# Hybrid-electric propulsion concept design for a radical short-medium range aircraft

Invited Presentation at the 2022 Workshop on Innovative Scientific/Technical Approaches for a Climate Neutral Air Transport

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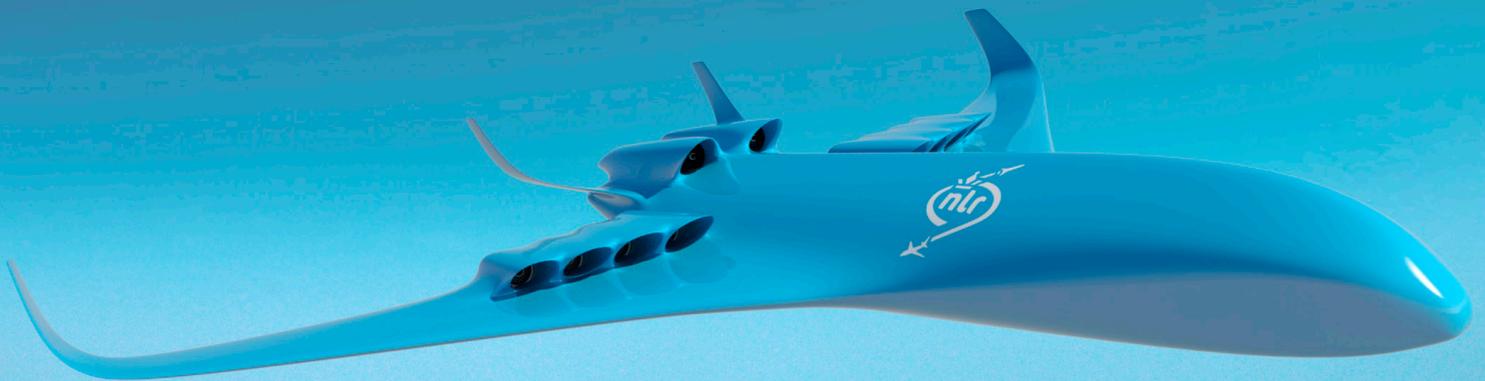**Dedicated to innovation in aerospace**

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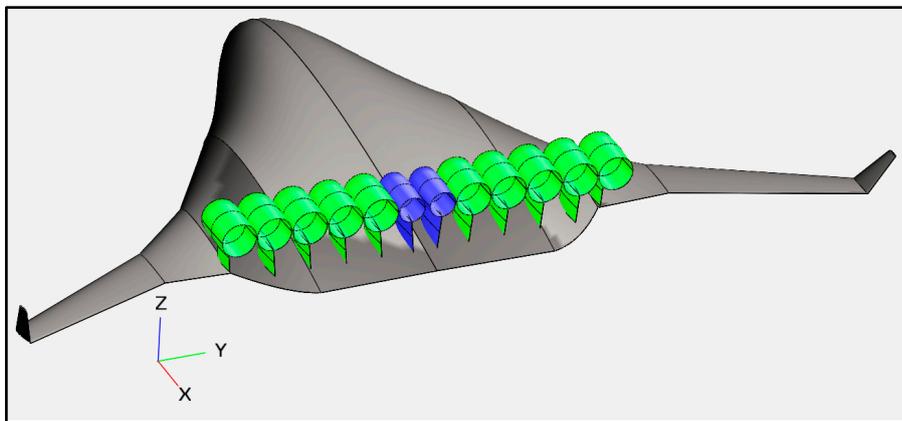


Illustration of a radical short-medium range aircraft with hybrid-electric propulsion.

## Problem area

Further reduction of greenhouse gas emissions is essential for aviation to accommodate the expected increase in air travel and to decrease its contribution to climate change. This calls for ambitious research and disruptive technology solutions, well beyond the continuous improvement of current aircraft technologies.

## Description of work

In the European Horizon 2020 project IMOTHEP (Investigation and Maturation of Technologies for Hybrid Electric Propulsion) key technologies for hybrid electric propulsion (HEP) are explored for application on regional and short-medium range (SMR) transport aircraft. Among others, a radical aircraft configuration based on a blended-wing-body (BWB) concept with turbo-electric (TE) powertrain and distributed electric propulsion (DEP) is considered for SMR missions. The explorations are driven by aircraft level requirements and design objectives, including a reduction of 40% in $CO_2$ emissions in comparison with 2014 State of Art aircraft.

## Results and conclusions

This report describes the design approach that was followed, including the definition and evaluation of appropriate reference and baseline aircraft configurations. It describes the considerations that were made for the selection of a suitable HEP powertrain architecture. It also presents the detailed outcomes of the design explorations and mission evaluations that were made for the TE powertrain of the BWB aircraft for application in SMR air transport. Reductions in Typical range (800 NM) mission fuel burn of up to 33% are predicted for the considered BWB configuration. However, of this 33%, about 22% results from the expected aircraft technology developments (like drag and weight reductions) up to 2035. About 10% fuel reduction is achieved actually due to the application of HEP and "more electric aircraft" (MEA) technologies. Furthermore, it is shown that reducing the maximum required design range from 2750 NM to 1200 NM has only very small impact of about 1% reduction in Typical range fuel burn.

## Applicability

The present study was focused on a BWB aircraft configuration for SMR missions with DEP and TE powertrain. But the design approach and the methods and tools that were used are more widely applicable, for example to other HEP architectures, like parallel or serial hybrid, and other aircraft configurations like regional or long range aircraft.

**GENERAL NOTE**
This report is based on a presentation held at the Workshop on Innovative Scientific/Technical Approaches for a Climate Neutral Air Transport on 22 – 24 Nov. 2022 in Barcelona, Spain.

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# Abbreviations

| ACRONYM | DESCRIPTION |
|---|---|
| BAS | Baseline aircraft |
| BHL | Bauhaus Luftfahrt, Germany) |
| BWB | Blended-wing-body |
| CON | Conservative |
| DLR | German Aerospace Centre, Germany |
| ECS | Environmental control system |
| GSP | Gas-turbine Simulation Program |
| HEP | Hybrid electric propulsion |
| HPC | High pressure compressor |
| IMOTHEP | Investigation and Maturation of Technologies for Hybrid Electric |
| IPS | Ice protection system |
| LPC | Low pressure compressor |
| MASS | IEEE International Conference on Mobile Ad Hoc and Sensor Systems |
| MTOM | Maximum take-off mass |
| NLR | Royal NLR - Netherlands Aerospace Centre |
| OAD | Overall aircraft design |
| OEI | One engine inoperative |
| ONERA | French Aerospace Lab, France |
| PTO | Power-offtake |
| RAD | Radical |
| REF | Reference aircraft |
| REFX | Existing reference aircraft, |
| REG | Regional |
| SMR | Short-medium range |
| TE | Turbo-electric |
| TLARs | Top-level aircraft requirements |
| TRL | Technology readiness level |
| TSFC | Thrust specific fuel consumption |

# Hybrid-electric propulsion concept design for a radical short-medium range aircraft

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**Abstract**

Ambitious targets for the coming decades have been set for the further reduction of aviation greenhouse gas emissions. Hybrid Electric Propulsion (HEP) concepts can offer potential for mitigation of these aviation emissions. To investigate this potential in an adequate level of detail, the European research project IMOTHEP (Investigation and Maturation of Technologies for Hybrid Electric Propulsion) explores key technologies for HEP in close relation with developments of aircraft mission and configuration. This paper presents conceptual level design investigations on radical HEP aircraft configurations for short-medium range (SMR) missions. In particular, a blended-wing-body configuration with turbo-electric powertrain and distributed electric propulsion is investigated using NLR's aircraft evaluation tool MASS. For the aircraft and powertrain design, representative top-level aircraft requirements have been defined in IMOTHEP and the reference aircraft for the assessment of potential benefits is based on the Airbus A320neo aircraft.

## 1. Introduction

The further reduction of greenhouse gas emissions is essential for aviation to accommodate the expected increase in air travel and at the same time to pursue its service to society and environment. This calls for ambitious research and disruptive technology solutions, well beyond the continuous improvement of current aircraft technologies. In the European Horizon 2020 project IMOTHEP (Investigation and Maturation of Technologies for Hybrid Electric Propulsion) [1] the exploration of key technologies for hybrid electric propulsion (HEP) is under investigation. This has to be addressed in close relation with developments of aircraft mission and configuration, to derive relevant specifications for the investigation of electric components, such as the power needs and the operational constraints. This interrelation in IMOTHEP between the integrated design on aircraft vehicle level and the developments of key technologies for HEP components is illustrated in Figure 1.
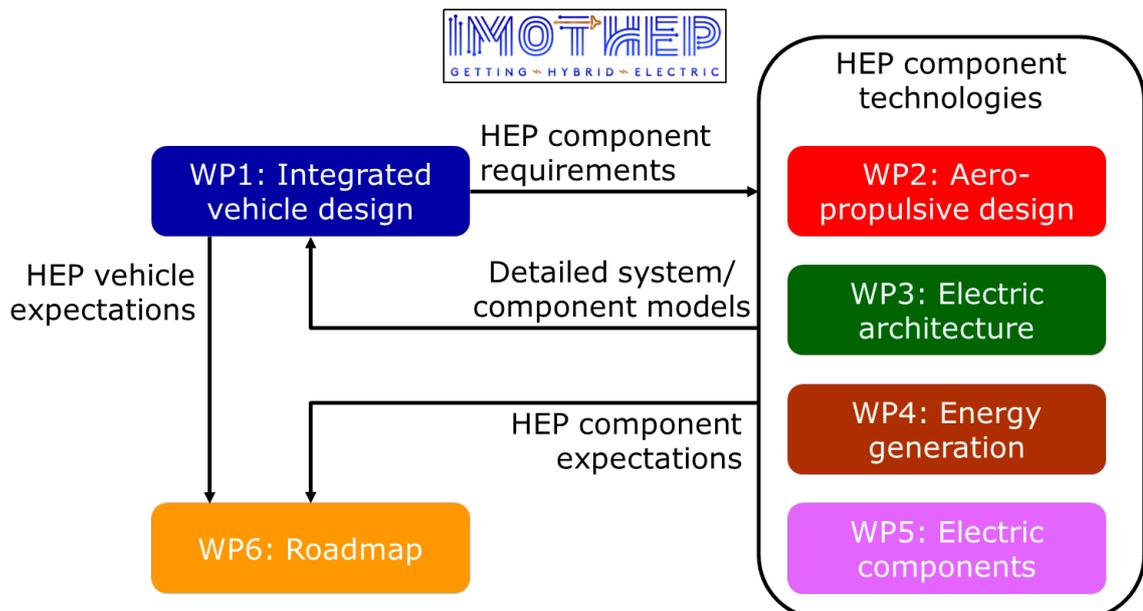
## IMOTHEP project overview



Figure 1: Global overview of the IMOTHEP project, illustrating the interrelation between the integrated design on aircraft vehicle level and the developments of HEP components technologies.

As part of the IMOTHEP project's activities on integrated vehicle design, conceptual level design investigations are executed on various aircraft configurations. These configurations are targeted for missions that contribute significantly to unwanted aviation emissions, i.e. regional (REG) missions and short-medium range (SMR) missions. For both mission types, different types of aircraft- and propulsion configurations are considered: conservative (CON) and radical (RAD). The conservative configurations include moderate technology developments without substantial design changes in the airframe. The radical configurations include more advanced technology developments in combination with unconventional airframe design. For the radical (RAD) configuration for the short-medium range (SMR) mission, in particular, a blended-wing-body (BWB) configuration with turbo-electric (TE) powertrain and distributed electric propulsion (DEP) is investigated.

