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HIGH EFFICIENCY REGIONAL AIRCRAFT CONCEPTUAL DESIGN AND ON-BOARD SYSTEMS PRELIMINARY STUDY

**Cesare Graziano^{*}, Matteo Cappo, Gesumino Fiore, Giacomo Montesoro,
Luca Boggero, Roberta Fusaro**

^{*} Politecnico di Torino
Department of Mechanical and Aerospace Engineering
Corso Duca degli Abruzzi, 24
10129, Torino, Italy
e-mail: cesare.graziano@studenti.polito.it

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Abstract. *A conceptual design of a new regional plane has been performed, investigating the application of the three lifting surfaces configuration and laminar fuselage on a larger aircraft. On-board systems have subsequently been sized and their installation validated in a CAD model. Finally, a flight simulation has been executed comparing the new design against a traditional regional aircraft, demonstrating its potential benefit on fuel consumption.*

1 INTRODUCTION

The attention of the civil aerospace industry in recent years has been focused on improving fuel efficiency, with the double goal of lowering airline operating costs and reducing pollution emissions.

In line with this aim of consumption reduction, the aerospace industry has looked for several solutions. There are extensive projects of research regarding innovative fuels, more efficient engines, better aerodynamics (e.g. new lifting surface configuration) and system architecture innovation (with the goal of reducing the power demand). These solutions were individually implemented on certain aircraft; Airbus A380 (efficient engines), Boeing B787 (innovative system architecture called “more electric”) and Piaggio P180 (three lifting surface configuration [1] [2]) are examples of that. By contrast, less attention has been devoted to the integration of these various solutions on the same aircraft, especially in the regional aircraft segment.

The present paper evaluates the possibility of integration of both aerodynamic and system innovative architecture through the study of the conceptual design for a new High Efficiency

Regional Aircraft (HERA). This work has arisen from a project of the university course “Integrated Aerospace System Design” held at Politecnico di Torino. The aim of the project was a preliminary sizing and installation of onboard systems within the conceptual design of a regional turboprop with an innovative architecture.

The first section of this paper reports the methodology adopted for the design, describing the workflow that was followed and highlighting its main steps. The second part illustrates the project requirements and the configuration adopted, concentrating on the description of the onboard systems (sizing and installation). In the second part close attention is also paid to the flight tests performed with a flight simulator. The last part discusses the results that have been obtained. The potential development of the present work and the feature that need to be improved are underlined within the text.

2 METHODOLOGY

In this section the work flow is described in all its steps, whose results will be more deeply examined in the following parts of this paper.

After the requirements definition, several aircraft architectures have been compared: a series of hand sketches were done for a better assessment of each solution, with its own PROs and CONs. Once the overall architecture was defined, a more detailed hand sketch has been done; the laminar fuselage has been adapted to the payload (i.e. the passengers) and the definitive fuselage layout and internal configuration has been decided.

In the second phase of the project, a preliminary on-board system sizing is achieved by using the computer aided system design software “ASTRID” [3]. Following the ASTRID procedure, the sizing procedure started with the definition of a typical mission profile for the aircraft, considering every phase of the flight including also eventual diversions. A calculation of the performances followed giving in output important data for the overall sizing of the aircraft (i.e. wing surface, required engines etc). Then, each on-board system has been sized; its power consumption has been estimated by using a set of equations, different for each type of system, integrated in the software. Finally an overall power budget and a weight and balance estimation for the whole aircraft is done: if ever a requisite had been compromised, a new solution would have been developed. Simultaneously a comprehensive 3D CAD model of the aircraft was created for a better evaluation of systems installation.

In the third phase of the project, the new designed aircraft has been modelled to be tested with a flight simulator. A typical mission has been flown with the HERA and with another regional aircraft of the same category by following the same in-route procedures. The results of the two simulations have finally been compared, validating the potential advantages of the HERA project.

3 THE HERA PROJECT

3.1 Design requirements

The HERA is a civil turboprop regional aircraft with a medium-short range.

Main requirements:

- Number of passenger: 90
- Extra payload: 2000 kg
- Maximum range at MTOW: 1500 km
- STOL
- High fuel efficiency

3.2 Aircraft configuration

The aircraft configuration and the systems have been conceived in order to satisfy these requirements. At this purpose a typical conceptual design approach has been followed taking into account both qualitative and quantitative evaluations [4].

The most challenging requirement from the configuration point of view concerns the need of high aerodynamic efficiency, in order to achieve a better fuel consumption. It has been satisfied choosing a non-common architecture for the aircraft, based on the Piaggio P-180 one. The configuration features a laminar flow fuselage and three lifting surfaces (high wing, canard and stabilizer), elements that allow to decrease the overall aircraft drag and increase the efficiency [1] [2].

The real effectiveness of this choice on such a large scale aircraft compared to the P180 has yet to be demonstrated, although, on grounds of the aerodynamic similarity principle, there are good reasons to believe in it. An in-depth research on this topic exceeds the purposes of this study, but could be a worthy starting point for future research.

3.2.1 Fuselage

To maintain a laminar boundary layer on fuselage skin as long as possible, its cross section diameter has to increase along the longitudinal axis in order to accelerate the flow and delay the boundary layer transition. Having circular diameter-increasing cross sections poses some constraints that prevent to adopt a traditional internal configuration with a unique planking level and a single aisle with a constant number of abreasts. Moreover, the nose part of the fuselage would be too small to host the cockpit and the center part would be too large and with an excessive void volume. So, the fuselage and the integration with the wings have been designed around the payload using a feedback approach to better use the internal volume:

- In order to shorten the fuselage, a variable number of abreasts has been chosen, with 4 abreasts in the front and aft part of the fuselage and 6 abreasts in the center part.

