechanical optimization and sizing is carried out in the frame of the sub-system collaborative design phase. To maximize the efficiency of the OM structures, composite materials are considered in combination with advanced lightweight structural designs, including sandwich and anisogrid construction. While the cylindrical body of the OM enclosing the internal subsystems will feature a composite sandwich construction with aluminium honeycomb core, the truncated conical payload adapter interfacing the spacecraft is designed in an anisogrid construction, additionally enabling accessibility to the internal subsystems of the OM. The preliminary design of these Orbital Module structures is based on optimization for minimum structural mass under analytically formulated design constraints, which represent the main failure mechanisms of the structures under compression and bending loading. Specifically, these include global buckling, local buckling, and material failure.

For the cylindrical body of the OM, the considered failure criteria include global buckling in the axisymmetric and non-axisymmetric mode, facesheet wrinkling, and core shear instability. In the preliminary optimizations, the critical loads are calculated using models proposed in the literature [5] [6] [7] for compression-loaded sandwich cylindrical shells. To account for reduction in structural performance caused by geometric imperfections, knockdown factors are applied on the global buckling loads. Although Stein and Mayers [8] found early in the 1950s that sandwich cylindrical shells are less sensitive to imperfections than monocoque cylinders due to the greater relative wall thickness, few studies have been conducted to study the imperfection sensitivity of sandwich cylinders, and to derive suitable knockdown factors for preliminary design purposes. The semi-empirical knockdown factors (KDF) proposed by NASA in the late 1960s [9] were shown to be overly conservative and outdated for sandwich cylindrical shells made of advanced composite materials [10], mainly due to evolutions in both the material performance and manufacturing processes. However, since the introduction of NASA SP-8007, no other guidelines have been published regarding the prediction of KDFs for sandwich cylindrical shells. Recently, various numerical investigations on the imperfection sensitivity of composite sandwich cylindrical shells based on the Single Perturbation Load Approach [11] [12] have shown that a knockdown factor of around 0.8 provides a reasonable and still conservative lower bound to the critical buckling load of axially loaded composite sandwich cylinders for aerospace applications. This is significantly larger than the more conservative KDFs resulting from NASA SP-8007, which are in the order of 0.65.

For the payload adapter (PLA), the governing failure criteria include global buckling of the structure, local buckling between intersecting ribs of the anisogrid structure, and material failure. The constraint formulations used for the critical loads of anisogrid conical shells were initially established by Vasiliev et al. [13] [14]. Regarding imperfection sensitivity and knockdown factors, anisogrid structures are reported to possess little sensitivity to geometric imperfections, owing to the shape stabilization effect, which causes the shell cross section to take on a circular shape upon axial compression, even if some initial imperfections are present. Although literature states that theoretical predictions for the critical buckling loads of anisogrid shells agree well with experiments due to this effect, a conservative knockdown factor of 0.8 was used in the preliminary design.

The OM structures were sized for the maximum enveloping line load resulting from the axial compressive force and bending moment. Preliminary loads analyses have shown that the dimensioning load case for the OM structures is the maximum longitudinal acceleration occurring at 2nd stage burnout. Concerning the structural materials, the carbon/epoxy material IM7/8552 was considered in the preliminary structures design, an intermediate strength composite material typical for aerospace applications. For sandwich construction, the commercial aluminum honeycomb Hexcel 1/8-4.5-0.001-5056 was used.

To determine preliminary optimized designs of the OM structures in the early design phase, numerical optimizations were carried out for the design variables characterizing the structures, using the in-built Matlab solver *fmincon* in combination with the *sqp* algorithm. The objective of the optimization is to minimize the structural mass while satisfying the inequality constraints given by the critical failure loads. The particular models used for the preliminary sizing of the cylindrical composite shell consider laminate anisotropy in terms of the stiffness in the axial and transverse direction, neglecting stacking sequence and coupling effects. For the cylindrical body of the OM, the continuous optimization variables were the core and facesheet thickness. The laminate considered had a balanced and symmetric quasi-isotropic layup. In order to obtain feasible optimum solutions, manufacturing constraints were imposed on the design variables. For the sandwich structures, these constraints included a lower bound on the honeycomb core thicknesses, as well as the minimum facesheet thickness required to maintain a balanced and symmetric layup. For the anisogrid PLA, the four design variables were the rib thickness, the rib width, and the angle and spacing of the geodesic ribs. The remaining design parameters describing the structure are dependent on these variables and follow from geometric relations. The integer number of circumferential ribs is required as an input for the optimization and was chosen as six according to preliminary studies, to obtain a reasonably dense system of ribs for the given PLA geometry. Furthermore, the truncated conical shell was assumed to end with a circumferential rib

DOI: 10.13009/EUCASS2017-221

at its lower and upper end, which is in accordance with the requirements for interface rings. The manufacturing constraints considered for the PLA include a lower bound on the rib width of 4 mm, as well as equal width of the helical and circumferential ribs, in order to comply with the continuous filament winding process used for manufacturing such structures. The optimization results in terms of the preliminary design and mass of the two major structural components of the OM are presented in Chapter 5.2.

3.4 Avionics Sub-systems Design and Optimization

The avionics subsystem concerns all the electrical and electronic equipment including SW interfacing the actuators and sensors. The functional analysis of the avionics leads to the identification of the following main functions:

- Ensure the flight safety of the mission during:
  - Captive flight
  - Release manoeuvre
  - Thrusted flight
- Manage the flight regarding GNC and mission timeline
- Manage communications:
  - Internal communications between launcher subsystems
  - External communications between launcher interfaces (PL, ground systems and carrier)
- Configuration and validation of the avionics with respect to a given mission, avionics missionization
- Components house-keeping, regarding equipment supervision (power management, thermal control, buses,…)

The integrated and modular avionics aim to maximize the launcher functional autonomy by reducing ground infrastructures and operations, as well as SW missionization to increase mission adaptability. The objectives for the avionics subsystem are the same as for the whole launch service:

- The reduction of mass and volume in order to increase capacities in the framework of nano-micro satellites
- Increase responsiveness to adapt to high-frequency rate launch service
- Improve overall RAMS in the ground and flight seeking the reduction of operations and autonomous avionics for long missions

The major axis of improvement and innovation proposed aiming those objectives are the following:

- Reduce Weight and volume
  - Unify communications, via a full-duplex communications bus covering the telemetry (sensors and supervision data interfaces) and control (actuators interfaces) functions
  - Reduce harness complexity and mass, using remote IO (input/output) components for each stage interfacing avionics equipment and avionics interfaces at each stage, enabling the inter-stage communications via a unique centralized point
  - Optimize battery use
  - Distribute and relocate as maximum as possible equipment to lower stages in order to reduce equivalent mass: the mass is proportional to the flight-time of the equipment, so the lower the mass is placed, the sooner it is released, and consequently less impact on mission efficiency

- Increase Responsiveness

  Define mission-configurable avionics SW, which is validated against a well-defined operational frame, allowing loading mission configuration for each campaign. As a result, the SW validation is optimized so reducing mission analysis and preparation time, always the configuration respects the operational frame. We obtain improvement in:
  - Increase SW missionization
  - Allow to use COTS and more flexible HW configuration and testing
  - Reducing ground segment facilities and mission submission operations

- Improve RAMS
  - HW/SW Architecture Modularity:

  An avionics architecture modular design guarantees validation, traceability and interchangeability between components, equipment, specifications and functions.