The conceptual design investigations are based on representative top-level aircraft requirements (TLARs) that have been defined in the IMOTHEP project by the industrial airframers who are partners in IMOTHEP. The reference aircraft for the SMR mission is based on the A320neo aircraft, slightly adapted to comply with the IMOTHEP TLARs for SMR. The design objectives for the conceptual investigations are based on the IMOTHEP project targets. These project targets are expressed as criteria for emission reductions. These criteria are set for all design studies and are based on the ambition to achieve 10% more reduction than the targets that were set for 2035 in the European research program Clean Sky 2 (CS2) [2]. This means for IMOTHEP a reduction of 40% in $CO_2$ emissions for SMR aircraft in comparison with 2014 State of Art[1].

The NLR investigations for the SMR-RAD configuration are done using the NLR tools for conceptual aircraft design and for mission evaluation MASS (Mission, Aircraft and Systems Simulation for HEP analysis) [3]. MASS includes models coming from various other tools, such as for flight mission modelling, aircraft modelling, electric components modelling and engine modelling (e.g. as provided by GSP: Gas-turbine Simulation Program [4]) and predicts fuel and energy consumption and emissions. This paper presents the conceptual level aircraft design investigations of HEP architectures for the SMR-RAD configuration. From these investigations, the main results for fuel consumption, emissions and propulsive equipment sizing for the BWB airframe in combination with a power train based on the all TE architecture are given.

## 2. Approach and Methods

In the IMOTHEP project, the conceptual level design evaluations for each of the aircraft configurations (i.e. the conservative (CON) and radical (RAD) configurations for the regional (REG) and short-medium range (SMR) missions) are executed in the same way according to the overall IMOTHEP design logic. All the aircraft design studies must comply with the TLARs that are defined in the project, separately for the REG and for SMR. The overall aircraft design (OAD) approach and tools that are operational at the project partners, are used for the modelling of aircraft and HEP components. Fast modelling methods are used to rapidly assess and compare the different aircraft configurations and propulsion options. The same technology assumptions on HEP components are consistently used in the various configuration studies. Specialised partners from industry and research provide the specific inputs for the HEP components in the power train, like advanced or simplified models or estimates of performances and masses for the different components and assumptions of technology developments in 2035.

The aircraft level conceptual design activities are executed by different partners of the IMOTHEP project:
- BHL (Bauhaus Luftfahrt, Germany) focus on the regional-conservative (REG-CON) configuration;
- DLR (German Aerospace Centre, Germany) focus on the regional-radical (REG-RAD) configuration;
- ONERA (French Aerospace Lab, France) focus on the short-medium-range-conservative (SMR-CON) configuration;
- NLR (Netherlands Aerospace Centre, Netherlands) focus on the short-medium-range-radical (SMR-RAD) configuration.

The different partners' tools are used for the study of the radical and conservative concepts. The quality and consistency of these tools is ensured by benchmarking the tools on the reference and baseline configurations. The actual HEP design studies yield the intended HEP aircraft configuration. This implementation of the IMOTHEP design logic is expressed in the Figure 2 below.
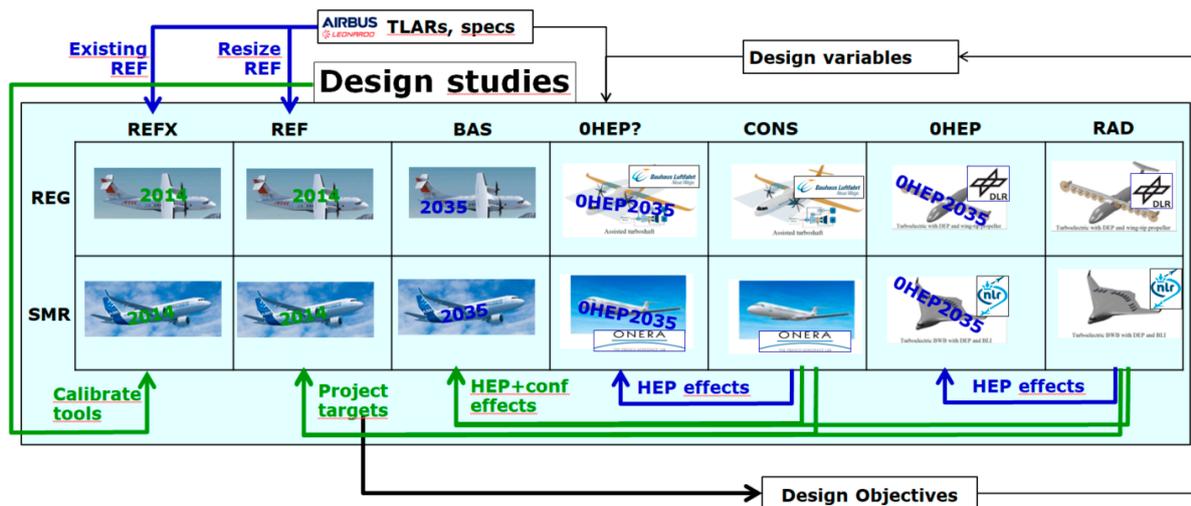
Figure 2: Illustration of the various configurations and their variants, and the design evaluations and assessments envisaged in the IMOTHEP design logic.

Besides the CON and RAD configurations, also several other configurations for the REG and SMR missions must be evaluated in order to make the right assessments according to the IMOTHEP design logic. These assessments are mentioned and illustrated by the green and blue arrows in the bottom of the Figure 2. The other configurations are the REFX, REF, BAS, and 0HEP configurations, as also illustrated in Figure 2. Each of the considered configurations are motivated and explained as follows:

- REFX = existing reference aircraft, operational in 2014, i.e. ATR42 for REG and A320neo for SMR:
  o intended for calibration (or validation) of all partners' tools against aircraft level performance values that are provided by the airframers in the IMOTHEP consortium.
- REF = reference aircraft, i.e. with 2014 technologies assumptions, but adapted to comply with IMOTHEP TLARs:
  o intended for assessment of the resulting performance values of the RAD and CON aircraft with HEP powertrains in comparison to 2014 state of the art aircraft performance.
- BAS = baseline aircraft, i.e. the same as REF but with 2035 technology assumptions:
  o intended for assessment of the resulting performance values of the RAD and CON aircraft with HEP powertrains in comparison to the performance of 2035 technologies in conventional aircraft powertrains.
- CON = conservative aircraft configuration with HEP and with 2035 technology assumptions:
  o intended for evaluation and assessment of the resulting performance values for the CON aircraft with HEP powertrains for the REG and SMR missions.
- RAD = radical aircraft configuration with HEP and with 2035 technology assumptions:
  o intended for evaluation and assessment of the resulting performance values for the RAD aircraft with HEP powertrains for the REG and SMR missions.
- 0HEP = aircraft with an innovative architecture (e.g. blended-wing-body (BWB) for SMR) but with conventional turboprop or turbofan propulsion, with no hybridization:
  o intended for assessment of the isolated configuration effects. This 0HEP is mainly relevant for the RAD configurations; their airframe configuration stems from the considered HEP aircraft. For the CON configurations, this 0HEP has appeared to be not relevant because it has no substantial difference with BAS.

The Table 1 below gives an overview of all these configurations that are considered in the IMOTHEP design logic, and the technologies, or requirements, that they cover.

Table 1 : All the configurations in the IMOTHEP design logic, and the technologies or requirements that they cover.

| Technologies included: | REFX | REF | BAS | CON | RAD | 0HEP |
|---|---|---|---|---|---|---|
| Existing operational reference aircraft | ✓ | | | | | |
| With compliance to IMOTHEP TLARs | | ✓ | ✓ | ✓ | ✓ | ✓ |
| With 2035 technology assumptions | | | ✓ | ✓ | ✓ | ✓ |
| With HEP powertrain | | | | ✓ | ✓ | |
| With radical aircraft configuration | | | | | ✓ | ✓ |

With this implementation of the IMOTHEP design logic (Figure 2) the concept design studies of the various aircraft configurations are executed. The specifications from the airframers for the reference configurations (REF) and for the TLARs are key inputs for these design studies. The full list of TLARs is long and includes several detailed values for operational requirements and is beyond scope of this paper. The main TLARs to be satisfied are summarized in the following Table 2, where all given values shall be considered as lower limit.

Table 2: The main TLARs considered in the concept design studies in IMOTHEP.

| TLARs | REG | SMR |
|---|---|---|
| Design Range | 400-600 NM (741-1111 km) | 1200-2750 NM (2222-5093 km) |
| Typical Range | 200 NM (370 km) | 800 NM (1482 km) |
| Number of PAX (Design Payload) | 40 (4240 kg) | 150 (15900 kg) |
| Design Cruise Mach number | 0.4 [0.4, 0.48] | 0.78 [0.78, 0.82] |
| Seat pitch | 30 in (0.762 m) | 30 in (0.762 m) |

It must be noted that the bandwidths specified for the design range and cruise Mach number are considered in IMOTHEP in order to account for certain flexibility in the aircraft capabilities. The reason is that some variation of a certain TLAR (e.g. the design range) may result in substantial benefits for the design objective (i.e. typical mission fuel burn). These variations in TLARS are assessed to some extent through sensitivity evaluations. Also, although the typical range is listed in the Table 2, it is not a TLAR per se but it is included here because it represents the range for which several additional requirements shall be fulfilled and for which the design objectives are evaluated. These requirements and objectives will be further explained below.

The IMOTHEP project's targets are the basis for the design objectives in the concept design studies, as indicated in Figure 2. The IMOTHEP project targets are based on the targets that were set for 2035 in Clean Sky 2 (CS2) [2], with an additional 10% reduction. The main target that is considered in IMOTHEP is the reduction of $CO_2$ emissions [1], which is directly proportional to fuel consumption. This leads to the following fuel reduction targets for IMOTHEP:
- for REG: -50% fuel consumption for typical missions.
- for SMR: -40% fuel consumption for typical missions.

All values given here are in comparison with 2014 State of Art aircraft. Technology assumptions for the different sub-systems in the various configurations are targeted at technology readiness level (TRL) 6 in 2035.

This paper deals mainly with the concept design study of the SMR-RAD configuration, which is executed according to the IMOTHEP design logic. Besides the SMR-RAD, also the SMR-REFX, -REF, -BAS and -0HEP configurations must be evaluated in order to make the intended assessments. This concept design study of the SMR-RAD and its related configurations will be further elaborated in the following sections.

### 3. Aircraft configurations for short-medium range (SMR)

### 3.1 Simulation tools for conceptual aircraft design and mission evaluation

The investigations for the SMR-RAD configuration and its variants are done using the NLR tool for conceptual aircraft design and mission evaluation: MASS (Mission, Aircraft and Systems Simulation for HEP analysis, [3]) (Figure 3). MASS includes models coming from various other tools, such as for flight mission modelling, airframe modelling, electric components modelling and engine modelling (GSP: Gas-turbine Simulation Program [4]). Besides for sizing of aircraft and powertrain components, MASS can be used for prediction of fuel and energy consumption and emissions for a given flight. Any HEP architecture can be modelled in MASS, including parallel HEP as illustrated in the schema in Figure 3, but also series HEP and TE architectures.
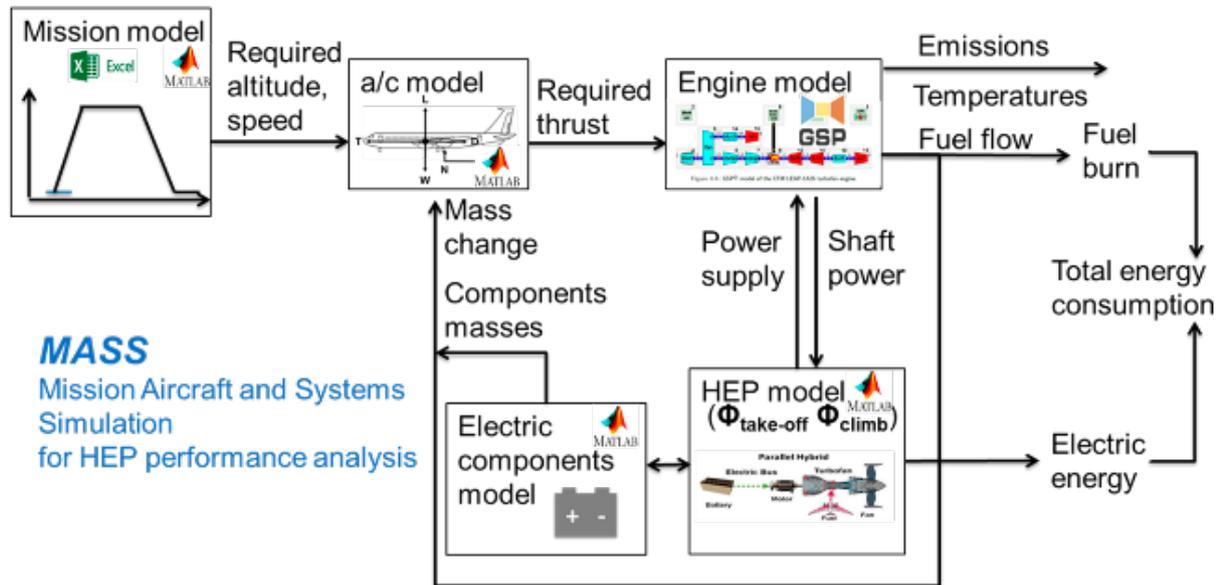
Figure 3: Illustration of the modelling and analysis process in MASS [3].