Passenger cabin roof is on the same level and features overhead luggage compartments on each side. The volume above the cabin roof hosts various system bays (e.g. electrical transformer-rectifier units, avionic equipment etc.);

- The cockpit has been located on a lowered deck in respect of the passenger cabin; the entrance is on the same level of the cockpit and the passenger cabin is reached through a low slope ramp;
- The main landing gear is retracted in a dedicated bay inside the fuselage instead of using external fairings, like on the ATR-42/72, which would spoil the aerodynamics of the fuselage shape. Moreover a retraction inside the engine fairing under the wing, like on the Dash Q-400, has been discarded in order to have a short landing gear strut and consequently a weight saving. (The retraction of the landing gear will be analyzed in detail in the last part of the paragraph.)

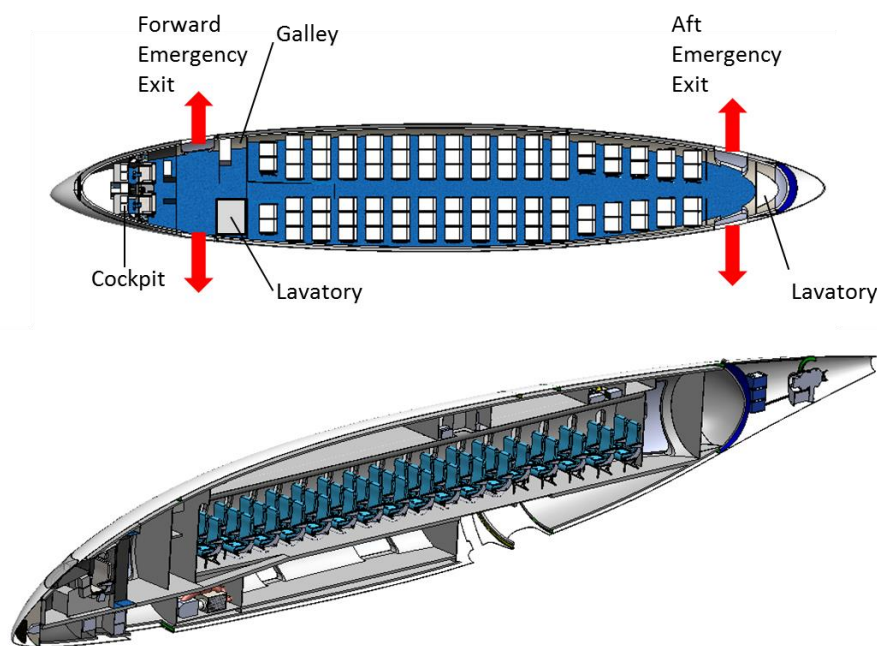


Figure 1: Internal configuration.

3.2.2 High wing layout and engines installation

The HERA features a high wing layout, chosen on the basis of multiple considerations:

- A high wing well fits with the choice of turboprop engines (pushing for the HERA), as it grants STOL performances to the aircraft: propellers can have a high diameter without the risk to strike the ground during take-off rotation. Moreover, having a larger diameter of the propellers with a low wing layout means having longer (and so heavier) landing gear struts, reason for which that configuration has been rejected;

- The high wing layout allows to introduce a non-containerizable cargo bay under the main deck between the nose wheel well and the main wheel well. In this way the internal volume is better used.
- Wing-fuselage junction is located above the passenger cabin roof, taking advantage of the large overhead space introduced by the laminar fuselage.

The pushing turboprop engines layout has been retained from P-180 although with a different purpose: in Piaggio's aircraft the pushing propellers allow mainly to have low noise level in cabin, while in the HERA project the main goal is to avoid a swirled and turbulent flow on the wing in the area next to the engines. In fact, with traditional tractor propellers a significant portion of the wing is invested by the highly turbulent flow downstream of the propeller and generates high drag; therefore, in order to avoid this phenomenon and reduce wing drag, pushing propellers have been selected.

As already said, the HERA is conceived to use available technologies and components in order to reduce the Time To Market (TTM). After the preliminary calculation of the performances with ASTRID, the selected engine is the Pratt & Whitney PW-150A, currently mounted on the DASH Q-400 and maximum rated for 5000 shp.

The main problem arose with the installation of the propulsion system: in fact the PW-150A features an air intake in the front part and the engine flow path is oriented according to the direction of flight; however using the engine in pushing configuration implies having the air intake in the aft section of the engine with the flow path oriented opposed to the direction of flight. Therefore there are two main solutions to install the engine:

1. Mount the engine with the flow path oriented according to the flight direction, with the air intake pointed towards the aircraft nose and the exhaust pointed to the aircraft tail. This would request a modification of the engine itself: the gearbox should be mounted in the aft part of the engine to mount a pushing propeller, but it would involve a complete rework of the exhaust section. Clearly these operations would cause time and costs to arise, so the solution has been discarded.
2. Mount the engine with its axis opposed to the flight direction, with the exhaust pointed towards the aircraft nose and the air intake pointed towards the aircraft tail. In this case only an accurate design of the air intake duct, exhaust ducts and engine fairing would be necessary without modifying the engine itself. The airflow must be deflected by 180° to enter the engine.

The second solution has been approved and examined in six different options, each one evaluated, using a scale from 1 (poor solution) to 5 (good solution), in terms of:

- Air intake effectiveness.
- Structural complexity.
- Spoiling effect on wing aerodynamics.

An overall score for each solution is computed simply by adding each single score; the highest overall score represents the best solution.

Due to the lack of technical details and drawings about the PW-150A, an approximate 3D CAD model has been built and its installation evaluated graphically.