  A modular software system is composed of encapsulated parts, called modules, each with a specific function. In this type of system, any module can be switched by another module with the same interfaces without affecting the rest of the system. If a module is modified/updated, the whole system regains its validated status by only validating the new version of the module.

DOI: 10.13009/EUCASS2017-221

The modules have access to common services and interfaces independently. Modules have their own dedicated memory and time partitions to ensure module independency.
From a HW point of view, modularity allows the use of COTS making the avionics non-dependent of ad-hoc equipment, so obtaining a more maintainable and upgradable system, increasing total obsolescence.

- Increase SW functions:

The overall system RAMS is increased by balancing ground segment functions to launcher SW functions, mainly on Safety and GNC subsystems, taking advantage of airborne reference scenario:
  - In-flight initialization: avionics initialization and release consent sequence in nominal and non-nominal scenarios
  - in–flight Launcher autonomous safety and integrated vehicle health monitoring (IVHM)
  - In-flight navigation performance assessment

The main axes of actuation described above are assessed through unified modelling language tools (UML/SysML), which provide:

- Metamodeling capabilities to define abstract components:
  - Ports, Classes, Components, Interfaces, Data, Messages and Information Flows.
- Traceability, exhaustive through all phases of systems engineering (Figure 5):
  - Specifications, Functions, Architecture components and IV&V elements.

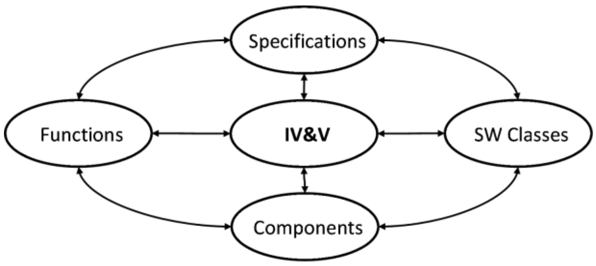


Figure 5: Systems engineering elements traceability

The tool traces the impact of an evolution in the design or a change in one specification through model components, improving maintainability, validation process and availability of the system. Moreover, the tool allows extracting and presenting the model information into documentation adapted to providers and subcontractors.

The preliminary architecture from Chapter 5.3 is based in the following additional design hypothesis:

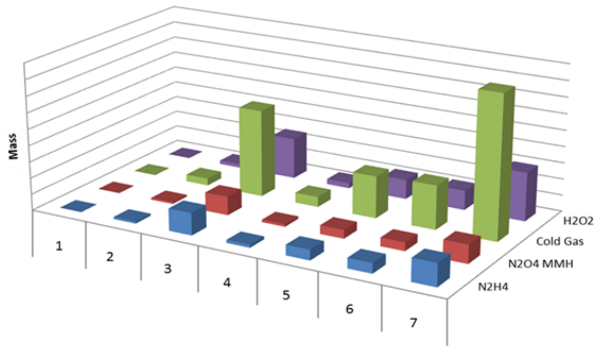
- The batteries are taken into account as part of the avionics. However, the electro-mechanical actuators power sources are considered part of the actuation control system, so an external avionics interface
- Safety subsystem components are considered redundant (batteries, equipment, buses and communications)
- Safety subsystem relies on devoted navigation instruments
- No auxiliary RACS system (Roll-Attitude control system)
- The harness, connectors and wiring are estimated based on launcher geometry and ECSS standards
- External communications considered are: telemetry and GNSS. Radar and neutralisation tele-command equipment are relocated on lower stages since their contribution to mission ends before OM mission flight.

A technology survey of potential candidates for equipment allows us to narrow the OM avionics mass to a maximum and minimum configuration. The cost, computational cost, power consumption and data flow budget criteria will affect the final configuration, and so mass and volume. Under those design considerations and hypothesis, the preliminary mass estimation for OM avionics is summed up in Chapter 5.3.

4. Trade-off Results

A detailed trade-off was carried out in order to assess the sensitivity of the total OM mass with respect to main attitude control and manoeuvring capabilities and according to various propulsion system propellants. Figure 6 provides a comparison, in term of total mass, between OM concepts characterised by different levels of orbit and attitude control capability and equipped with propulsion systems using four diverse propellant technologies: monopropellant hydrazine system, hypergolic nitrogen-tetroxide/monomethyl-hydrazine system, nitrogen cold gas system and monopropellant hydrogen peroxide system.

DOI: 10.13009/EUCASS2017-221



ID	Concept main functions description
1	OM 3-axis attitude control.
2	OM 3-axis attitude control, Collision Avoidance Manoeuver (CAM), LV orbit injection error correction.
3	OM 3-axis attitude control, CAM, LV orbit injection error correction, LV boosters flight attitude control.
4	OM 3-axis attitude control, OM spin up/down, CAM, LV orbit injection error correction, 1st/2nd LV stages flight attitude control.
5	OM 3-axis attitude control, OM spin up/down, CAM, LV orbit injection error correction, 1st/2nd LV stages flight attitude control, Circularisation manoeuver.
6	OM 3-axis attitude control, OM spin up/down, CAM, LV orbit injection error correction, 1st/2nd LV stages flight attitude control, Controlled Re-entry Manoeuver.
7	OM 3-axis attitude control, OM spin up/down, CAM, LV orbit injection error correction, 1st/2nd LV stages flight attitude control, Circularisation manoeuver, Controlled Re-entry Manoeuve.

Figure 6: Trade-off synthesis on OM mass Vs Functions Vs Propellant technology

According to this preliminary analysis, hydrogen peroxide was selected as reference propellant because considered the best compromise between performances and limitations induced by hydrazine based systems.

Compared to already flown systems using hydrazine, a system based on hydrogen peroxide will see a drop in delivered specific impulse. However hydrogen peroxide has higher density and lower freezing point proposing a less complex system. By utilizing the propellant non-toxic and clean properties, it has been shown that the system complexity both on-board and on ground is much reduced while maintaining a weight equal to a hydrazine system. In this way, the system performance will be competitive and have an attractive low cost.