## 3.2 Calibration against existing reference aircraft (SMR-REFX)

The calibration in IMOTHEP of the MASS tool is based on the evaluation of the existing reference configuration for SMR (SMR-REFX), which is the A320neo with its 2014 technologies as it is operational in service. The comparison is based on fuel consumption figures provided by Airbus for the Typical Range mission of 800 NM (1482 km). For the NLR evaluations of the SMR-REFX configuration, the aircraft model of the A320neo and engine model of the CFM-LEAP-1A turbofan are based on existing models available at NLR [3].

**Assumptions for aircraft and mission**
For the evaluations of the SMR-REFX configuration, the following assumptions are applied. These assumptions are also used for the evaluations of the SMR-REF, SMR-BAS, SMR-0HEP and SMR-RAD configurations. Some assumptions will be slightly modified for some of the configurations, which will be explained when relevant.

- Payload:
  - According to the TLARs, the design-payload is 150 PAX@106 kg=15900 kg. The maximum payload is 20000 kg=189 PAX@106 kg, but that is not used in the REFX evaluations, only in some specific evaluations of the other configurations. The design-payload of 150 PAX@106 kg=15900 kg will be used in all SMR evaluations.
- Range:
  - The typical range mission at design-payload is evaluated, in which the distance on ground between take-off and landing is 800 NM (1482 km).
- Atmosphere:
  - In all mission evaluations, international standard atmosphere (ISA) [5] is assumed.
- Take-off and landing:
  - In all mission evaluations, take-off and landing are assumed at airports on sea level.
- Taxi:
  - The mission evaluation includes taxiing at airports, in which the taxi definitions given in CeRAS [6] are adopted: taxi-out is defined as 540 s taxiing at constant speed of 30 kts (15.4 m/s) yielding 8.33 km, and taxi-in is defined as 300 s taxiing at constant speed of 30kts (15.4 m/s) yielding 4.63 km. The total taxi distance results in 12.96 km and will be used in all SMR-RAD evaluations.
- Fuel burn:
  - The actual mission fuel burn figure that is calculated in NLR's mission evaluation is the "Block-off/Block-on fuel", i.e. trip-fuel + taxi-fuel. Here trip-fuel is the fuel consumption from brake release on takeoff at the departure aerodrome to the landing touchdown at the destination, and taxi-fuel is the fuel consumption during taxi-out and taxi-in. The "Block-off/Block-on fuel" will be evaluated in all SMR-RAD evaluations of NLR.

- Reserve-fuel:
  - o Definitions given in CeRAS for reserve-fuel are as follows: reserve-fuel is the sum of contingency fuel, alternate fuel, final reserve fuel, additional fuel, extra fuel, i.e. total fuel on board minus trip-fuel and taxi-fuel. The reserve-fuel needed for the SMR typical mission is approximately 3000 kg. Therefore, for simplification, the assumption in NLR's mission evaluation is to use a fixed reserve-fuel of 3000 kg, which will be used in all SMR-RAD evaluations of NLR.
- Cruise:
  - o Definitions given in CeRAS for SMR cruise condition is Mach 0.78 flight speed at an initial altitude of 35 kft (10.7 km).
- Power-offtake (PTO):
  - o Mechanical power off-takes from the low-pressure turbine (LPT) shaft, for power supply to non-propulsive on-board systems like pumps and generators, are taken into account. CeRAS applies a fixed PTO of 52 kW per engine. This value (104 kW in total) will be used in all SMR-RAD evaluations of NLR.
- Bleed-offtake:
  - o Bleed air off-takes from the low-pressure and high-pressure compressors (LPC, HPC), for power supply to non-propulsive on-board systems like ECS and IPS, are taken into account. CeRAS applies a fixed bleed off-take of 0.98 kg/s per engine. This value (1.96 kg/s in total) will be used in all SMR-RAD evaluations of NLR.

With these assumptions, the existing reference configuration SMR-REFX has been evaluated with the NLR MASS tool for the typical mission. The evaluation results for the SMR-REFX configuration are given in Table 3.

Table 3: Design evaluations overview for SMR-REFX. The green cell indicates the key outcome of the evaluation, i.e. the typical mission fuel burn.

| Refx variants | Wing area [m2] | range [km] | cruise altitude [km] | Cruise TAS [m/s] | Reserve fuel [kg] | payload [kg] | mOE [kg | average total PTO [kW] | average total bleed (kg/s) | cruise L/D | cruise Fn [kN] | cruise SFC [g/kNs] | cruise fuel flow kg/s | fuel burn [kg] | mTO [kg] | TakeOff max(CL) | Landing max(CL) | max TT4 [K] | max N1 [%] | TTC [min] 1.5kft - 35kft |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| **Refx** | | | | | | | | | | | | | | | | | | | | |
| Refx1: 800 NM | 123 | 1482 | 10.7 | 231 | 3000 | 15900 | 44300 | 104 | 1.96 | 16.1 | 39.7 | 15.46 | 0.61 | 4855 | 68055 | 1.56 | 1.67 | 1770 | 98.1 | 34.3 |

The key results and conclusions from the evaluations of the SMR-REFX configuration are the following:
1. The SMR-REFX evaluation results in a typical mission fuel burn of 4855 kg. This is an acceptable small deviation of less than 5% from the reference value for the typical mission fuel burn.
2. From this result it is concluded that the MASS tool is sufficiently calibrated as conceptual design tool for the SMR-RAD study.

## 3.3 Evaluation of the reference SMR aircraft (SMR-REF)

For the evaluations of the SMR-REF configuration, the same aircraft and engine models are used as for the REFX configuration, i.e. the models of A320neo and CFM-LEAP-1A. But for the SMR-REF, the aircraft model is re-designed to comply with the IMOTHEP TLARs. The assumptions for aircraft and mission as described above for the SMR-REFX are also used for the SMR-REF evaluations.

**TLARs**
The re-design of the SMR-REF aircraft is simplified to the key design variable wing area ($S_w$), which is determined such that the mission fuel is minimized and the aircraft complies with the following TLARs:
1. Wing span: must be lower than 36 m.
2. Payload:
   a. Design payload: is based on 150 PAX, i.e. 150 x 106 kg = 15900 kg.
   b. Maximum payload: 20 t = 20000 kg.
3. Design range: the bandwidth for the range, as indicated in Table 2, is considered as follows:
   a. a configuration Ref1 is defined that shall be able to fulfil a mission with a range of 2750 NM (i.e. 5093 km) at the design payload (i.e. 15900 kg).
   b. a configuration Ref2 is defined that shall be able to fulfil a mission with a range of 1200 NM (i.e. 2222 km) at the design payload (i.e. 15900 kg).
4. Design mission: an Initial Cruise Altitude (ICA) of 33 kft (10.06 km) at ISA+10 conditions shall be fulfilled:

     a.   this is assumed to be fulfilled by the cruise altitude of 35kft (10.7 km) at ISA conditions that is used in all design missions, which is for density approximately equivalent to 33 kft (10.06 km) at ISA+10.

5.   Take-off field length (TOFL): must be lower than 2200m at ISA+15 conditions:
     a.   this is assumed to be fulfilled by the approximately equivalent TOFL of less than 2000 m at ISA conditions, which is used in all missions.

6.   Climb time: must be lower than 35 min from 1500 ft (0.46 km) to 33kft (10.06 km) at ISA+10 conditions:
     a.   this is assumed to be fulfilled by the approximately equivalent climb time of less than 35 min from 1500ft (0.46 km) to 35kft (10.7 km) at ISA conditions, which is used in all missions.

7.   Approach speed: must be lower than 138 kts (71 m/s) in all missions.

8.   Landing distance: must be lower than 1800 m in all missions.

9.   Typical range:
     a.   800NM (1482 km) at the default cruise conditions (Mach 0.78, 35 kft (10.7 km); CeRAS [6]).
     b.   Additional Typical range missions are defined:
          i.   default cruise conditions for max payload = 20 t (20000 kg)
          ii.   default payload at high cruise speed conditions: Mach 0.82, 35 kft (10.7 km)
          iii.   default payload, speed at high cruise conditions: 39 kft (11.9 km), with top of climb (TOC) climb rate of at least 100 ft/min (0.5 m/s) at ISA conditions.

10.   One engine inoperative (OEI): SMR-REF shall be able to fulfil a mission at maximum take-off mass (MTOM) including take-off (TO) and climb until 15 kft (4.6 km) at a climb rate greater than 100 ft/min (0.51 m/s).

**Design constraints**

For all missions that shall comply with these TLARs, several design constraints are checked for violations in order to assess the feasibility of the design. These design constraints are based on the following criteria: the maximum allowable values of $C_L$ (aircraft level lift coefficient), $F_n$ (aircraft level net thrust force), $N_1$ (turbofan engine low pressure spool rotational speed) and $T_{T4}$ (turbofan engine high pressure turbine inlet temperature) shall not be exceeded. The allowable values for $N_1$ and $T_{T4}$ are based on or derived from EASA Type certification sheet of the CFM-LEAP-1A engine [7]. The design constraints are determined as follows:

- $C_L$: the values of $C_L$ during the mission that are found from the MASS simulations shall always remain below the maximum possible $C_L$ value. Because the current conceptual modelling does not include high fidelity methods to calculate the maximum possible $C_L$ value, we estimate this value from the A320neo aircraft characteristics and mission specifications. This maximum possible $C_L$ value occurs in the low speed mission segments of take-off and landing.
  - For rotation at take-off, the speed of A320neo shall be 150 kts (77.2 m/s) or more [8]. For straight-and-level flight at MTOM (i.e. 79 t (79000 kg) for the A320neo in REF) this leads to $C_L = \frac{MTOM \times g}{S_{wing} \times 1/2 \rho v^2} = 1.73$. For the rotation we assume that 10% extra lift is needed for change of flight path angle, yielding $maxC_L = 1.9$.
  - For landing, we assume a final approach speed for A320neo of at least 131.5 kts (67.6 m/s) at maximum landing mass (MLM = 67400 kg [8]. For straight-and-level flight this leads to $maxC_L = \frac{MLM \times g}{S_{wing} \times 1/2 \rho v^2} = 1.92$. The SMR TLARs require an approach speed $v_{app}$ <138 kts (71 m/s), so $v_{app}$=137 kts (70.5 m/s) is used in all missions.
  - With these estimated $maxC_L$ values, the variations of the wing area (i.e. the key design variable) shall be made such that the design constraint $C_L < maxC_L$ is always fulfilled.
- $F_n$ (net thrust force): the maximum take-off thrust of 120 kN per engine according to CFM-LEAP-1A type certificate [7] shall never be exceeded.
- $N_1$ (low pressure spool rotational speed): $N_1$ shall remain below 101% of the maximum design speed according to CFM-LEAP-1A type certificate [7].
- $T_{T4}$ (high pressure turbine inlet temperature): $T_{T4}$ shall remain below 1850 K.