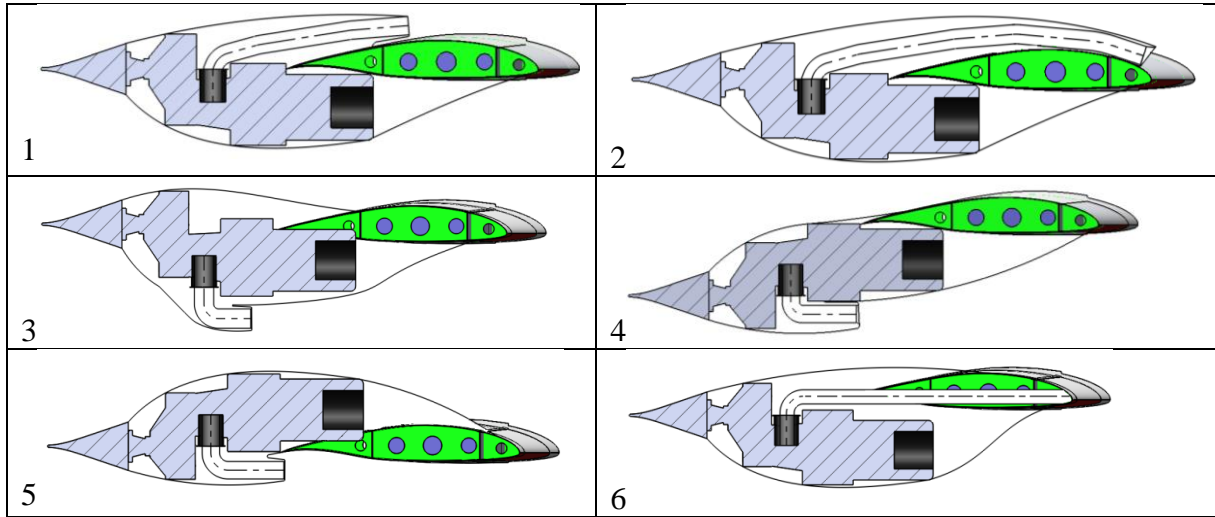


Figure 2: Possible engine installation solutions.

Option	Air intake effectiveness	Structural complexity	Spoiling effect	Overall score
1	1	4	4	9
2	2	4	2	7
3	5	4	5	14
4	5	4	5	14
5	1	3	1	5
6	3	1	4	9

Table 1: Trade-off for engine installation solutions.

1. Engine mounted below the wing with air intake in the upper part of the engine. PRO: it does not spoil the airflow on the upper surface of the wing (the one that provide 2/3 of the wing lift). CON: an air intake located on wing upper surface may result in wing wake also at low angle of attack and it could lead to an inflight shut down.
2. Engine mounted below the wing with air intake duct that stretches up to the leading edge. PRO: the air intake would not result in wing wake. CON: air intake fairing would spoil the laminar airflow on the wing in those sections; a long air intake duct would cause dynamic pressure losses.
3. Engine mounted below the wing with air intake in the lower part of the engine. PRO: the airflow on the wing is not spoiled. CON: none (in first approximation).

4. Engine mounted below the wing in an upside down manner. PRO: the airflow on the wing is not spoiled. CON: reduced clearance between propeller and ground during take-off rotation; risk of ground strike; rework of the engine lubrication system required.
5. Engine mounted above the wing in an upside down manner with air intake in the lower part. PRO: none respect to other options. CON: engine fairing would excessively spoil the airflow on wing upper surface; air intake may result in wing wake at negative angles of attack; rework of the engine lubrication system required.
6. Engine mounted below the wing with air intake duct inside the wing. PRO: it does not spoil the airflow on the wing. CON: the air intake duct breaks the structural continuity of wing spars; a long air intake duct would cause dynamic pressure losses.

Even if the trade-off considers equal options n.3 and n.4, the n.3 has been considered the best due to the risk of ground strike presented by the n.4.

3.2.3 Landing gear

HERA landing gear retains the ATR architecture: a nose steering landing gear with a single 2-wheels bogie and a main landing gear composed of two 2-wheels bogies. Because of the high attention paid to the aerodynamics of the aircraft, there are no external fairings attached to the fuselage: both nose and main landing gear retract in internal bays. Both gears use air-oil type shock struts to absorb impact on landing, vibrations and shock from movement of the airplane on the ground.

Nose landing gear has been designed to have the following important features:

- Extraction favored by the air flow and by the weight of the landing gear itself: in case of emergency the extraction can be performed without using the electro-hydraulic system.
- The hydraulic actuation is performed with a compact actuator which features different piston areas; the system has been conceived to make the actuator moving in the favorable direction during gear retraction. The retraction indeed is the most power-demanding phase for the actuator because it has to generate a torque to contrast both those generated by the aerodynamic drag and the weight of the landing gear.

Referring to Figure 3 it is possible to observe the nose gear with its main components: the shock strut (1 and 4, respectively inner and outer shock strut) has its trunnion on the main frame; the extraction-retraction mechanism is composed of a two drag struts linkage (5 and 6) commanded by a hydraulic actuator (7) and doors opening is performed with a linkage connected to the shock strut. Inner and outer shock struts are connect by a torsion link (2), element, that hinged to the steering collar (3), allows the nose gear to steer.

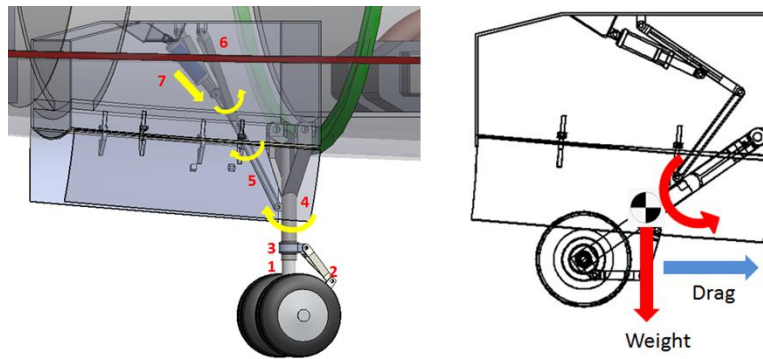


Figure 3: Nose landing gear overview.