Hydrogen peroxide has seen extensive use in a reaction control systems in many past manned and unmanned aircrafts, rockets and spacecraft's. A few examples are:

- Bell X-1 high-speed aircraft (1948-1958)
- Mercury spacecraft (1959-1963)
- Centaur upper stage as used on Atlas and Titan launch vehicles (1962-1983)
- Soyuz manned re-entry capsule (1966 – still in use)

It has then been supplanted in later vehicle design by hydrazine-based system for its higher energetic level. Nowadays, interests in hydrogen peroxide have increased again, notably due to the fact that hydrazine has been placed on the REACH list, for its green properties (non-toxic) and overall low cost and high availability.

In a second phase of the trade-off, various configurations of the propulsion system architecture were studied in order to trade between, overall propulsion system performances, attitude control capabilities, complexity of the system total mass and costs.

Three propulsion architectures considered are:

- Concept 1: it is equipped with a main propulsion system based on four H_2O_2 Hot Gas Thrusters (HGT) for main orbital manoeuvers and an auxiliary H_2O_2 Reaction Control System (2x3 or 2x2 Low Thrust HGT for fine 3axis or 2-axis attitude control).
- Concept 2: it is equipped with a main propulsion system based on four H_2O_2 high thrust HGT for main orbital manoeuver and coarse attitude control.
- Concept 3: it is equipped with a main propulsion system based on four H_2O_2 HGT for main orbital manoeuver and an auxiliary Cold Gas Reaction Control System (2x3 or 2x2 Low Thrust N_2 Thrusters for fine 3axis or 2-axis attitude control).

A weighted multicriteria analysis (Figure 7) was carried out on these concepts by considering two different End of Life (EoL) disposal strategies: controlled re-entry and uncontrolled re-entry after 25 years of natural decay.

Main results of the preliminary trade-off and design showed that to meet main performance requirements to ensure multi-mission flexibility, multi-payload capabilities and 3-axis attitude control as well as to meet constraints induced by regulations (mainly REACH and LOS), a very interesting solution is represented by a system equipped with four hydrogen peroxide HGT opportunely mounted in order to ensure full OM 3-axis controllability.

A final trade-off analysis has been carried out to find the best compromise between payload and mission flexibility, end of life disposal strategy and system mass. Four cases have been considered (initial orbit provided by the launch vehicle booster stages has a negative perigee at -50 km and the same inclination and apogee altitude of the target orbit):

- Case A1: Nominal target altitude (SSO 600 km) with controlled re-entry (50 km x 600 km)
- Case B1: Nominal target altitude (SSO 600 km) with uncontrolled re-entry in 25-years (500 km x 600 km)
- Case A2: Increased target altitude (SSO 800 km) with controlled-reentry (50 km x 800 km)

DOI: 10.13009/EUCASS2017-221

- Case B2: Increased target altitude with uncontrolled (SSO 800 km) re-entry in 25-years (500 km x 800 km)

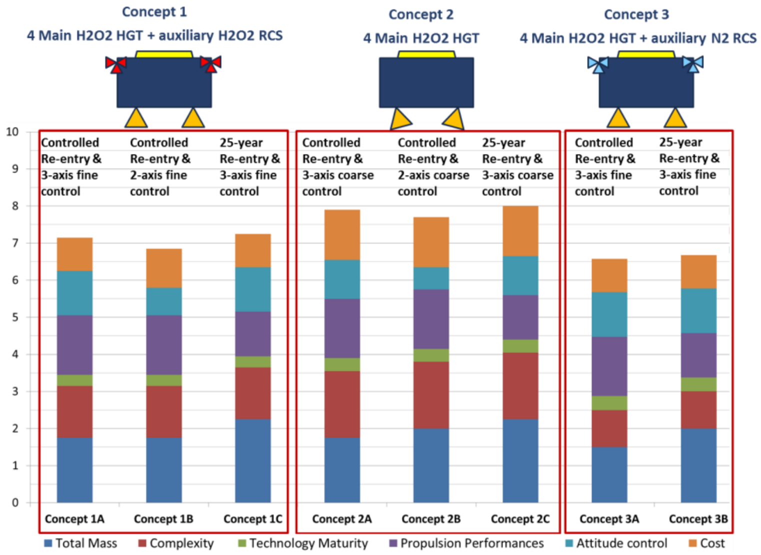


Figure 7: OM propulsion architecture, attitude control and EoL disposal strategy trade-off synthesis

According to the results (Table 3), case B2 is selected as reference concept because it represents a good compromise in terms of mass and mission flexibility. Actually, it is 6% heavier than A1 but it is able to provide either improved payload injection performances or controlled re-entry according to mission needs.

Table 3: Final trade-off preliminary results

		Case						Case			
		A1	B1	A2	B2			A1	B1	A2	B2
Mass Budget						DeltaV Budget					
Propellant Mass	kg	81	67	106	89						
AOCS Prop Mass	kg	20	19	20	20	AOCS ΔV	m/s	95	95	95	95
Circ Prop Mass	kg	48	45	67	62	Circ ΔV	m/s	243	243	309	309
EoL Prop Mass	kg	13	2	19	7	EoL ΔV	m/s	157	27	208	80
Dry Mass	kg	105	98	115	108	Total ΔV	m/s	495	365	612	484
Total Mass	kg	186	165	221	197	Available ΔV	m/s	521	371	640	501

5. Preliminary System Design

Following the trade-off activities, the reference concept B2 for the ALTAIR Orbiter has been selected and a preliminary design is in progress. This concept responds to payload mass and envelop needs expressed in Chapter 2 for a single Payload. Multiple payload configurations will be addressed later in the design phase.
ALTAIR Orbital Module current design consists in a 200 kg class spacecraft able to interface with Fairing, Payload Module (PLM) and Launch Vehicle 2nd Stage (Figure 8).
The core of the vehicle is a green H_2O_2 monopropellant propulsion and attitude control system accommodated in a lightweight structure. The hydrogen peroxide propulsion system main elements are an aluminium alloy Tank and four H_2O_2 200 N Hot Gas Thrusters. It ensures mission flexibility by providing multiple boosts capabilities for both orbital and attitude control.
OM primary structure is mainly based on composite technologies to reduce mass and ensure mechanical strength and stability. It consists in two main sections: a conical Payload Adapter and a cylindrical Central Structure. Main interfaces with fairing and booster sections are aluminium alloy monolithic rings. OM/PLM interface is designed to be compatible with the PAS 381S (15") Separation System, a COTS pyro-free low shock satellite separation mechanism developed by RUAG [17]. In order to reduce global height of the launch vehicle, part of the OM is inside the fairing and only the Central Structure will be exposed to external environment.
A thermal conductive aluminium alloy sandwich Base Panel closes the OM bottom part. It provides accommodation support for propulsion and attitude control system equipment as well as for avionics.
Actually, most of the electronic components of the launch vehicle will be centralised in the OM avionics suite. So far, a preliminary accommodation concept consists in four boxes to group on board computer and controllers, TC/TM and internal communication electronics, navigation equipment, including GNSS cards and inertial navigation

DOI: 10.13009/EUCASS2017-221

units, as well batteries. An umbilical connector hub is expected to ensure electrical interfaces with both launch vehicle boosters (during flight) and ground segment equipment (during ground operations). A second connector hub will provide OM/PLM electrical interface. Finally, TC/TM and GNSS antennas will be mounted on the cylindrical structure to provide ground segment communication capabilities during flight.