**SMR-REF design evaluations**

Two separate REF configurations are considered for the different design ranges: the Ref1 configuration for the long design range of 2750 NM (5093 km) (TLARs 3.a in the list above) and the Ref2 configuration for the short design range of 1200 NM (2222 km) (TLARs 3.b in the list above). For both the Ref1 and Ref2 configurations, the re-design of the wing area is performed by checking the design constraints described above for the various missions as prescribed by the TLARs. Both the Ref1 and Ref2 configurations are used for the assessment of the SMR-RAD configuration.

Table 4: Design evaluations overview for SMR-REF. Explanations of missions: 800NM@MP means: 800 NM (1482 km) mission at maximum payload of 20000 kg. 800NM@MA means: 800 NM (1482 km) mission at maximum altitude of 11.9 km. OEI means: one engine inoperative mission including take-off + climb until 15 kft at climb rate > 100 ft/min. The yellow cells indicate the key variables that are set for each evaluation. The green cells indicate the key outcome of an evaluation, i.e. the typical mission fuel burn. The red cells indicate the violation of constraints in an evaluation. The orange cells indicate an active constraint in an evaluation. The grey cells indicate that the data are not relevant.

| Ref variants and missions | Wing area [m2] | range [km] | Cruise altitude [km] | Cruise TAS [m/s] | Reserve fuel [kg] | payload [kg] | mOE [kg | average total PTO [kW] | average total bleed (kg/s) | cruise L/D | cruise Fn [kN] | cruise SFC [g/kNs] | cruise fuel flow kg/s | fuel burn [kg] | mTO [kg] | TakeOff max(CL) | Landing max(CL) | max TT4 [K] | max N1 [%] | TTC [min] 1.5kft-35kft |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| **Ref1** | | | | | | | | | | | | | | | | | | | | |
| Ref1: 2750NM | 123 | 5093 | 10.7 | 231 | 3000 | 15900 | 44300 | 104 | 1.96 | 16.3 | 42.0 | 15.35 | 0.64 | 15269 | 78469 | 1.80 | 1.67 | 1815 | 101.0 | 34.3 |
| Ref1: 800NM@MP | 123 | 1482 | 10.7 | 231 | 3000 | 20000 | 44300 | 104 | 1.96 | 16.3 | 41.7 | 15.36 | 0.64 | 5053 | 72353 | 1.66 | 1.78 | 1788 | 99.3 | 34.3 |
| Ref1: 800NM 0.82M@35kft | 123 | 1482 | 10.7 | 243 | 3000 | 15900 | 44300 | 104 | 1.96 | 12.5 | 51.2 | 15.48 | 0.79 | 5344 | 68544 | 1.57 | 1.67 | 1771 | 100.8 | 34.3 |
| Ref1: 800NM@MA | 123 | 1482 | 11.9 | 230 | 3000 | 15900 | 44300 | 104 | 1.96 | 16.5 | 41.8 | 15.11 | 0.63 | 4884 | 68083 | 1.56 | 1.66 | 1770 | 99.8 | 34.3 |
| Ref1: OEI | 123 | 492 | 4.6 | 130 | 15500 | 15900 | 44300 | 52 | 0.98 | 16.6 | 56.8 | NaN | 0.68 | 2863 | 78563 | 1.69 | n.a. | 1833 | 101.4 | 61.5 |
| Ref1: OEI | 120 | 492 | 4.6 | 130 | 15500 | 15900 | 44084 | 52 | 0.98 | 16.6 | 56.6 | NaN | 0.67 | 2845 | 78329 | 1.73 | n.a. | 1829 | 101.6 | 61.5 |
| Ref1: 800NM | 123 | 1482 | 10.7 | 231 | 3000 | 15900 | 44300 | 104 | 1.96 | 16.1 | 39.7 | 15.46 | 0.61 | 4855 | 68055 | 1.56 | 1.67 | 1770 | 98.1 | 34.3 |
| | | | | | | | | | | | | | | | | | | | | |
| **Ref2** | | | | | | | | | | | | | | | | | | | | |
| Ref2: 1200NM | 123 | 2222 | 10.7 | 231 | 3000 | 15900 | 44300 | 104 | 1.96 | 16.1 | 40.1 | 15.44 | 0.62 | 6902 | 70102 | 1.61 | 1.66 | 1778 | 98.6 | 34.3 |
| Ref2: 1200NM | 113 | 2222 | 10.7 | 231 | 3000 | 15900 | 43580 | 104 | 1.96 | 16.4 | 39.1 | 15.49 | 0.61 | 6748 | 69228 | 1.73 | 1.79 | 1772 | 97.9 | 34.3 |
| Ref2: 800NM@MP | 113 | 1482 | 10.7 | 231 | 3000 | 20000 | 43580 | 104 | 1.96 | 16.5 | 40.8 | 15.41 | 0.63 | 4957 | 71536 | 1.79 | 1.91 | 1784 | 98.6 | 34.3 |
| Ref2: 800NM@MP | 112 | 1482 | 10.7 | 231 | 3000 | 20000 | 43508 | 104 | 1.96 | 16.5 | 40.7 | 15.41 | 0.63 | 4948 | 71456 | 1.80 | 1.93 | 1783 | 98.6 | 34.3 |
| Ref2: 800NM 0.82M@35kft | 113 | 1482 | 10.7 | 243 | 3000 | 15900 | 43580 | 104 | 1.96 | 12.7 | 49.9 | 15.52 | 0.77 | 5222 | 67702 | 1.69 | 1.79 | 1765 | 100.1 | 34.3 |
| Ref2: 800NM@MA | 113 | 1482 | 11.9 | 230 | 3000 | 15900 | 43580 | 104 | 1.96 | 16.5 | 41.4 | 15.12 | 0.63 | 4804 | 67284 | 1.68 | 1.79 | 1763 | 99.5 | 34.3 |
| Ref2: OEI | 113 | 492 | 4.6 | 130 | 7000 | 15900 | 43580 | 52 | 0.98 | 17.5 | 47.8 | NaN | 0.58 | 2433 | 68913 | 1.62 | n/a | 1769 | 100.8 | 61.5 |
| Ref1: 800NM | 113 | 1482 | 10.7 | 231 | 3000 | 15900 | 43580 | 104 | 1.96 | 16.3 | 38.6 | 15.52 | 0.60 | 4744 | 67224 | 1.68 | 1.79 | 1762 | 97.3 | 34.3 |

For the Ref1 configuration the results given in Table 4 represent the following:

- The first row (Ref1: 2750 nm…) presents the results for the long design range mission (2750 NM@35 kft, i.e. 5093 km@10.7 km) with design payload (15900 kg). It is found that the take-off mass (mTO) is 78469 kg, which is below MTOM (79000 kg), and that the maximum LPT shaft speed of the engine is just feasible (101%).
- In the following 3 rows the mission evaluations for the TLARs of maximum payload (20000 kg), maximum cruise speed (Mach 0.82@35 kft (10.7 km)) and maximum altitude 39 kft (11.9 km) are evaluated, and neither any violation of any criteria occurs.
- In the following row the mission evaluation for the failure case "OEI at TO with MTOM" (TLARs 10.) is evaluated, where it is found that the maximum LPT shaft speed of the engine is just not feasible but accepted (101.4%, rounded to 101%). It must be noted that here the reserve fuel was set to 15500 kg. This does not represent actual fuel, but merely a mass component in order to achieve a take-off mass (mTO) of 78563 kg, which is (approximately) equal to maximum take-off mass (MTOM).
- Then the wing area is reduced from 123 m² to 120 m² and the operating empty mass (mOE) is decreased accordingly by 216 kg, which is based on an estimation of 20% wing mass of mOE. The same OEI mission is evaluated, where it is found that the maximum $N_1$ (LPT shaft speed of the engine) that occurs during the mission is not feasible (101.6%, rounded to 102%).
- The last row for Ref1 presents the results for the typical mission (800 NM@35 kft, i.e. 1482 km@10.7 km), where no violation of any criteria (as given in the last 6 columns on the right) occurs and a total mission fuel burn of 4855 kg is found.

Similarly, for the Ref2 configuration the results given in Table 4 represent the following:

- The first row (Ref2: 1200 nm…) for the SMR-Ref2 aircraft in the Table 4 presents the results for the short design range mission (1200 NM@35 kft, i.e. 2222 km@10.7 km) with design payload (15900 kg). It is found that the mTO (70102 kg) is far below MTOM (79000 kg) and no violation of any criteria occurs.

- The second row presents the results for the short design range mission (1200 NM@35 kft, i.e. 2222 km@10.7 km) with design payload (15900 kg), but here the wing area is reduced from 123 m$^2$ to 113 m$^2$ and the operating empty mass (mOE) is decreased accordingly by 720 kg. Also here no violation of any criteria occurs.
- In the following row the mission evaluation for the TLARs of maximum payload (20000 kg) is evaluated. It is found that the maximum $C_L$ that is required in the mission occurs during landing and is just feasible and is accepted ($C_L$=1.91).
- In the following row the wing area is slightly further reduced from 113 m$^2$ to 112 m$^2$ and the mOE is decreased accordingly by 72 kg compared to the 123 m$^2$ wing area. Again the mission evaluation for the TLARs of maximum payload (20000 kg) is evaluated. Now it is found that the maximum $C_L$ during landing is just not feasible ($C_L$=1.93).
- In the following 2 rows the mission evaluations for the TLARs of maximum cruise speed (Mach 0.82@35 kft (10.7 km)) and maximum altitude (11.9 km) are evaluated, and no violation of any criteria occurs.
- In the following row the mission evaluation for the failure case "OEI at TO" is evaluated, and no violation of any criteria occurs. Likewise, as for the SMR-Ref1 configuration, it must be noted that here the reserve fuel was set to 7000 kg in order to achieve a take-off mass (mTO) of 68913 kg.
- The last row presents the results for the typical mission (800 NM (1482 km)), where no violation of any criteria occurs and total mission fuel burn of 4744 kg is found.

The key results and conclusions from the evaluations of the SMR-REF configuration are the following:
1. For the SMR-Ref1 configuration, with maximum design range of 2750 NM@35 kft (5093 km@10.7 km), the wing area cannot be reduced below 123 m$^2$. The critical case is the failure case "OEI at TO with MTOM".
2. For the SMR-Ref1 configuration the total mission fuel burn for the typical mission is **4855 kg**.
3. For the SMR-Ref2 configuration, with maximum design range of 1200 NM@35 kft (2222 km@10.7 km), the wing area can be reduced to 113 m$^2$. The critical case is the mission evaluation for the TLARs of maximum payload (20000 kg).
4. For the SMR-Ref2 configuration the total mission fuel burn for the typical mission is **4744 kg**.

## 3.4 Evaluation of the baseline SMR aircraft (SMR-BAS)

Also, for the evaluations of the SMR-BAS configuration, the aircraft model of the A320neo and engine model of the CFM-LEAP-1A turbofan are based on the existing models available at NLR. The assumptions for aircraft and mission and the TLARs and the design constraints as described above for the SMR-REFX and SMR-REF are also used for the SMR-BAS evaluations.

**2035 technologies assumptions**

The SMR-BAS models of aircraft and engine are first updated for 2035 aircraft EIS technologies and then re-designed for the IMOTHEP TLARs. This re-design of the SMR-BAS aircraft is also simplified to the key design variable wing area. The 2035 technologies are based on the assumptions described in Table 5.