While the retraction of the nose landing gear features a traditional architecture, the retraction of the main landing gear is much more constrained by fuselage shape and its internal structure.

The longitudinal position of the main landing gear must be located rearward of the maximum excursion of the aircraft center of gravity; because of this, the retracted gear must have a small lateral encumbrance to fit inside the fuselage without using external fairings.

The problem indeed is that the fuselage diameter narrows near the c.g. longitudinal position; so a non-common retraction solution has to be designed:

- Main landing gear shock struts are hinged to the main airframe at the same longitudinal position, but while the left gear is hinged on the rear part of the frame, the right gear trunnion is located on the front part of the frame.
- This fact allows to retract the landing gear staggered in respect of each other; the right shock strut turns in a plane tilted rearward by 23° while the right one turns in a plane tilted forward by 18° .
- In this way, when the main landing gear is retracted, left and right gear are longitudinally aligned, assuring the less possible lateral encumber.

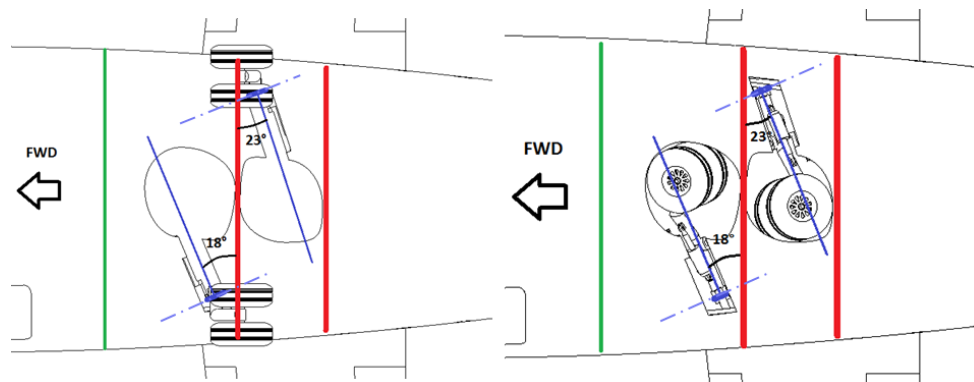


Figure 4: Main landing gear retraction angles.

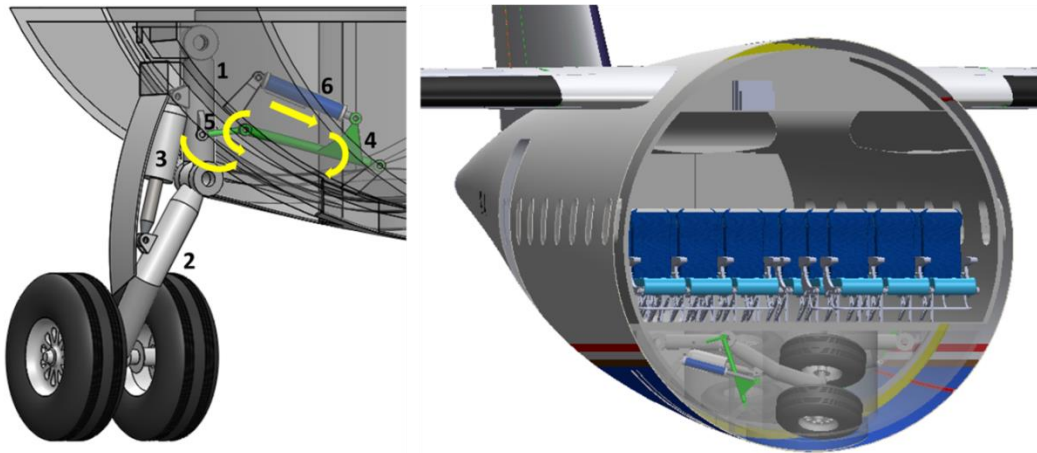


Figure 5: Main landing gear overview.

A particular linkage system performs the retraction of the main gear (Figure 5): a hydraulic actuator (6) hinged to the structure pushes (towards the favorable direction during the retraction i.e. the most power demanding phase) on the link (4 and 5), whose rotations make the main shock strut (1 and 2) turn towards the landing gear bay. When the landing gear is completely extended, the linkage 4-5 rotates slightly beyond the extended position, locking the gear down.

As it is possible to observe, the main landing gear layout does not feature a traditional shock strut composed by an inner and an outer part (like the nose landing gear) but it features a hinge between the two parts of the strut (1 and 2) and an external shock absorber like the ones of the F-18 fighter.

Furthermore each main gear wheel is fitted with disc-type hydraulic brakes modulated by an antiskid system.

3.3 Onboard systems

Starting from the general idea of an advanced aircraft in its category, it made sense trying to use system philosophies as innovative as the configuration itself; however it has to be noted that the project has been intended to have short Time-To-Market, therefore requiring already tested technologies.

In the last years, either in the military and civilian aviation, one of the biggest trends followed is moving toward a more (or all) electric configuration. This system philosophy implies reducing or eliminating many subsystems, especially the hydraulic and pneumatic ones, and lets their functions be done by the electrical system [5]. In this way it is possible to increase fuel efficiency and safety, to reduce system complexity and decrease operating costs.

Major applied examples of this design are Boeing 787 (pneumatic system replaced by electric), Airbus A350 (electric actuation) and Lockheed Martin F35 fighter (hydraulic and pneumatic eliminated – all electric aircraft).

In our case it has been decided to implement an all-electric architecture, eliminating the hydraulic and pneumatic systems altogether.