The reference concept selected does not expect a controlled re-entry. This means the overall design will be carried out by applying Design-for-Demise techniques to ensure a safe uncontrolled re-entry with a casualty risk (risk of damaging for people on ground) less than 10^{-4}. For example, employment of titanium alloy will be limited in order to reduce risks, while the tank will be developed using aluminium alloy which is expected to demise during atmospheric re-entry. Casualty risk analysis will be addressed later, nevertheless, a vehicle with less than 200 kg should be compliant with the LOS requirements.

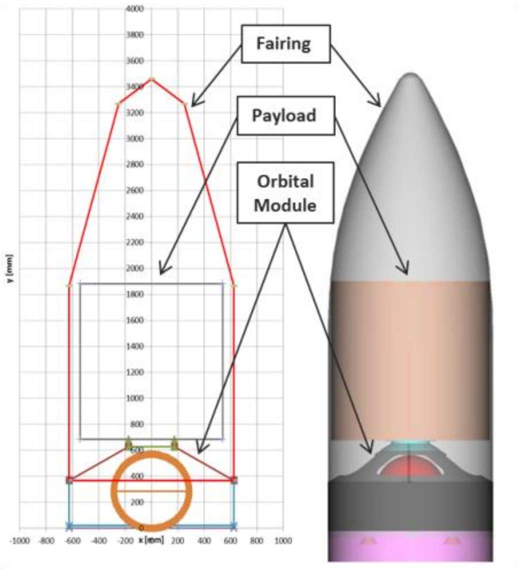


Figure 8: Current status of ALTAIR ORBITER Configuration.

Current high level technical specifications for the reference OM are listed in Table 4 while its overall architecture is shown in Figure 9. Related Product Breakdown Structure (PBS) is provided in Figure 10.

Table 4: Orbital Module Mass Budget and Overall Dimensions (Maximum allowed value)

Mass	Avionics Mass	kg	26
	Structure & TCS Mass	kg	36
	Propulsion/AOCS Hardware Dry Mass	kg	45
	Orbital Module Dry Mass	kg	107
	Propellant Mass	kg	90
	Total Orbital Module Mass	kg	197

Dimensions	Height	m	0.68
	Diameter	m	1.25

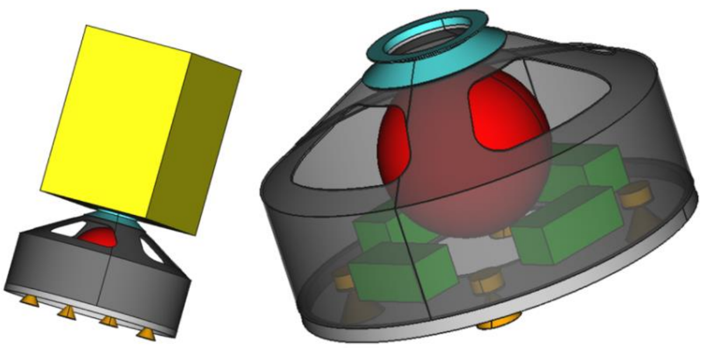


Figure 9: Preliminary ALTAIR Orbital Module CAD views with (left) and without payload (right).

DOI: 10.13009/EUCASS2017-221

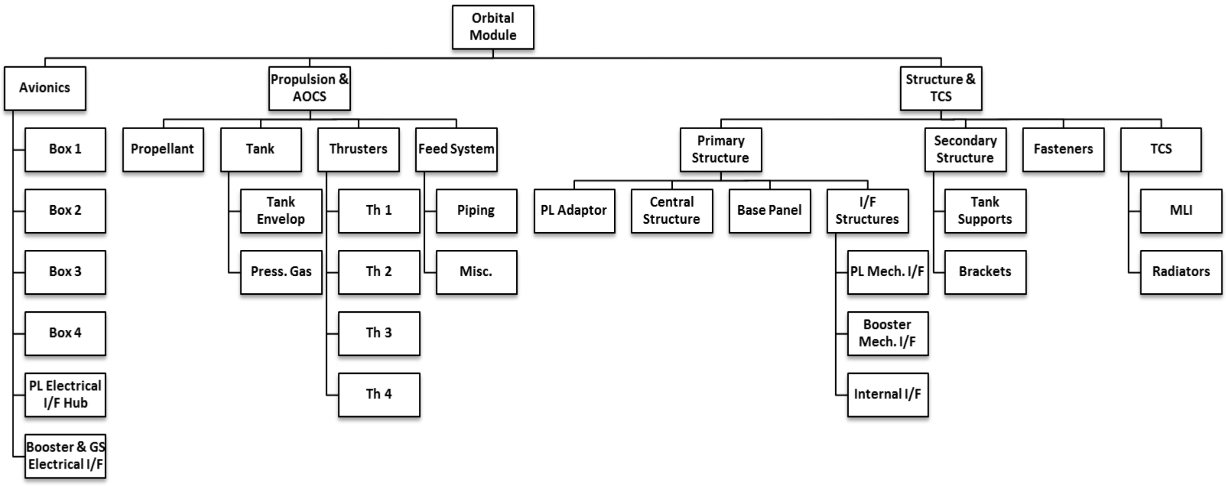


Figure 10: ALTAIR OM PBS

5.1 Propulsion System Design

The propulsion subsystem designed by Bertin Technologies is based on the blow down H_2O_2 monopropellant propulsion technology currently being developed by Nammo Raufoss AS [18] [19].
It consists in four 200 N Hot Gas Thrusters and a demisable aluminium Tank able to store up to 90 kg of hydrogen peroxide (Figure 11). In order to provide a coarse 3-axis control capability (yaw, pitch and roll), the thruster will be mounted with a double inclination with respect to the vertical axis of 5°.