Table 5: The 2035 aircraft EIS technologies and improvement assumptions for SMR-BAS, adopted from [9].

| Aircraft Component | Improvement Measure | Affected parameter |
|---|---|---|
| Turbofan engine | Higher BPR, components improvement | TSFC -10% wrt NEO<br>T/W +3,7% wrt NEO<br>Wetted area to be adjusted |
| Wing | Lightweight material | Mass -10% wrt 2014 |
| Fuselage | Lightweight material | Mass -5% wrt 2014 |
| Landing gear | Lightweight material | Mass -15% wrt 2014 |
| Pylons | Lightweight material | Mass -5% wrt 2014 |
| Furnitures (seats, galleys, catering,…) | Lightweight materials | Mass -25% wrt 2014 |
| Aerodynamics | Morphing wing, turbulent coating, shock control, optimized winglet | +3.3% on L/D<br>-5% on CD0 wing<br>-50% on CD wave<br>-10% on CD induced (all wrt. 2014). |

These 2035 technologies assumptions were implemented in the SMR-BAS models in the following way:
- 2035 in comparison to 2014 turbofan engine assumptions:

- o A thrust specific fuel consumption (TSFC) reduction of 10% in comparison to the 2014 CFM-LEAP-1A engine has been implemented in the SMR-BAS models, accounting mainly for the expected fan propulsive efficiency improvements.
  - o Thrust-over-weight ratio (T/W) +3.7% in comparison to NEO: apply 3.7% reduction on engine mass, i.e. ~0.037*3000 kg=111 kg per engine, so 222 kg decreased mOE has been implemented in SMR-BAS models.
  - o Wetted area to be adjusted: because UHBR engines have larger fan diameter but shorter length, the change in nacelle wetted area is a bit speculative. Therefore, no change in nacelle wetted area has been implemented in SMR-BAS models.
- 2035 in comparison to 2014 component mass assumptions:
  - o Wing mass -10%: i.e. ~0.1*8800 kg=880 kg decreased mOE has been implemented in SMR-BAS models.
  - o Fuselage mass -5%: ~0.05*8800 kg=440 kg decreased mOE has been implemented in SMR-BAS models.
  - o Landing gear mass -15%: ~0.15*2200 kg=330 kg decreased mOE has been implemented in SMR-BAS models.
  - o Pylons mass -5%: ~0.05*650 kg=32.5 kg per pylon, so 65 kg decreased mOE has been implemented in SMR-BAS models.
  - o Furnitures mass -25%: ~0.25*2440 kg=610 kg decreased mOE has been implemented in SMR-BAS models.
- 2035 in comparison to 2014 aerodynamics assumptions:
  - o +3.3% on L/D: 3% reduction applied on CD has been implemented in SMR-BAS models.
  - o -5% on CD0 wing: Decrease CD0_wing by 5% has been implemented in SMR-BAS models.
  - o -50% on CD wave: Decrease CD_wave by 50% has been implemented in SMR-BAS models.
  - o -10% on CD induced: Decrease CD_induced by 10% has been implemented in SMR-BAS models.

The 2035 technologies assumptions yield total mass reduction on aircraft level (i.e. total decreased mOE) of 2547 kg.

**SMR-BAS design evaluations**

Just like for the SMR-REF, also for the SMR-BAS two separate BAS configurations are considered for the different design ranges: the Bas1 configuration for the long design range of 2750 NM (5093 km) (TLARs 3.a in the list above) and the Bas2 configuration for the short design range of 1200 NM (2222 km) (TLARs 3.b in the list above). For both the Bas1 and Bas2 configurations, the re-design of the wing area is performed by checking the design constraints described above for the various missions as prescribed by the TLARs. Both the Bas1 and Bas2 configurations are used for the assessment of the SMR-RAD configuration.

Table 6: Design evaluations overview for SMR-BAS. Explanations of missions: 800NM@MP means: 800 NM (2222 km) mission at maximum payload of 20000 kg. 800NM@MA means: 800 NM (2222 km) mission at maximum altitude of 11.9 km. OEI means: one engine inoperative mission including take-off + climb until 15 kft at climb rate > 100 ft/min. The yellow cells indicate the key variables that are set for each evaluation. The green cells indicate the key outcome of an evaluation, i.e. the typical mission fuel burn. The red cells indicate the violation of constraints in an evaluation. The orange cells indicate an active constraint in an evaluation. The grey cells indicate that the data are not relevant.

| BAS Variants and missions | Wing area [m2] | range [km] | Cruise alti-tude [km] | Cru-ise TAS [m/s] | Reserve fuel [kg] | payload [kg] | mOE [kg] | Aver-age total PTO [kW] | Aver-age total bleed (kg/s) | Cru-ise L/D | cruise Fn [kN] | cruise SFC [g/kNs] | cruise fuel flow kg/s | fuel burn [kg] | mTO [kg] | Take-Off max (CL) | Landing max(CL) | max TT4 [K] | max N1 [%] | TTC [min] 1.5kft - 35kft |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| **Bas1** | | | | | | | | | | | | | | | | | | | | |
| Ref1@2035 | 123 | 5093 | 10.7 | 231 | 3000 | 15900 | 41753 | 104 | 1.96 | 18.4 | 35.2 | 14.16 | 0.50 | 11912 | 72565 | 1.66 | 1.60 | 1779 | 96.8 | 34.3 |
| Bas1 resized wing area | 108 | 5093 | 10.7 | 231 | 3000 | 15900 | 40734 | 104 | 1.96 | 18.9 | 33.6 | 14.27 | 0.48 | 11470 | 71104 | 1.86 | 1.79 | 1769 | 95.7 | 34.3 |
| Bas1: 800NM@MP | 108 | 1482 | 10.7 | 231 | 3000 | 20000 | 40734 | 104 | 1.96 | 18.9 | 33.8 | 14.26 | 0.48 | 3929 | 67663 | 1.77 | 1.92 | 1756 | 94.7 | 34.3 |
| Bas1: 800NM@MP | 107 | 1482 | 10.7 | 231 | 3000 | 20000 | 40666 | 104 | 1.96 | 18.9 | 33.7 | 14.26 | 0.48 | 3921 | 67587 | 1.78 | 1.93 | 1755 | 94.7 | 34.3 |
| Bas1: 800NM 0.82M@35kft | 108 | 1482 | 10.7 | 243 | 3000 | 15900 | 40734 | 104 | 1.96 | 14.6 | 41.2 | 14.26 | 0.59 | 4116 | 63751 | 1.66 | 1.79 | 1739 | 95.3 | 34.3 |
| Bas1: 800NM@MA | 108 | 1482 | 11.9 | 230 | 3000 | 15900 | 40734 | 104 | 1.96 | 19.1 | 34.1 | 13.89 | 0.47 | 3782 | 63416 | 1.66 | 1.79 | 1737 | 94.6 | 34.3 |
| Bas1: OEI | 108 | 492 | 4.6 | 130 | 12000 | 15900 | 40734 | 52 | 0.98 | 20.3 | 43.8 | NaN | 0.48 | 1979 | 70613 | 1.73 | n/a | 1719 | 96.5 | 61.5 |
| Bas1 : 800NM | 108 | 1482 | 10.7 | 231 | 3000 | 15900 | 40734 | 104 | 1.96 | 18.7 | 32.1 | 14.39 | 0.46 | 3773 | 63407 | 1.66 | 1.79 | 1737 | 93.5 | 34.3 |
| **Bas2 (=Bas1)** | | | | | | | | | | | | | | | | | | | | |
| Ref2@2035 | 113 | 2222 | 10.7 | 231 | 3000 | 15900 | 41074 | 104 | 1.96 | 18.5 | 33.0 | 14.32 | 0.47 | 5369 | 65343 | 1.63 | 1.72 | 1748 | 94.4 | 34.3 |
| Bas2 resized wing area | 108 | 2222 | 10.7 | 231 | 3000 | 15900 | 40734 | 104 | 1.96 | 18.7 | 32.4 | 14.37 | 0.47 | 5302 | 64937 | 1.70 | 1.79 | 1744 | 94.0 | 34.3 |
| Bas2: 800NM@MP | 108 | 1482 | 10.7 | 231 | 3000 | 20000 | 40734 | 104 | 1.96 | 18.9 | 33.8 | 14.26 | 0.48 | 3929 | 67663 | 1.77 | 1.92 | 1756 | 94.7 | 34.3 |
| Bas2: 800NM | 108 | 1482 | 10.7 | 231 | 3000 | 15900 | 40734 | 104 | 1.96 | 18.7 | 32.1 | 14.39 | 0.46 | 3773 | 63407 | 1.66 | 1.79 | 1737 | 93.5 | 34.3 |

For the Bas1 configuration the results given in Table 6 represent the following:
- The first row (Ref1@2035) is for the SMR-Ref1 aircraft with all the 2035 technologies assumptions described above taken into account. It presents the results for the long design range (2750NM (5093 km)) mission with the design payload (15900 kg). It is found that the mOE is 41753 kg and the take-off mass (mTO) is 72565 kg, which is below MTOM (79000 kg). No violation of any criteria (as given in the last 6 columns on the right) occurs.
- The second row presents the results for the long design range (2750NM (5093 km)) mission with design payload (15900 kg), but here the wing area is reduced from 123 m$^2$ to 108 m$^2$ and the mOE is decreased accordingly by 1019 kg. Also, here no violation of any criteria occurs.
- The third row presents the mission evaluation results for the TLARs of maximum payload (20000 kg). It is found that the maximum $C_L$ in the mission occurs during landing and is just feasible and accepted ($C_L$=1.92).
- In the fourth row the wing area is slightly further reduced from 108 m$^2$ to 107 m$^2$. Again, the mission evaluation for the TLARs of maximum payload (20000 kg) is evaluated. Now it is found that the maximum $C_L$ during landing is just infeasible ($C_L$=1.93).
- In the following 3 rows the mission evaluations for the TLARs of maximum cruise speed (Mach 0.82@35 kft (10.7 km)), for the maximum altitude 39 kft (11.9 km), and for the "OEI at TO with MTOM" mission are evaluated, and no violation of any criteria occurs.
- The last row for Bas1 presents the results for the typical mission, where no violation of any criteria occurs and total mission fuel burn of 3773 kg is found.

For the SMR-Bas2 configuration the results given in Table 6 represent the following:
- The first row for Bas2 (Ref2@2035) is for the SMR-Ref2 aircraft with all the 2035 technologies assumptions described above taken into account. It presents the results for the short design range mission (1200NM (2222 km)) with design payload (15900 kg). It is found that the mOE is 41074 kg and the take-off mass (mTO) (65343 kg) is below MTOM (79000 kg). No violation of any criteria occurs.
- The second row presents the results for the short design range mission (1200NM (2222 km)) with the design payload (15900 kg), but here the wing area is reduced from 113 m$^2$ to 108 m$^2$ and the mOE is decreased accordingly by 340 kg. Also, here no violation of any criteria occurs.
- It must be noted that the constraint for maximum $C_L$ during landing ($C_L$=1.92) that was found in the maximum payload (20t) mission evaluation of the SMR-Bas1 aircraft is also valid for the SMR-Bas2 aircraft, because for both the mOE is 40734 kg. Therefore, the wing area the SMR-Bas2 aircraft can also not be reduced below 108 m$^2$.
- The last row for Bas2 presents the results for the typical mission (800 NM (1482 km)), where no violation of any criteria occurs and total mission fuel burn of 3773 kg is found.

It is found that for SMR-BAS the typical mission results do not depend on the design range requirement, because the maximum payload requirement is the sizing condition. The key results and conclusions from the evaluations of the SMR-BAS configuration are the following:
1. For the SMR-Bas1 configuration, with maximum design range of 2750 NM@35 kft (5093 km@10.7 km), the wing area can be reduced to 108 m$^2$. The critical case is the mission evaluation for the TLARs of maximum payload (20000 kg).
2. For the SMR-Bas1 configuration the total mission fuel burn for the typical mission is **3773 kg**.
3. For the SMR-Bas2 configuration, with maximum design range of 1200 NM@35 kft (2222 km@10.7 km), the wing area can be reduced to 108 m$^2$. The critical case is the mission evaluation for the TLARs of maximum payload (20000 kg).
4. For the SMR-Bas2 configuration the total mission fuel burn for the typical mission is **3773 kg**.