This is done for two principal reasons:

- The pneumatic system with its compressor bleed is a major source of power losses, especially in a turboprop engine because of its limited airflow.
- The hydraulic system adds a great amount of weight because of its lines running inside the aircraft and is a substantial complication in operation and maintenance.

The electric system is therefore the only power distribution system across the vehicle, powering all other subsystem, including flight controls. This is a substantial innovation for a civil aircraft, especially for a regional, since even the B787 retains the hydraulic system for primary flight controls and other functions.

3.3.1 Flight Control System

In accordance with the general idea of an all-electric aircraft, the flight control system has been designed to be electrically actuated, in regard to primary and secondary flight controls either.

As a general rule, Electro-Hydro-Static Actuators (EHSA or simply EHA) have been used in primary controls, while traditional electro-mechanical actuators (EMA) have been used in the secondary ones. All electric actuators (EHA and EMA) are powered by 270 V DC.

While secondary controls have been electrically powered for much time, electrical actuation of primary controls is a substantial innovation in the aerospace field; in fact, it is currently only fully employed on the Lockheed F-35 and used in conjunction with traditional actuations on the Airbus A380 and A350.

As stated above, since exact design of control surfaces in respect of flight dynamics is beyond the aim of this project, control surfaces extension and excursion have been chosen based on similar aircrafts.

Primary Flight Controls (elevator, rudder and ailerons)

Two or three EHAs for each surface (right and left) are employed, configured in hot redundancy; in normal operation both devices work at moving the elevator, while, if one fails, the remaining one is able to move the surface on its own.

Secondary Flight Controls (spoiler, wing flaps/slats and canard flaps)

Spoiler Spoilers deserves a special mention because, while are considered secondary flight controls, they are intended to be positively used as primary flight controls.

Asymmetrical deployment is used to increase lateral controllability, while symmetric extension allows decreasing lift and increasing drag, either in flight or during landing.

For this reasons they are actuated by EHAs as ailerons, elevator and rudder.

There are six spoilers spanwise the wings, three for each side, each one powered by one EHA; there is no redundancy for each spoiler since the loss of one of them has not been assessed critical for flight safety.

Leading edge high lift devices (slats) A slotted type slat is used; three surfaces are installed for each side (inboard, mid-wing and outboard). Slats are powered by two rotary electric motors, located in the fuselage; motors are geared to work in “torque-sum”, allowing, in case of failure, one motor to power the whole system at a reduced speed. Motion is transmitted to aerodynamic surfaces via a shaft running inside the wing leading edge and is transformed from rotational to linear by a rack and pinion system

Trailing edge high lift devices (flaps) A slotted type is used; two surfaces are installed for each side (inboard and outboard of the engine). Flaps are powered in a similar fashion to slats, but the rack and pinion system is substituted by a ball-screw one.

Canard flap An additional movable plain surface is installed on the canard trailing edge. While the canard is fixed and not used as a control device, the needs comes from the aircraft longitudinal equilibrium when flaps and slats are used; due to the particular three lifting surfaces HERA’s configuration, wing high lift devices deployment would determine a pitch moment that would require an excessive elevator deflection to be equilibrated; the canard flap, deployed automatically when flaps and slats are commanded, reduces the pitch moment required to the elevator. The solution has been derived from the Piaggio P180.

Canard flaps are powered in a similar fashion to wing flaps; no redundancy is provided since canard flap failure can be overcome by reduced wing flap deployment and increased elevator use.

	Surfaces	Actuators	Type	P tot	Weight (single act.)	Tot weight
Ailerons	2	2/surface	EHA	3874 W	14,16 kg	56,64 kg
Flaps	4	2	EMA	1275,50 W	55,01 kg	120,6 kg
Slats	6	2	EMA	102 W	5,48 kg	10,96 kg
Rudder	1	3	EHA	6314,40 W	38,78 kg	116,34 kg
Spoilers	6	1/surface	EHA	4335 W	15,68 kg	94,16 kg
Elevators	2	2/surface	EHA	4612 W	16,59 kg	66,36 kg
Canard flaps	2	1/surface	EMA	116,8 W	2,32 kg	5 kg
			Total	20,6 kW	Total	459 kg

Table 2: Control surfaces, actuator number, type, weight and power required.

Power requirement for each actuator has been computed using ASTRID software. Maximum hinge moment has been estimated from surface extension and deflection. It is worth noting

that flight controls have different maximum allowable deflection in relation with airspeed; for example, elevators can be fully deflected only up to the design maneuvering speed and spoilers have an in-flight detent. For each surfaces the maximum hinge moment has been selected. Actuation time has been imposed based on figures derived from similar aircraft. From actuation time and maximum hinge moment, the maximum power requirement has been calculated for each surface and a suitable actuator selected.

3.3.2 Fuel system

The fuel system has been based upon a traditional architecture for a short-medium haul aircraft. Four integrated wing fuel tanks (two inboard tanks and two mid wing tank) host 3600 liters of JET A-1, the fuel necessary for the mission and the reserves . Although a wide volume is present inside the center wing box, fuel tanks have been located more outboard in order to mitigate the structural bending moment due to the lifting force on the wings. For the same reason, the outboard wing tanks are thought to be kept full as long as possible, using at first the fuel in the inboard tanks.

Electrical boost pumps (115 V AC) feed the Fuel Control Unit (FCU) of each engine: each fuel tank contains two boost pump in cold redundancy (one active and one in stand-by); in case of failure only one of the eight boost pumps must be able to feed both the engines. There is also a DC alimented boost pump to perform the APU start-up.

In order to save weight there are no dedicated transfer pumps; fuel can be moved through tanks using a fuel bleed along the feeding line, with boost pumps being driven faster when the transfer is performed. A set of controlled and non-return valves allow to perform the best fuel transfer possible in order to avoid excessive pressure drops in the feeding line.