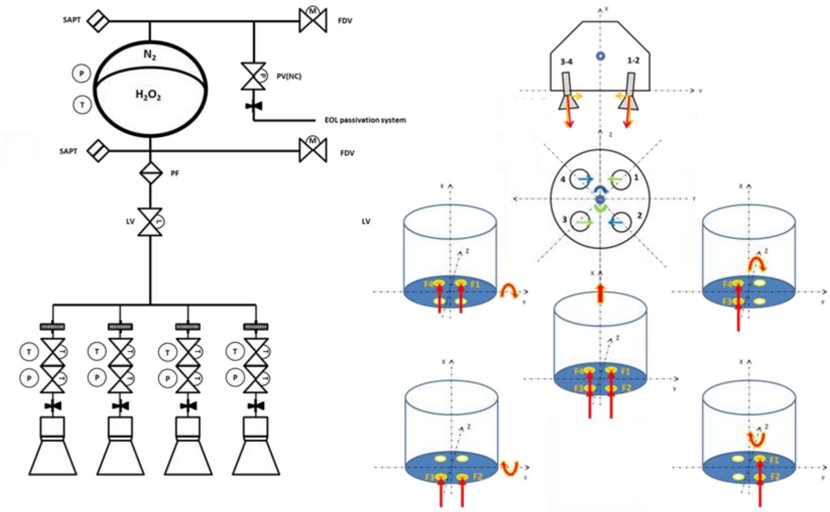


Figure 11: ALTAIR OM Propulsion System Scheme (left) and attitude control thruster mounting and activation strategy (right)

The choice of H_2O_2 as propellant, along with the other advantages listed in the trade-off, is also interesting in the context of ALTAIR as the main hybrid propulsion uses the same oxidizer. There are therefore commonalities between the two systems which would help in offering an overall low cost launch vehicle; the advantage of having a single fluid to be filled in the rocket is an example, along with increased modularity of the launch vehicle.
Nammo's technology is based on 87.5% per weight hydrogen peroxide, as it is commercially available in high quantity. It has on top a low intrinsic cost and a stable and predictable quality. For the same reasons, it has been chosen as oxidizer for Nammo's hybrid technology, which makes it an even better choice as commonalities between the two systems can be found.
The thrusters and the propellant tank (Figure12) from Nammo have been primarily developed in the context of Ariane 5 ME, with work starting as early as 2011. The development is now pursued for the future European launchers. The mission was, and still is, to propose a cost effective system capable of producing the level of

DOI: 10.13009/EUCASS2017-221

performance required for those launchers (thrust level, precision of the orbit insertion, controllability of the upper stage,…), using low-cost and green propellants.

Figure 12: Early design of Nammo's 200 N flight-weight thruster for Ariane 5 ME in a vacuum chamber test cell (left) and Nammo's propellant tanks for hydrogen peroxide (right) (Source: Nammo)

The thruster design is composed of two main components, the catalyst unit and the nozzle. The flow of incoming H_2O_2, controlled by the flow control valve, is decomposed through the catalyst pack into water vapour and gaseous oxygen at a temperature of about 700°C without the need of any ignition devices. The hot gases are then expanded through the optimized nozzle to develop the thrust. The development has been focused so far on a 200 N class thruster (nominal thrust) but the technology would be easily scalable for other thrust levels. The thruster is designed to operate in both pulse and steady state mode operations. During testing in vacuum chamber (2 mbar ambient pressure), a specific impulse above 163 s has been demonstrated, along with a cumulative total impulse of 187 kNs over more than 500 activations (on a single thruster) and a minimum impulse bit (MIB) of 3 Ns. The throttling range for this thruster is in the order of 5:1 with marginal loss of efficiency. With its flow control valve, the thruster has a mass of about 1.5 kg. However, the performance of the H_2O_2 thruster is indissociable from that of the flow control valve and both must be optimized according to the requirement of the ALTAIR OM mission.

The development of the propellant tank was initiated by Nammo in 2014. A prototype has since then been manufactured and cycled over 100 times to demonstrate the expulsion efficiency and the life cycle. Due to the environment in which the tank must operate (ie microgravity), the propellant has to be constantly separated from the pressuring gas. The design makes thus use of an elastomer diaphragm to separate the two fluids. Current Nammo's tank has a capacity of 74 kg with a blow-down ratio above 4. The materials for the tank domes and the diaphragm have been chosen to ensure a storability of the propellant of several months. With a design pressure of 70 bars and an outer diameter of 530 mm, Nammo's propellant tank has a mass of less than 15 kg.

In the frame of the ALTAIR OM development, Nammo's thrusters and tank design shall be modified and optimised in order to meet the system requirements. In particular, a larger tank will be needed in order to store the 90 kg of hydrogen peroxide expected. As a first estimation, this would lead to an overall tank mass of about 21 kg.

Table 5 provides a synthesis of the current design of the ALTAIR Orbital Module Propulsion system.

Table 5: ALTAIR OM Propulsion System main characteristics

Propellant	H_2O_2 87.5% per weight	90 kg	
Pressurisation	Blow Down	Ratio 4:1	
	N_2 Gas	2 kg	
	Design pressure	70 bar	
Hardware Components	PED Tank	21 kg	PED Tank (Al Alloy with membrane)
	4 x 200N HGT	4 x 1.5 kg (6 kg)	Including flow control valve
	Propellant Feed System	7 kg	Fill& Drain Valves (FDV), Latch Valves (LV), Normally Closed Valve (PV), Filters (PF), Pressure transducers (SAPT), Thermal and Pressure Sensors (T, P), piping and supports.
	Dry Mass	44.5 kg	Including 10% margin and 5 % residual fuel
	Dry/Wet mass ratio	50%	Including 10% margin and 5 % residual fuel
Performances	Vacuum nominal thrust	4 x 200 N (800 N)	Thrust range for a thruster 50 – 220 N
	Nozzle expansion ratio	75	
	Operative pressure	~ 20 bar	
	Isp	~ 160 s	
	MIB	< 3 Ns	