## 3.5 Evaluation of the 0HEP SMR aircraft (SMR-0HEP)

**SMR-0HEP BWB concept design**
For the evaluations of the SMR-0HEP configuration a Blended Wing Body (BWB) aircraft concept was adopted from an earlier study at ONERA [10], see Figure 4. In this study, comparable TLARs and EIS 2035 technology assumptions were used as in the IMOTHEP SMR evaluations. In the SMR-0HEP evaluations, the inputs for the BWB aircraft definition are taken from ONERA's BWB concept study [10], which investigates the optimized BWB aircraft concept with conventional propulsion by two CFM-LEAP-1A turbofan engines with 2035 EIS technology assumptions. Because the TLARs that were used in ONERA's BWB concept study are not exactly the same as the TLARs in IMOTHEP, the SMR-0HEP design evaluations may yield constraints that are violated. Therefore in the

SMR-0HEP evaluations, similar design variations are considered as for the REF and BAS configurations, i.e. only based on the variation of wing area.
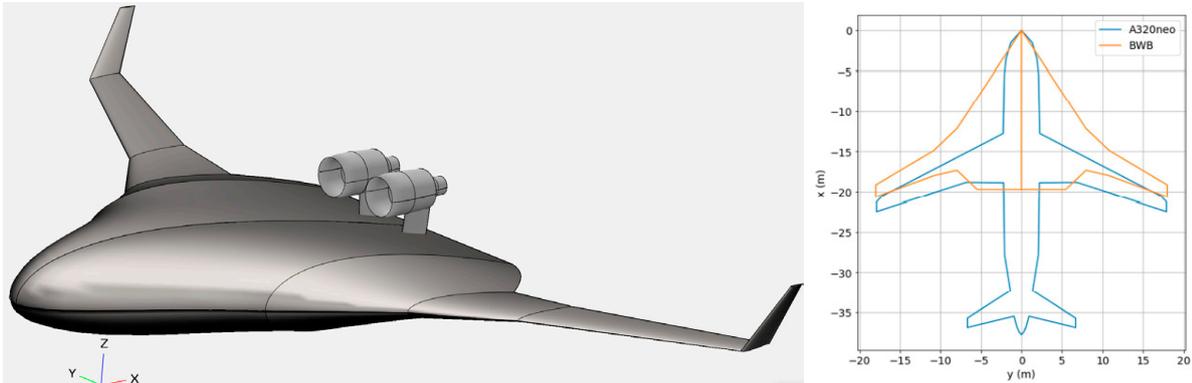


Figure 4: Illustration of the BWB SMILE aircraft geometry based on an ONERA concept study [10]. This geometry is the basis for the SMR-0HEP configuration. The figure presents the 3D shape with 2 CFM-LEAP-1A engines mounted on the rear center body (left picture) and the approximate planform (right picture, orange contour, in comparison with A320neo approximate planform in blue contour).

The main input parameters for the evaluations with NLR's MASS tool are the global aircraft sizing data like shape and global dimensions, masses, drag polars. The BWB shape and global dimensions are adopted from [10] and are illustrated in Figure 4. This BWB geometry accounts for a 150 PAX cabin layout and its projected wing area is 268.6 m$^2$. The mOE of the BWB is also adopted from [10]: mOE=36042 kg.

**SMR-0HEP aerodynamic characterization**

The drag polars for the BWB clean configuration without engines have been evaluated by Reynolds-averaged Navier-Stokes (RANS) analyses. The computational fluid dynamics (CFD) software ENSOLV [11] of NLR has been used for the calculations for the relevant flight conditions. These conditions comprise a number of speed-altitude combinations that are representative for the considered mission (Figure 5). These conditions are expressed by the Mach number and altitude in ISA. The resulting drag polars are depicted below in Figure 5. The drag polar data comprise the aircraft level lift coefficient ($C_L$) and drag coefficient ($C_D$) versus the angle of attack ($\alpha$). The lift and drag coefficients have been evaluated for sequences of $\alpha$. The data above certain maximum values of $\alpha$ have been excluded from further processing. These maximum values of $\alpha$ are indicated by the grey circles in Figure 5. The maximum $C_L$ values in take-off and landing conditions on sea level are about 0.66. Of course, high-lift systems may well increase these maximum $C_L$ values, but this is currently beyond the scope of this study. Furthermore, the BWB clean body has been aerodynamically designed for optimal cruise operation at around Mach 0.78 on 41 kft (12.6 km) altitude and $C_L$ of about 0.27. Therefore this cruise condition is considered for the design range and the typical range missions of the SMR-0HEP configuration.
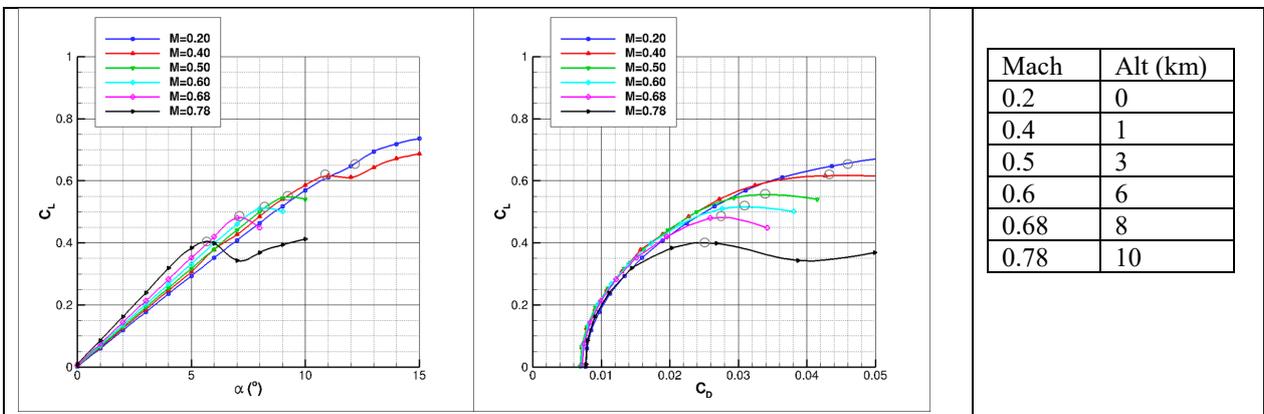


| Mach | Alt (km) |
|------|----------|
| 0.2  | 0        |
| 0.4  | 1        |
| 0.5  | 3        |
| 0.6  | 6        |
| 0.68 | 8        |
| 0.78 | 10       |

Figure 5: Illustration of the drag polars for the BWB clean configuration (left), showing $C_L$ versus $\alpha$ (the angle of attack) and versus $C_D$ (the aircraft level drag coefficient), and based on a reference area $S_{ref} = 268.6$ m². Also, the maximum values of $\alpha$ are indicated by the grey circles, the data for higher values of $\alpha$ have been excluded from further processing. The speed-altitude combinations that are representative for the considered mission are listed on the right, expressed by the Mach number and altitude in ISA.

For an efficient incorporation of the drag polars in the MASS tool, a map of $C_D$ as function of $C_L$ and Mach number is generated using surrogate modelling methods. Polynomial methods of various orders and artificial neural networks (ANN) with various numbers of hidden nodes are evaluated. A feedforward ANN [12] with 9 hidden nodes is found to give the most accurate representation of the drag polar data and is therefore used in the MASS evaluations. On top of this drag polar representation of the BWB clean configuration, extra parasite drag contributions are added for the engines. A fixed value of 11.9 drag counts was estimated in [10] and has been used here too.

**SMR-0HEP design evaluations**

The SMR-0HEP configuration represents the radical airframe design of the BWB, but with conventional turbofan propulsion. Just like for SMR-BAS, also here the GSP based implementation of the CFM-LEAP-1A in MASS with 10% TSFC reduction due to 2035 technologies is used as turbofan model. Just like for the SMR-BAS, also for the SMR-0HEP two separate 0HEP configurations are considered for the different design ranges: the 0Hep1 configuration for the long design range of 2750NM (TLARs 3.a in the list above) and the 0Hep2 configuration for the short design range of 1200NM (2222 km) (TLARs 3.b in the list above). For both the 0Hep1 and 0Hep2 configurations, the re-design of the wing area is performed by checking the design constraints described above for the various missions as prescribed by the TLARs. Both the 0Hep1 and 0Hep2 configurations are used for the assessment of the SMR-RAD configuration.

Table 7: Design evaluations overview for SMR-0HEP. Explanations of missions: 800NM@MP means: 800 NM (2222 km) mission at maximum payload of 20000 kg. 800NM@MA means: 800 NM (2222 km) mission at maximum altitude of 11.9 km. OEI means: one engine inoperative mission including take-off + climb until 15 kft at climb rate > 100 ft/min. The yellow cells indicate the key variables that are set for each evaluation. The green cells indicate the key outcome of an evaluation, i.e. the typical mission fuel burn. The red cells indicate the violation of constraints in an evaluation. The orange cells indicate an active constraint in an evaluation. The grey cells indicate that the data are not relevant.

| 0HEP Variants and missions | Wing area [m2] | Range [km] | Cruise altitude [km] | Cruise TAS [m/s] | Reserve fuel [kg] | payload [kg] | mOE [kg] | average total PTO [kW] | average total bleed (kg/s) | fuel burn [kg] | mTO [kg] | TakeOff max(CL) | Landing max CL | max TT4 [K] | max N1 [%] | TTC [min] 1.5kft - 35kft |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| **0Hep1** | | | | | | | | | | | | | | | | |
| 0Hep1: 800NM@41kft, MP | 269 | 1482 | 12.6 | 230 | 3000 | 20000 | 36042 | 104 | 1.96 | 3754 | 62796 | 0.59 | 0.71 | 1743 | 97.6 | 35.0 |
| 0Hep1: 800NM@41kft, MP, resized wing area | 297 | 1482 | 12.6 | 230 | 3000 | 20000 | 37934 | 104 | 1.96 | 3889 | 64823 | 0.55 | 0.66 | 1756 | 98.3 | 35.0 |
| 0Hep1: 800NM@41kft@0.82M | 297 | 1482 | 12.6 | 242 | 3000 | 15900 | 37934 | 104 | 1.96 | 3764 | 60598 | 0.51 | 0.62 | 1734 | 98.2 | 34.9 |
| 0Hep1: 2750NM@41kft | 297 | 5093 | 12.6 | 230 | 3000 | 15900 | 37934 | 104 | 1.96 | 10391 | 67225 | 0.57 | 0.62 | 1767 | 99.4 | 35.0 |
| 0Hep1: OEI | 297 | 491 | n/a | n/a | 11500 | 15900 | 37934 | 52 | 0.98 | 1880 | 67215 | 0.68 | n/a | 1825 | 100.9 | n/a |
| 0Hep1: typical mission 800NM@41kft | 297 | 1482 | 12.6 | 230 | 3000 | 15900 | 37934 | 104 | 1.96 | 3737 | 60571 | 0.51 | 0.62 | 1734 | 96.7 | 35.0 |
| | | | | | | | | | | | | | | | | |
| **0Hep2** | | | | | | | | | | | | | | | | |
| 0Hep2: 1200NM@41kft | 297 | 2222 | 12.6 | 230 | 3000 | 15900 | 37934 | 104 | 1.96 | 5057 | 61891 | 0.52 | 0.62 | 1741 | 97.2 | 35.0 |
| 0Hep2: typical mission 800NM@41kft | 297 | 1482 | 12.6 | 230 | 3000 | 15900 | 37934 | 104 | 1.96 | 3737 | 60571 | 0.51 | 0.62 | 1734 | 96.7 | 35.0 |

For the 0Hep1 configuration the results given in Table 7 represent:

- The first row presents the mission evaluation results for the TLARs of maximum payload (20000 kg). It is found that the maximum $C_L$ in the mission occurs during landing and is infeasible ($C_L$=0.71).
- In the second row the wing area is increased from 269 m² to 297 m² and the operating empty mass (mOE) is increased accordingly by 1892 kg. Again, the mission evaluation for the TLARs of maximum payload (20000 kg) is evaluated. Now it is found that the maximum $C_L$ during landing is just feasible ($C_L$=0.66).
- In the following 3 rows the mission evaluations for the TLARs of maximum cruise speed (Mach 0.82@41 kft (12.6 km)), for the long design range (2750 NM (5093 km), 15900 kg design payload), and for the "OEI at TO with MTOM" mission are evaluated, and no violation of any criteria occurs.