The system features a refuel station near the left wing leading edge with a fuel receptacle manifold: a pressure refueling can be performed in 7 minutes with a fuel pressure of 1,6 bar.

In pumps selection, it has been considered that a boost pump has to give the fuel only with enough pressure to overcome the pressure drops along the line; pipe line pressure drops have been estimated using ASTRID and considering the location of each pump in respect of the FCUs of both engines (FCU inlet fuel pressure: $p_{\min} = 1,2$ bar, $p_{\max} = 2,5$ bar).

The power budget for this system has been evaluated considering the fuel request in each phase of the mission in function of the throttle percentage and relative engine SFC; maximum required power by electrical pumps is about 1800 W during the take-off phase.

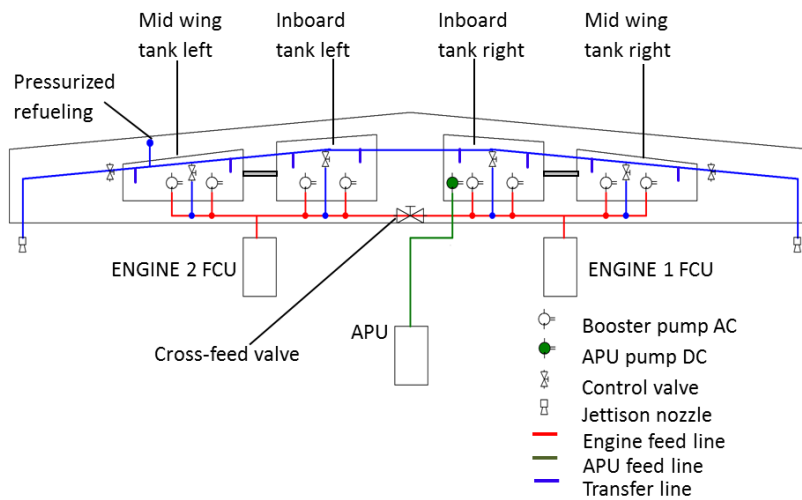


Figure 6: Fuel system scheme.

3.3.3 Anti-Ice System

The anti-ice/de-ice system has been designed to be an electric one, according to the idea of an all-electric aircraft. The protected surfaces are the leading edges of each lift surface, of air intakes, antennas and sensors. The surfaces are protected using Joule heating run by 230 volt AC VF, the voltage natively generated by the alternators. On each surface there are two types of electro-thermal protection:

- An extended area protected by cyclical current applications with de-icing effect.
- A limited area protected by continuous current application with anti-ice effect.

Their surfaces define the power required by the system and therefore an optimization has been done in order to reduce required power.

The power needed has been calculated considering the power density for each type of surface with the aid of ASTRID and a power budget has been obtained.

Small users such as antennas and sensors have been protected by the continuous type only; to estimate the power required it has been considered the total surface for each type of small users and this value has been multiplied by the number of redundancy of them.

The cruise altitude of HERA has been chosen to be 7000 m, for this reason the system has been considered active during all phases of flight plan. The required power for this system has been calculated to be 117.2 kW.

3.3.4 Environmental Control System

The Environmental Control System (ECS) design has been based upon an estimation of thermal load inside the aircraft with the aid given by ASTRID. The used model refers to:

- Thermal gradient between external and internal of the vehicle, considering the wet surface (dimension and material) and the altitude.
- Heat generated by sun radiation, through glass surfaces.
- Heat generated by crew and passengers' metabolism.
- Heat generated by on board systems.

Concerning the pneumatic system, a bleedless architecture has been chosen, because bleeding air from propulsion system brings to a power reduction and therefore a decrease in efficiency. In order to estimate dedicated compressors and motors power, a range of temperatures in which passengers comfort is guaranteed has been considered and then power has been estimated for each possible altitude referring to flight profile. The most critical value defined the needed power.

The architecture of the system consists of two compressors with a motor for each one, two air intakes (one for each compressor) and two air cycle 3-wheels CAUs, sub-freezing type; the chosen configuration features a 50% recirculation, reducing fuel consumption. The power is given by the two compressors that work using 50% of their power (each compressor with its motor has been designed to generate the whole power needed); in this way the system will continue pressurizing the cabin even if one of the motors/compressors and one of the CAUs fail simultaneously. Compressors are powered by 270 V DC and are rated to deliver 44,42 kW in normal conditions, increasing to 88,84 if one fails.

The critical condition used in designing the ECS is the subsonic cruise during a cold night with only 2 pilots on board; in this case the power required by compressors is about 88 kW, which is driven by two motors delivering 55.52 kW of electric power each, accounting for mechanical losses. In case of failure, a single motor is able to deliver the required 111 kW of total power.

3.3.5 Avionic System

In the avionic system design, a functional analysis method has been employed [6]. The functional analysis is a top-down approach, which starts from a few high-level avionic functions (namely Flight Control, Navigation and Communication) and proceeds dividing them in low-level sub functions until basic avionic needs are reached. This method enables to assess which avionic equipment is effectively needed and helps in designing the whole system.

The most modern trend in airliners is the Modular Integrated Avionic, which goes beyond the traditional concept of one box for each device. Although this concept is extremely promising, few airplanes already use it and the whole avionic system would have to be designed from sketch, surely increasing costs and development time. With reduced TTM as a goal, a traditional architecture has been chosen, though it could be substituted with the modern one.

Regarding the cockpit configuration, almost all modern airliners are equipped with the so-called “glass cockpit”, since it provides a more rational, integrated, flexible and comprehensive human-machine interface, reducing crew workload and increasing piloting safety. Therefore also the HERA features an electronic cockpit architecture, designed in a similar way to that present on the B787.

Avionic devices have been selected from the available ones already in commerce: this would speed up the production process, although it has to be noted that some of the selected avionic should be modified or integrated in order to work.