DOI: 10.13009/EUCASS2017-221

5.2 Structure Preliminary Design

So far, other structural element including the base panel, main mechanical interfaces, secondary structure (brackets and supports) as well as the thermal control sub-system (Multi-layer insulation system, radiators, etc.), have been preliminary sized and estimated by using the simplified approach provided by the MDO tool.
On the other hand, Table 6 and Table 7 present the design of the cylindrical airframe section and the conical payload adapter of the Orbital Module, which were obtained from the sub-system design optimization presented in Chapter 3.3.
The critical failure mode for the cylindrical section of the OM is global buckling of the shell, whereas local rib buckling in the vicinity of the larger diameter was identified as the governing failure mode for the anisogrid PLA. The cylindrical body has a predicted weight of approximately 6.7 kg, while the preliminary mass of the conical PLA structure is around 0.71 kg. This mass budget does not account for interface rings required for the connection of adjacent substructures, which will contribute to the overall structural mass of the OM.
The optimizations have shown that manufacturing constraints – in terms of minimum facesheet and core thickness – significantly restrict the structural efficiency of composite sandwich cylinders for the limit loads encountered in the OM, whereas the structural performance of anisogrid conical shells is negligibly affected by constraints on the minimum feasible rib width. For the cylindrical sandwich section, the lower bounds for both facesheet and core thickness were reached, limiting the areal density to approximately 2.5 kg/m². Sensitivity studies have shown that the influence of these manufacturing constraints on the minimum-weight design is very high at lower load levels and becomes less pronounced at higher loads, once the design variables vary from their lower boundaries. Similarly, for the anisogrid PLA, the lower bound on the rib width of 4 mm, which corresponds approximately to the minimum width of composite filaments that are wound around a rotating mandrel, was reached in the optimizations. However, comparative investigations with the theoretical minimum-weight designs have shown that manufacturing constraints in terms of minimum rib width and equal width of helical and circumferential ribs have minor impact on the minimum-weight design. Sensitivity studies have furthermore shown that the optimum is dependent on the number of circumferential ribs and can be reduced by increasing the rib count.

Table 6: Preliminary design of the cylindrical OM section made of composite sandwich construction

Design variable	Value
Core thickness	2.0 mm
Facesheet thickness	0.75 mm

Table 7: Preliminary design of the PLA made of composite anisogrid construction

Design variable	Value
# Circumferential ribs	6
# Geodesic ribs	74
Rib thickness	2.4 mm
Rib width	4.0 mm

Design variable	Value
Geodesic rib spacing at base section	104 mm
Circumferential rib spacing at top section	100 mm
Geodesic rib angle at base section	10.8°
Geodesic rib angle at top section	38.2°

Current status of the OM mechanical subsystems is summarised in Table 8:

The base panel accommodating main avionics and propulsion system equipment has been preliminary designed by considering a sandwich panel with aluminium alloy skins (ensuring thermal dissipation) and aluminium alloy honeycomb core. Its mass is estimated at 3.5 kg.

Table 8 provide a synthesis of the current structural mass budget for the ALTAIR OM.

Table 8: Preliminary OM Structural and thermal control subsystem mass budget

Primary Structure Mass	kg	24
Secondary Structure & TCS	kg	2.5
Fasteners	kg	4.0
OM Structure and TCS (with 10% Margin)	kg	33.5

The fairing structure of the ALTAIR launcher is yet to be sized and optimized. Regarding the global fairing architecture, a straight fairing, consisting of a cylindrical base section transitioning into an aerodynamically shaped nosecone will be used. System-level sensitivity studies have shown that the aerodynamic drag of the fairing is the

DOI: 10.13009/EUCASS2017-221

major factor influencing the payload performance of the launch vehicle, while the fairing mass has a minor impact on the payload capability. Owing to its low drag characteristics, a von-Karman profile was chosen as the most suitable nosecone shape for the ALTAIR launch vehicle, based on a multi-criteria trade-off process considering drag, mass, and manufacturability. An aerodynamically optimized fairing shape is expected to significantly improve the launcher performance and to outweigh the increased manufacturing complexity and cost. The base diameter of the fairing corresponds to that of the upper stage, while the determination of the nosecone length is subject of current CFD studies aiming to identify the minimum-drag fairing length for the maximum dynamic pressure phase. The preliminary faring profile is depicted inFigure13, in which the contours of the conical payload adapter and the payload envelope are indicated with black lines.

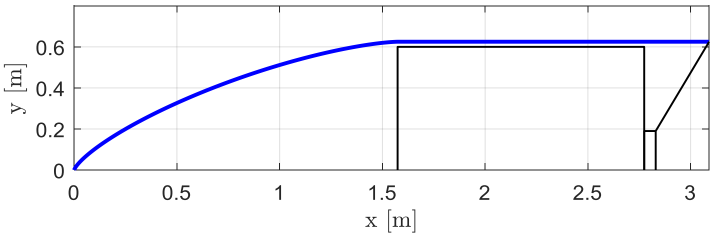


Figure 13: Preliminary fairing geometry

The preliminary fairing mass was estimated to be approximately 99 kg based on its wetted area, using a semi-empirical power law derived from the fairing mass and area of existing small satellite launch vehicles, disregarding the specific construction type and the mass of the individual fairing subsystems. Regarding the structural concept adopted in the fairing, composite sandwich construction with honeycomb core is considered, mainly owing to its high structural performance and technology maturity combined with good intrinsic acoustic attenuation and thermal insulation capabilities, which are expected to reduce the amount of acoustic blanketing and TPS required to satisfy the criteria for sound pressure level and temperature range within the payload bay.

5.3 Avionics Preliminary Architecture

According to assumption and methodology described in Chapter 3.4, the OM avionics mass estimation is less than 25 kg, including the following functional elements:

- On Board Computer (OBC) and controllers
- Power sources
- Navigation
- Telemetry
- Internal Communication
- Harness and wiring

The OM avionics model is described in Figure 14.

The ports represent the data flow and the power between equipment. Avionics equipment outside OM are represented in pink, the lower stage power source and the lower stage communications link. The avionics external interfaces, here the GNSS system, the actuators interfaces and the PL are in blue. Remark that the PL interface is linked to a PL hub managing multiple PL's power and data communications. Here the injection actuator is considered a separated interface, in order to segregate data/power and actuation. However, in future evolutions both equipment could be considered integrated.