- The last row for 0Hep1 presents the results for the typical mission, where no violation of any criteria occurs and total mission fuel burn of 3737 kg is found.

For the 0Hep2 configuration the results are given in Table 7.

- Obviously, the same increased wing area (297 m$^2$) as for the 0Hep1 must be applied to 0Hep2, because the 0Hep2 must also be feasible for the maximum payload (20000 kg) mission evaluation. And obviously, the maximum cruise speed (Mach 0.82@41 kft (12.6 km)) mission and the "OEI at TO with MTOM" mission are also feasible for that wing area, as was shown for the 0Hep1.
- The first row presents for the 0Hep2 with increased wing area (297 m$^2$) the mission evaluation for the short design range (1200 NM (2222 km), 15900 kg design payload), and no violation of any criteria occurs.
- The last row for 0Hep2 presents the results for the typical mission, where no violation of any criteria occurs and total mission fuel burn of 3737 kg is found.

Just like for SMR-BAS, also for SMR-0HEP it is found that the typical mission results do not depend on the design range requirement, because the maximum payload requirement is the sizing condition. The key results and conclusions from the evaluations of the SMR-0HEP configuration are the following:

1. For the SMR-0Hep1 configuration, with maximum design range of 2750 NM@41 kft (5093 km@12.6 km), the wing area must be increased to 297 m$^2$. The critical case is the mission evaluation for the TLARs of maximum payload (20000 kg).
2. For the SMR-0Hep1 configuration the total mission fuel burn for the typical mission is **3737 kg**.
3. For the SMR-0Hep2 configuration the total mission fuel burn for the typical mission is **3737 kg**.

## 3.6 Evaluation of the radical HEP SMR aircraft (SMR-RAD)

**SMR-RAD BWB concept design**

For the SMR-RAD initial configuration the same underlying assumptions are used as for the SMR-0HEP configuration. The BWB airframe with the updated wing area of 297 m$^2$ and updated mOE of 37934 kg that was found for the 0HEP is adopted here. This is because a sensible comparison between SMR-RAD and SMR-0HEP must be made. For SMR-RAD, only the propulsion system is changed from turbofan (i.e. the 2 CFM-LEAP-1A engines with 2035 EIS technology assumptions for 0HEP) to a HEP architecture. Of course, this change from turbofan to HEP powertrain implies changes in mass and energetic efficiencies of the propulsion system, which will be addressed in this section.

**Potential HEP architectures for SMR-RAD**

HEP can be implemented in various ways into the propulsion powertrain. Common HEP architectures for aircraft have been proposed by NASA [13], see Figure 6.
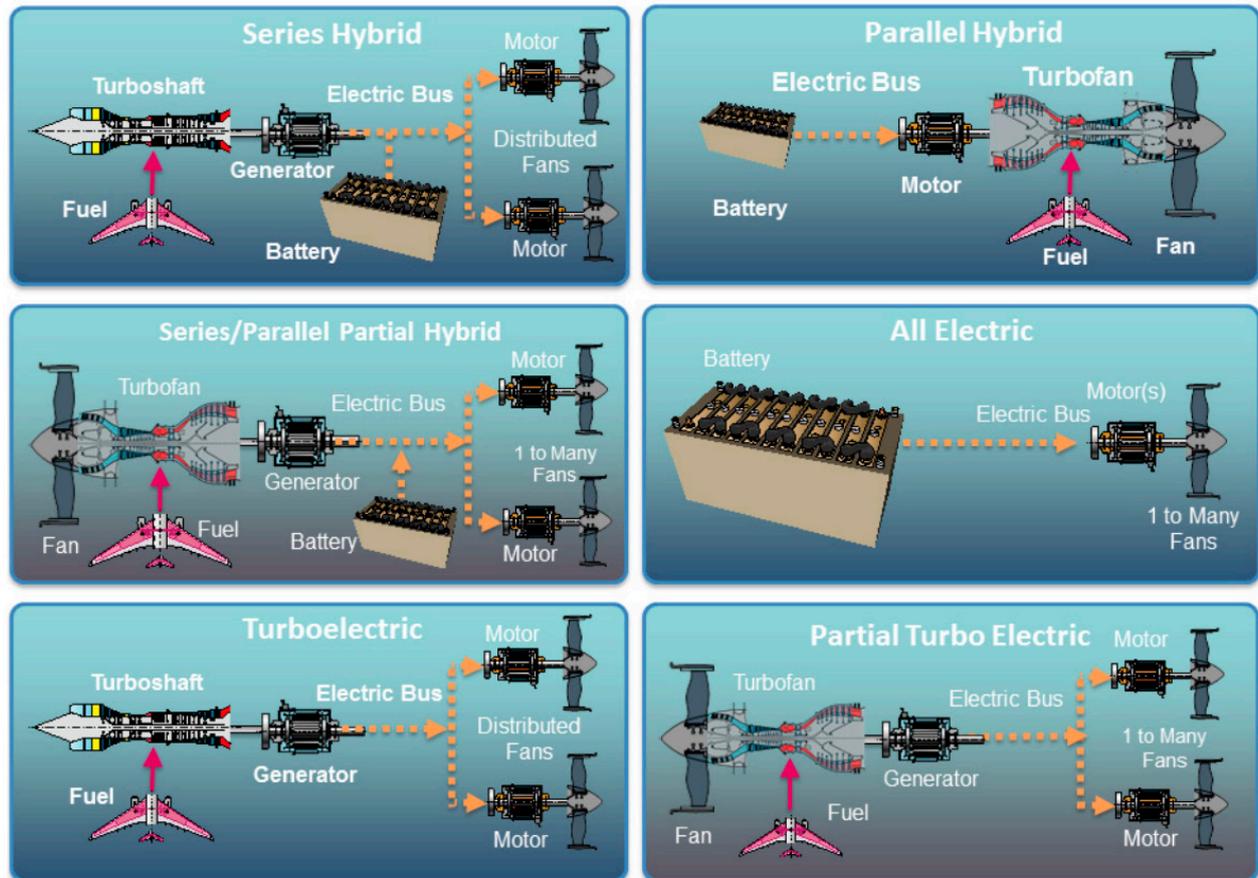
Figure 6: Common HEP architectures for aircraft as proposed by NASA [13].

**Series hybrid architecture for SMR-RAD**

As indicated in Figure 6, the series hybrid architecture comprises a turboshaft combustion engine that powers an electric generator. This generator, together with an electric battery pack, powers electric motors that drive one or multiple fans. If we look at the IMOTHEP project targets, i.e. a 40% reduction in fuel consumption for typical missions of the SMR aircraft, then this series hybrid architecture has not much potential in comparison to the other HEP architectures. In particular, in comparison to the turboelectric architecture, which is the same as the series hybrid architecture except for the battery pack. This is because the battery pack has a much lower specific energy and specific power value than the electric generator, turboshaft engine and kerosene fuel for this SMR configuration. Typical energy values for the typical mission (800 NM) of the SMR aircraft are based on the power train. The mission fuel burn is about 4 t; gas turbine mass is about 2 t; generator mass is about 1 t. The overall efficiency of gas turbine and generator is about 40%. This yields an average overall specific energy for the typical mission of about 12 kWh/kg * 4000 kg * 0.4 / 7000 kg =~ 2700 Wh/kg. This is almost 11 times higher than the specific energy of present state of the art lithium ion battery cells (about 250 Wh/kg). Hence, in the series hybrid architecture, any kg of fuel that would be replaced by batteries will add at least about 11 kg of battery mass. For the typical mission, this 11 kg additional battery mass would require approximately 0.4 kg additional fuel, which would yield an effective fuel saving of 0.6 kg fuel for each 11kg battery mass that would be added, resulting in an added battery mass of about 18 kg per kg of effective fuel saving. In other words, 100 kg effective fuel saving would require 1800 kg of added battery mass. However, this added battery mass is so substantial that snowball effects due to increasing system- and structural masses must be taken into account and very quickly the mOE and mTO will be exceeding their limits. This architecture, in comparison to the turbo electric architecture, is therefore expected to provide only very limited reduction in typical mission fuel burn but at the cost of a large mass penalty, and substantial additional cost and energy for the battery. Therefore it can be concluded that for the IMOTHEP project targets the series hybrid is not a viable architecture for the IMOTHEP SMR-RAD configuration.

**Parallel hybrid architectures for SMR-RAD**

As indicated in Figure 6, the parallel hybrid architecture includes a battery-electric powertrain in parallel with a fuel-powered gas turbine powertrain, which together drive one or multiple fan shafts. The optimum gas turbine

powertrain for SMR aircraft is typically a twin-turbofan engine configuration: more than one turbofan because of redundancy (safety in engine failure case) and less than 3 turbofans because of highest efficiency (larger is better). In the parallel hybrid architecture the electric motor support allows to down-size the gas turbine (core) of the turbofan if this support is applied in the critical flight phase which is top-of-climb for the turbofan. However, for twin engine aircraft another critical flight phase is a one-engine-inoperative (OEI) failure condition. If this occurs in a missed approach procedure at the end of a mission, then batteries are empty. The remaining single down-sized gas turbine must be capable to safely complete this procedure without the electric motor support (because batteries are empty). TBD: motivate why this yields no potential? Add figure from previous slides? Therefore the potential for down-sizing the turbofan is expected to be very limited. Moreover, potential gain in cruise efficiency for this down-sized turbofan is expected to be very limited because turbofan engines tend to get more efficient as they get larger. Hence this parallel hybrid architecture is also not the first to consider in this study.

**Series/Parallel partial hybrid architectures for SMR-RAD**
As indicated in Figure 6, the series/parallel partial hybrid architecture is equivalent to the series hybrid architecture, except for the turboshaft engine that is replaced by a turbofan engine. The only difference is that a fan is added to the turboshaft with an almost loss-less interface (i.e. a direct drive shaft connection or a gearbox). This fan will provide a part of the thrust, besides the thrust from the electric driven fans. The advantage of the turbofan is the almost loss-less power transfer from gas turbine (core) to the fan. But the feasibility of a turbofan instead of a turboshaft engine does depend on installation aspects such as sufficient space to install a large fan on the engine, including free flow paths for inlet and outlet of the fan duct. The advantage of the electric driven fans is the scalability and flexibility to increase fan area and distribute propulsors over the airframe, just like in the turbo-electric architecture. Because of the similarity with the series hybrid architecture and for similar reasons, also for this series/parallel partial hybrid architecture it is unlikely that the battery could bring any benefit in fuel consumption in comparison to the turbo-electric architecture. Therefore this series/parallel partial hybrid architecture is also not the first to consider in this study.

**All-electric architectures for SMR-RAD**
As indicated in Figure 6, the all-electric architecture includes only a battery-electric powertrain that drives one or more fans. Obviously, from the considerations on batteries given above for the series hybrid architecture, it can be concluded that this all-electric architecture is not viable for SMR aircraft. As indicated for the series hybrid architecture, any kg of fuel that would be replaced by batteries will add at least about 11 kg of battery mass. The typical mission fuel burn is about 4 t, which would require 44 t of batteries to replace the complete fuel mass. Of course, the battery-electric power train would eliminate the gas turbine and electric generator, which would reduce the mass by about 3 t. But still, the overall mass gain of about 41 t would yield substantial further mass increase of airframe and systems due to the snowball effect. Hence it is concluded that the all-electric architecture is not a viable architecture for the IMOTHEP SMR-RAD configuration.