The entire avionic system is based on data-bus architecture. Although there are two data-buses connecting avionic devices with each other, the system has been designed to work with just one in case of failure.

Most of the avionic devices are physically placed in two avionic bays, one under the cockpit and the other near the center of gravity, above the passenger cabin. The separated places are required because some devices (like the Inertial Navigation System) must be installed near the center of gravity; this solution provides also an enhanced safety margin, because redundant avionic systems are not in the same place, protecting them from catastrophic events (as fires). Both places are designed to be easily accessible, facilitating maintenance.

3.3.6 Electric system

The electric system has been designed to be powered by 5 starter-generators (two for each engine and one for the Auxiliary Power Unit). Although aircraft systems needs four different voltages, current is generated at 230 V AC VF only and is subsequently transformed; in this way it has been possible to achieve an overall weight reduction. The system scheme is represented in the figure below.

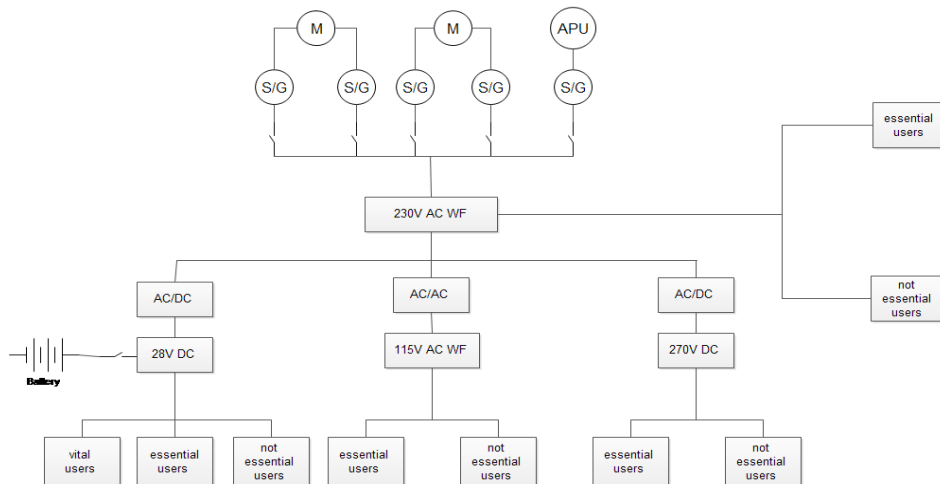
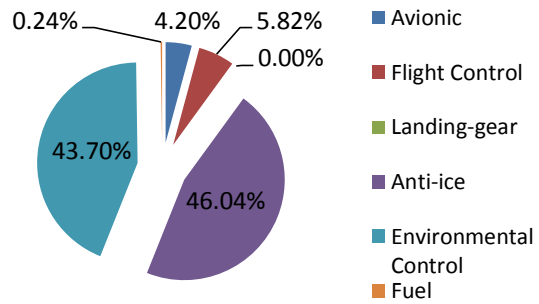


Figure 7: HERA electrical system scheme.

The design point for the electric system load is the subsonic cruise, when the power consumption is highest; the estimation of required power during this phase is showed in Table 3 while the power subdivision in reported in Table 4.

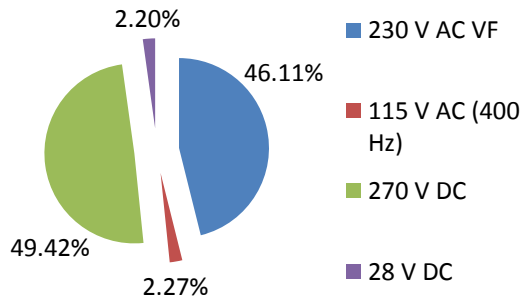
System	Power [kW]
Avionic	10,70
Flight Control	14,80
Landing-gear	0
Anti-ice	117,20
Environmental Control	111,25
Fuel	0,61
Total	254,57

Table 3: System power consumption.



Voltage	Power [kW]
230 V AC VF	117,40
115 V AC (400 Hz)	5,78
270 V DC	125,85
28 V DC	5,60

Table 4: Power subdivision.



The high power share of the 270 VDC electric line has been chosen due to the low weight, high efficiency and reliability of ATRU (Auto-Transformer Rectifier Unit), also used on the Boeing 787 Dreamliner.

Generators can supply up to 130 kVA each and they are heavily redundant; in fact two generators can power the whole aircraft. This allows for three random failures with all engines running and one failure with one engine out without any loss in functionality. A 28 VDC battery pack is installed in order to power the vital users in emergency condition, in ground handling and for APU starting. The chosen battery type is lithium-polymer with a specific energy of 200 Wh/kg and an available discharge time of 45 minutes. In order to obtain a mass estimation of generators, transformers, converters and batteries a relation between power and weight, considering the voltage, has been used [3]. The overall generators weight estimation is about 383 kg, while for converters and transformers the result is about 44 kg.

With a such designed systems configuration, the secondary power required by the aircraft is less than 5% of the maximum primary power given by the engines (254,63 kW on 7457 kW that is 3,41%); this is a result that follows the recent trend of reducing the impact of secondary power on the engines [7].

3.4 CAD model and overall data

During the design process, a comprehensive CAD model has been created.

The model features the aerodynamic surfaces, external fuselage, principal structural elements, interiors, engines and propellers.

CAD has helped in choosing and assessing on-board systems physical location, selecting cabin and cockpit configuration and finally evaluating weights inertial characteristics.



Figure 8: HERA CAD model.