DOI: 10.13009/EUCASS2017-221

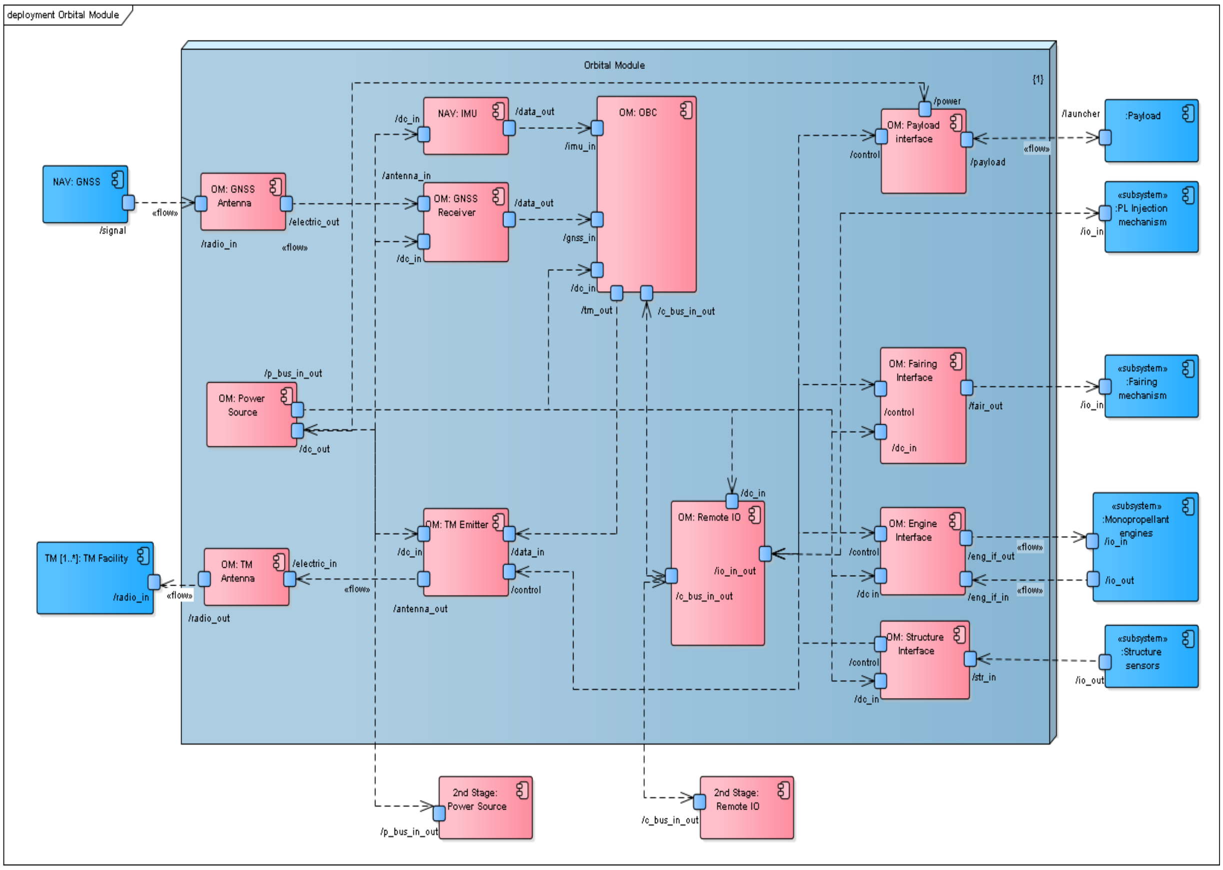


Figure 14: OM avionics preliminary architecture

6. Preliminary Mission Design

The preliminary mission profile has been simulated and optimised using the space trajectory optimization tool GMAT, developed by NASA (gravity model JGM-2, atmospheric model MSISE90) [20]. In order to verify compliance with 25 years re-entry time, the software STELA [21] developed by CNES has been used.
Thanks to the versatility of the OM design, various potential missions can be carried out. As an example, a mission targeting the injection of 150 kg of payload in a SSO 800 km is analysed.

The mission profile consists in the following phases (Figure15 and Figure 16):

- OM Orbital injection (provided by SLV the booster) : -50 km x 600 km, 97.8°
- 1st coast phase: OM achieves the first transfer orbit apogee at 600 km
- 1st boost: OM performs the first finite burn (800 N for 95 s) to achieve the second transfer orbit 800 km x 600 km (Propellant used: 48 kg)
- 2nd coast phase: OM achieves the second target orbit apogee at 800 km
- 2nd boost: OM performs the second finite burn (600 N for 25 s) to achieve the final target orbit SSO 800 km (Propellant used: 10 kg)
- Nominal mission operations: OM passive synchronisation with final payload injection orbit parameters and payload deployment (0.5 days)
- 3rd boost: OM performs the EoL Disposal finite burn (200 N for 115 s) to achieve the final decommissioning orbit 500 x 800 km (Propellant used: 15 kg)
- End of life: the OM is turned off on the decommissioning orbit which allows a natural decay and an uncontrolled re-entry after 25 years.

DOI: 10.13009/EUCASS2017-221

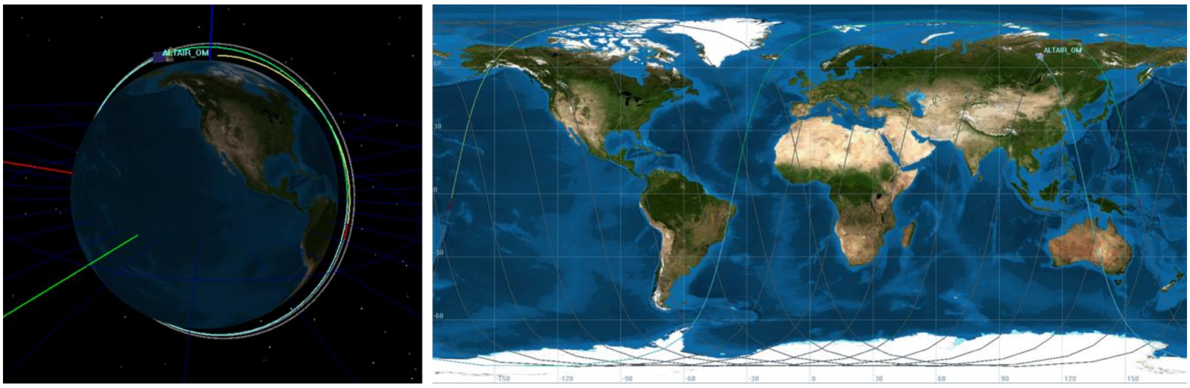


Figure 15: OM Mission Trajectory (GMAT results)