Weight and dimensions	
Length	28,01 m
Wing span	27,89 m
Wing area	62,36 m ²
Passengers	90 (100 kg each with hand luggage)
Cargo	2000 kg
Operating Empty Weight	10091 kg
Maximum Take Off Weight	24020 kg
Performances	
Engines max power	5000 shp (2x)
Range (MTOW)	1500 Km (810 Nm)
Maximum speed (23000 ft)	650 Km/h (351 KTAS – 250 KIAS)
Ceiling (MTOW)	9750 m (32000 ft)
Take-off distance (MTOW)	1112 m (3648 ft)
Landing distance (MLW)	914 m (2998 ft)

Table 5: HERA general technical specifications

3.5 Flight Simulation

As a final validation for the whole work, a flight simulation has been performed.

The simulation has been run on a flight simulator based on the blade element theory, which allows to compute aerodynamic forces acting on the aircraft. In this way it is possible to test a new aircraft without real flight tests, wind tunnel tests or complete CFD, although with limited accuracy.

A simulated model of the HERA has been created based on the developed CAD, complete of its internal systems.

Several simulated flight tests have been done and eventually a comparative test with an ATR 72 has been performed.

The model created is based on the ATR 72-500 and its performances have been tuned to match the real ATR ones, derived from available data.

The flight test consisted of a typical regional route from Turin “S. Pertini” Airport (LIMF) to Munich “F. J. Strauss” Airport (EDDM), using a real word route flown by a known regional airline (complete of instrumental procedures). The target was flying the route with the lowest possible fuel consumption, regardless of time required. The total distance, comprehensive of procedures, was 711 km.

The route was flown by HERA and ATR either, respecting the same ATC constraints during departure and arrival. However each aircraft was allowed to climb, cruise and descent at their optimum configuration for fuel saving, reported in the table below.

	Climb		Cruise		Descent	
	ATR	HERA	ATR	HERA	ATR	HERA
Airspeed [KIAS]	135	150	175	170	175	170
Airspeed [KTAS]	-	-	237 – M 0.39	242 – M 0.40	-	-
Power	Max Clb	Max Clb	-	-	Flt Idle	Flt Idle
Altitude [ft]	-	-	20000	23000	-	-

Table 6: HERA and ATR fuel saving flight profiles.

Both aircraft started from gate with their tanks full and payload had been added until MTOW was reached. After landing and taxing to the arrival gates, the remaining fuel quantity was annotated. Since each aircraft has its own different tank capacity and in a typical flight only the needed fuel is loaded, their payload was increased by the unneeded flight quantity. The needed fuel has been computed taking in account the actual fuel burnt in the tests multiplied by a factor of 1.40, which is a rough estimate of the fuel required for diversion, holding and reserves for this specific route.

	HERA	ATR 72-500
MTOW	24099 kg	22500 kg
OEW	10091 kg	13847 kg
Fuel Initial	2307 kg	5000 kg
Fuel Final	1100 kg	4004 kg
Fuel Used / Needed	1207 kg / 1690 kg	996 kg / 1394 kg
Payload (w/ tanks full)	11701 kg	3653 kg
Payload (w/ fuel needed)	12318 kg	7259 kg
Flight time	1h 57'	1h 57'

Table 7: Flight simulation testing results.

In order to perform an efficient comparison, results have been normalized in respect of valuable variables (JET A-1 specific weight assumed 0.8 kg/l) and the following Figures of Merit have been considered.

Name	Unit	HERA	ATR 72-500	Var
Payload-Range Spec. Fuel Cons.	Kg/ton*100 km	13.78	19.30	- 28%
Passenger-Range Spec. Fuel Cons.	l/pass*100 km	2.36	2.50	- 5,6%
Payload-Range-OEW Spec. Fuel Cons.	Kg/ton*100 km*ton	1.366	1.394	-2 %

Table 8: Performance comparison.

From the Payload-Range and Passenger-Range consumptions ($Fuel/(Payload \cdot range)$) it is possible to assess that the HERA consumes less fuel for a given route in respect of its payload or passengers.

The Payload-Range-OEW ($Fuel/(Payload \cdot range \cdot OEW)$) is a very important result for this research; since the HERA OEW is less than the ATR one, it would have been possible that fuel savings derived from the minor structural weight instead from the innovative architecture. Dividing the Payload-Range consumption by the OEW of each aircraft, the HERA performs better than the ATR; in other words, this indicates that it would not be possible to obtain the same result simply by trying to reduce OEW of the ATR. In fact, this could be a proof of the better performance of the three lifting surfaces and laminar fuselage architecture. In regard to this, the variation seems quite small, but it has to be said that the simulator model of the HERA has been created in a very conservative way. In particular, it was not possible to find the BSFC map (Brake Specific Fuel Consumption) of the PW150 and a conservative estimation has been needed; in addition, the software used was not able to effectively calculate the fuselage lift which, though small, is significant for the HERA fuselage shape. It appears safe to assume that the real results could be better than the simulation ones.

As a final remark, however, it must be noted that this type of simulation has limited accuracy and, as said earlier, much data has required an approximate estimation; therefore these results have to be taken just as an indicative starting point for further in-depth studies.

4 CONCLUSION

A new regional aircraft has been designed in a conceptual form, integrated with on-board systems and simulator tested.

The first part of the work highlighted that it is feasible to apply the three lifting surfaces and laminar fuselage concepts, currently employed only on a small business plane, to a regional aircraft. In fact, it has been demonstrated that internal spaces and structures can be successfully adapted to this unconventional architecture; moreover, an advanced gear retraction system has been conceptualized, avoiding drag-increasing external fairings.

The system design phase has helped defining the power and weight requirements of the aircraft and, though not specific to this plane, has showed an effective integration in the new configuration.

The simulation part of the work, although with its limited validity, has showed encouraging results, underlining that there is a fuel-saving potential in this configuration.

The aircraft design is far from complete and even the simulation results are of limited reliability; however it can be interpreted as a small piece of evidence that the Piaggio P180 configuration could be effectively used in larger airplanes.

The whole work could be an excellent starting point for future research and more detailed design.

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