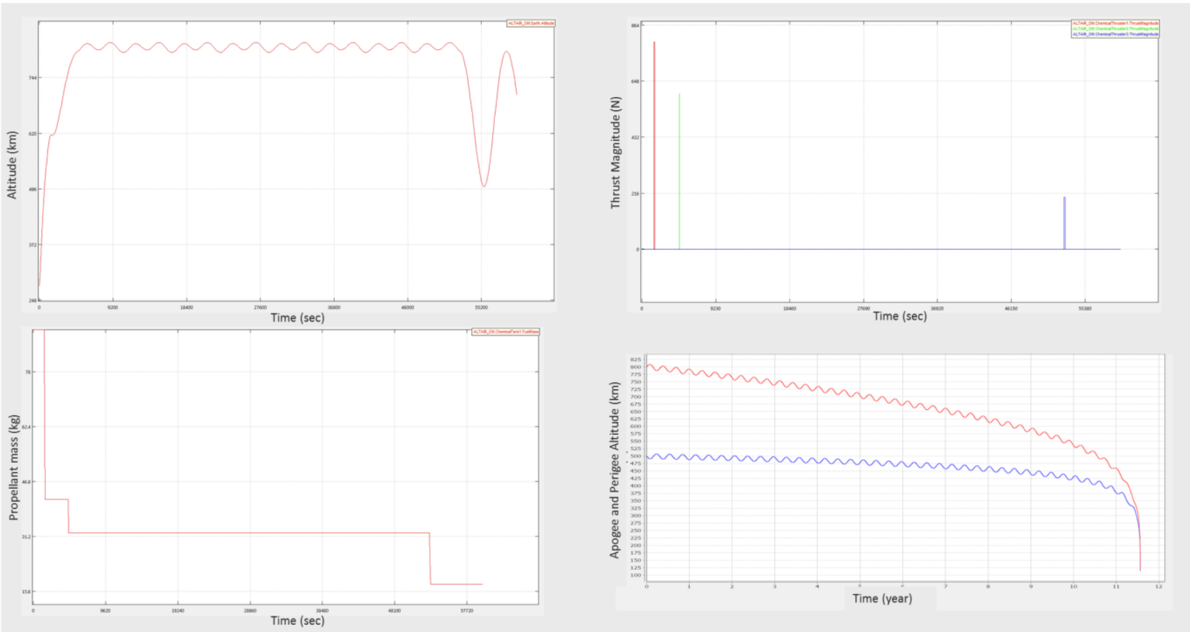


Figure 16: OM altitude, thrust magnitude, propellant mass consumption and apogee/perigee altitude evolution in function of time

It shall be noticed that part of the fuel not consumed in the simulation is dedicated to attitude and orbit control functions (~17 kg).
Further optimization runs are expected to consolidate the mission analysis as well as to optimise the strategy and propellant consumption.

7. Future Works

Future works relating with ALTAIR OM design consist in a consolidation of the mission and system design through a sub-system level collaborative engineering sustained activity. This design phase will allow optimising both propellant consumption and sizing of all the subsystems. Casualty risk assessment will be also addressed in order to validate the End of Life strategy.

8. Conclusions

ALTAIR is an international project carried out in the frame of the European Union's Horizon 2020 research innovation program. This project aims to prepare the development of an air-launch system dedicated to provide a cost effective and dedicated space access solution for microsatellites. In order to ensure a reliable and flexible orbit access capability, the design of a green propelled orbital module is carried out. This paper showed the methodology

DOI: 10.13009/EUCASS2017-221

applied and the progress in the design of such system. Current works led to the definition of a 200 kg class vehicle equipped with a monopropellant hydrogen peroxide propulsion system. A preliminary design of all the subsystem including the lightweight structure and the avionic has been proposed. This concept provides the capability to perform several orbital manoeuvers in order to ensure extended mission capabilities and to serve simple or multiple payloads. Furthermore, the concept is able to implement a decommissioning strategy to ensure a natural atmospheric reentry in less than 25 years.

Acknowledgments

ALTAIR project has received funding from the European Union’s Horizon 2020 research and innovation programme under grant agreement No 685963.
This paper was partially supported by Nammo Raufoss AS which provided information and technical expertise for the propulsion system design.

References

[1] N. Berend et al. ALTAIR (Air Launch Space Transportation System using an Automated Aircraft and an Innovatove Rocket) - General Overview & First Results". 67th International Astronautical Congress 2016.
[2] C. Dupont et al. ALTAIR - Design & Progress on the Space Launch Vehicle Desig. 7th European Conference for Aeronautics and Space Sciences - EUCASS 2017.
[3] C. Dupont, A. Tromba. ROXANE, Design of a new affordable European microsatellites launch system. The 4S Symposium 2016.
[4] W. J. Larson, J. R. Wertz. Space Mission Analysis and Design, Third Edition. Space Technology Library, Microcosm Press and Springer.
[5] C.D., Reese und C.W., Bert. Simplified Design Equations for Buckling of Axially Compressed Sandwich Cylinders with Orthotropic Facings and Core. Journal of Aircraft. 1969.
[6] N.F., Dow und B.W., Rosen. Structural Efficiency of Orthotropic Cylindrical Shells Subjected to Axial Compression. AIAA Journal. 1966.
[7] W.G., Heath. Sandwich Construction: Correlation and Extension of Existing Theory of Flat Panels Subjected to Lengthwise Compression. Aircraft Engineering Aerospace Technology. 1960.
[8] Stein, M. und Mayers, J. Compressive Buckling of Simply Supported Curved Plates and Cylinders of Sandwich Construction. s.l. : Technical Report, National Advisory Committee for Aeronautics (NACA), 1952.
[9] SP-8007, NASA. Buckling of Thin-Walled Circular Cylinders. s.l. : NASA Space Vehicle Design Criteria, 1968.
[10] Casado, V.M., et al., et al. Effect of Initial Geometrical Imperfections on the Buckling Load of Cylindrical Sandwich Shells Under Axial Compression. 2013.
[11] G., Cha und M.R., Schultz. Buckling Analysis of a Honeycomb-Core Composite Cylinder with Initial Geometric Imperfections. s.l. : NASA/TM–2013-217967, 2013.
[12] A.C., Orifici und C., Bisagni. Perturbation-based imperfection analysis for composite cylindrical shells buckling in compression. Composite Structures. 2013.
[13] A.F., Razin und V.V., Vasiliev. Development of Composite Anisogrid Spacecraft Attach Fitting. 2004.
[14] V.V., Vasiliev, et al., et al. s.l. : AIAA/CIRA. Anisogrid Conical Adapters for Commercial Space Application. 13th International Space Planes and Hypersonics Systems and Technologies, 2005.
[15] Hühne, C., et al., et al. Robust design of composite cylindrical shells under axial compression - Simulation and validation. Thin-Walled Structures. 2008.
[16] Schultz, M.R., et al., et al. Buckling design and imperfection sensitivity of sandwich composite launch vehicle shell structures. 2016.
[17] PAS 381S (15") Separation System, RUAG, technical sheet, www.ruag.com/space
[18] J. Rønningen et al. Development of a high-performance hydrogen peroxide monopropellant thruster for launcher applications. Space Propulsion 2016.
[19] L. Solli et al. Development of a propellant tank for hydrogen peroxide for launcher applications. Space Propulsion 2016.
[20] The GMAT Development Team. GMAT User Guide R2016a - General Mission Analysis Tool. NASA.
[21] STELA User’s Guide. CNES.