**Partial turbo-electric architectures for SMR-RAD**
As indicated in Figure 6, the partial turbo-electric architecture contains a turbofan engine and adds to that an electric powertrain that drives one or more fans. The Figure 7 below shows a simplified assessment of the fuel efficiency of such a partial turbo-electric architecture for a BWB aircraft sized for the considered SMR mission. The turbofan engines are based on a CFM-LEAP-1A engine model, where the effects of mechanical shaft power takeoff (PTO) for the electric generators are included. It is illustrated that electric power takeoff from the turbofan engine can only be beneficially used for thrust if the propulsive efficiency of the electric fans (etaProp) is higher than 90%. If a 95% propulsive efficiency is assumed, then only about 1% reduction of typical mission fuel consumption can be achieved if the electric power off-take from the turbofan is about 50% or more of the total power of the turbofan. In all cases, an overall efficiency of the electric powertrain (etaEM) of 90% is assumed. Hence it is concluded that the partial turbo-electric architecture is not a viable architecture for the IMOTHEP SMR-RAD configuration.
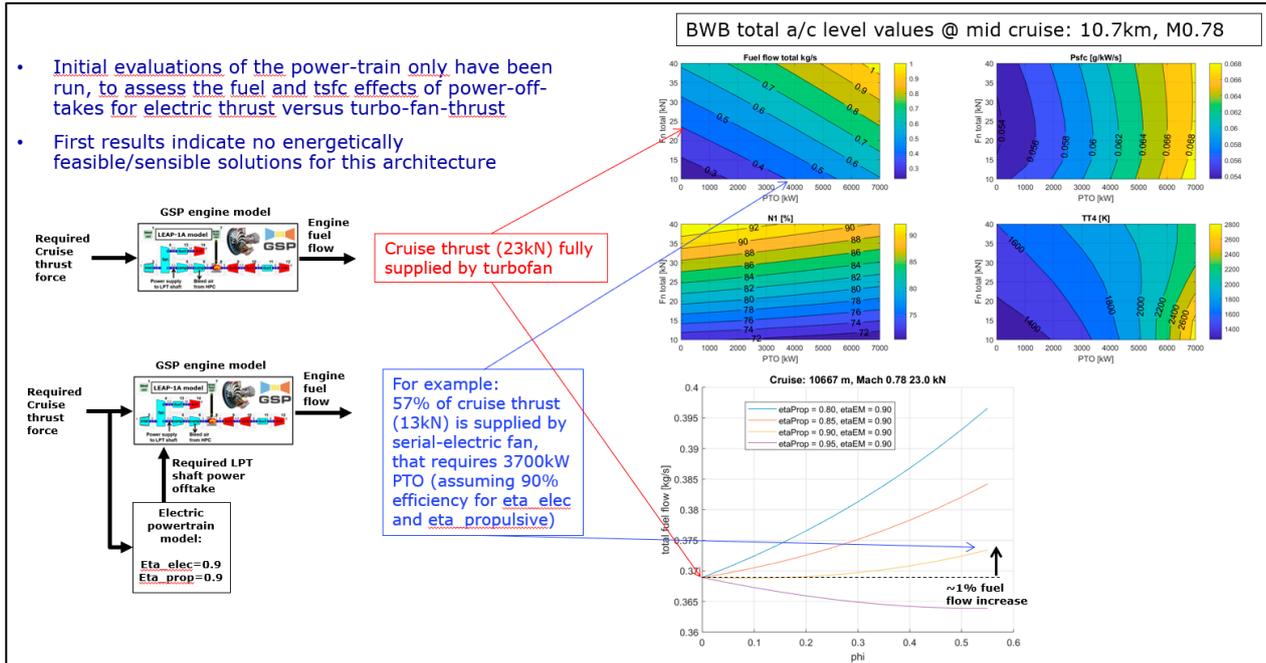
Figure 7: Simplified assessment of the fuel efficiency of the partial turbo-electric architecture for a BWB aircraft sized for the considered SMR mission.
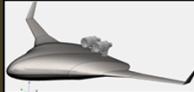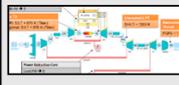
**Turbo-electric architectures for SMR-RAD**

As indicated in Figure 6, the turbo-electric architecture comprises a turboshaft combustion engine that powers an electric generator. This generator powers electric motors that drive one or multiple fans. As already mentioned above for the series hybrid architecture, the turbo-electric architecture has potential for an energy-efficient architecture because it does allow for large fan area by distributed propulsion but it does not have the burden of excessive battery mass. Therefore the turbo-electric architecture will be further investigated in the following sections.

**Propulsion system mass estimations**

The mass changes due to the update from turbofan propulsion of the 0HEP configuration to turbo-electric propulsion of the RAD configuration are partly estimated and partly resulting from the propulsion system sizing that is made with NLR's MASS tool. An overview of the components considered and their mass estimates is given in the Table 8. For the considered SMR-RAD turbo-electric configuration the 2 turbofan engines are replaced by turboshaft engines and electric generators, for power generation, and ducted electric fans for thrust generation. Each of these turboshaft engines and ducted fans requires a pylon and nacelle for proper installation on the BWB airframe. The location of each of these engines and fans is currently assumed extrados on the rear centre body of the BWB, symmetric in its centre-vertical symmetry plane, as illustrated in Figure 9. This location may yield benefits in terms of noise shielding and boundary layer ingestion (BLI) for the fans, but also may have disadvantages for maintenance or thrust vectoring. The positioning of these components on the BWB airframe is still under further investigation and not yet finally converged. Therefore these noise shielding and BLI aspects are beyond the scope of this paper. This paper is mainly focussed on the conceptual design and sizing of the propulsion components and the TE powertrain.

Table 8: Overview of the mass changes due to the update from turbofan propulsion of the 0HEP configuration to turbo-electric propulsion of the RAD configuration.



The mass estimation of the turbofan is based on actual data of the CFM-LEAP-1A engine, with a wet engine mass of 2990 kg [7]. For A320neo, the nacelle- and auxiliary systems have a mass of about 1200 kg [9] and the pylon mass is about 650 kg [15]. For 0HEP we take into account the mass reductions due to the 2035 EIS technology assumption (similar to BAS, as given in Table 5, section 3.4), yielding 2879 kg for the engine and 625 kg for the pylon.

For the SMR-RAD propulsion system, the masses of the turboshaft engine and the power turbine are estimated at about 700 kg and 300 kg, respectively. The masses of the nacelle and pylon components are estimated in the following way for the SMR-RAD configuration. The pylon mass is dominated by its structural sizing for transfer of thrust forces. Because the total thrust force on aircraft level for SMR-RAD is not very different from SMR-0HEP, the total mass of all pylons on SMR-RAD is assumed to be equal to the total mass of all pylons on SMR-0HEP, i.e. 1250 kg. The nacelle mass of a ducted electric fan, of the same size as the CFM-LEAP-1A nacelle with a fan diameter of about 2 m, is assumed to be one third of the mass of the CFM-LEAP-1A nacelle, i.e. 400 kg. This is because of the much simpler construction and system installation for the ducted electric fan. For example the thrust reverser, with a mass of about 400 kg for one CFM-LEAP-1A nacelle, is not needed in the ducted fans because of assumed reversed rotation capability of the electric fans. The ducted electric fans are sized on the basis of the fan area, which is an output of the ducted fan model. The nacelle mass is assumed to proportional to the nacelle wetted area, and therefore also proportional to the ducted fan diameter because the nacelle length is assumed to be constant. Consequently for the ducted electric fan nacelle, the diameter-specific nacelle mass is 400 kg divided by 2 m fan diameter yielding 200 kg/m.

For simplicity, the same structures for pylon and nacelle are assumed for the ducted fans and for the turbo-generators (i.e.: the assembly of turboshaft engine, power turbine and electric generator). For the SMR-RAD pylons this implies that the total pylon mass on aircraft level, i.e 1250 kg, comprises n+2 pylons for n ducted fans and 2 turbo-generators. So for example for 8 ducted fans, the pylon mass is 1250 kg for 8+2=10 pylons, i.e. 125 kg per pylon. For the SMR-RAD nacelles for the ducted fans and turbo-generators the mass equals the nacelle diameter times the diameter-specific nacelle mass. The turbo-generators are assumed to have a nacelle of 1 m diameter. So for example for 8 ducted fans of 1.33 m diameter the nacelle mass is 200 kg/m times 1.33 m is 266 kg, and for the turbo-generators of 1 m diameter the nacelle mass is 200 kg/m times 1.0 m is 200 kg, so total nacelle mass on aircraft level is 8*266+2*200=2528 kg.

The mass of the fan rotor (i.e. 18 carbon composite blades and metallic rotor hub) of the CFM-LEAP-1A is estimated at 100 kg for a fan area of about 3.1 m$^2$. The fan rotor of the ducted electric fan is also assumed to consist of 18 composite fan blades and metallic rotor hub, at a similar areal mass of about 32 kg per m$^2$ fan area. The mass estimation of the electric powertrain components, like generators, power electronics and electric motors, is handled internally by the MASS tool in relation with the thrust requirements during the mission.

**Energetic efficiencies of the propulsion system**

The energetic efficiencies of the CFM-LEAP-1A turbofan engines in the 0HEP configuration are incorporated in the GSP engine model. For the turbo-electric propulsion system, the energetic efficiencies of the powertrain components are incorporated in the turbo-electric component models. The main turbo-electric components are the turboshaft engine with power turbine, the electric generator with AC-DC converter, the electric distribution system with power cables, switches and buses, the electric motors with inverters and power electronics, the ducted fans. For each of these main components a more or less elaborate modelling approach is followed, as illustrated in Figure 8. The turboshaft engine is included as a gas turbine cycle model, developed with DLR's GTLab environment [16], and dedicated to the required approximate power levels. The gas turbine cycle model predicts the fuel mass flow ($\dot{m}_{fuel}$) and the HPT inlet temperature ($T_{T4}$) as a function of the required power, altitude and Mach number during the mission. The constraint for this turboshaft engine is only on HPT inlet temperature: $T_{T4}$<1900 K. The turboshaft engine drives a specific power turbine that is dedicated to power the electric generator through a direct drive shaft. The assembly of turboshaft, power turbine and electric generator constitutes the complete turbo-generator component. The SMR-RAD powertrain contains two of such turbo-generators in parallel for redundancy. The electric components in the powertrain comprise the generator with AC-DC converter, the electric distribution system and the electric motors with DC-AC inverters. These electric components are included by simplified models based on 2035 technology level estimates of specific power and efficiency values [14]. Also, the required cooling system (CS) equipment sizing is included in a similar simplified way. The assumed values are listen in Table 9. The pessimistic values listen in Table 9 are used in the initial evaluations of SMR-RAD. The optimistic values listen in Table 9 are used in the evaluations of the design variations for SMR-RAD that are described in the following section.

Table 9: Turbo-electric power train 2035 technology assumptions [14] for SMR-RAD. The pessimistic assumptions are rather conservative and close to current state of the art numbers. The pessimistic assumption are used in the initial SMR-RAD evaluations. The optimistic assumptions are estimates based on various public sources. The effects of these optimistic assumptions are assessed in the SMR-RAD design variations as described in the following section.

| Parameter | Pessimistic | Optimistic |
|---|---|---|
| Electric motor specific power [kW/kg] | 11 | 17 |
| Electric motor power factor | 0.95 | 0.95 |
| Electric motor efficiency | 0.96 | 0.98 |
| Converter/inverter specific power [kVA/kg] | 20 | 30 |
| Converter/inverter efficiency | 0.99 | 0.99 |
| Cooling system specific power [kW/kg] | 0.68 | 0.68 |
| Generator specific power [kW/kg] | 20 | 20 |
| Generator efficiency | 0.98 | 0.98 |

The ducted fans are included by a simplified model based on polytropic pressure-duct equations [14]. With this model, the fan pressure ratio ($FPR$), the ducted fan shaft power ($P_{shaft,DF}$) and the propulsive efficiency ($\eta_{prop}$) can be predicted as a function of true air speed, altitude, net thrust force and ducted fan exhaust area ($V_{TAS}, h, F_{n,DF}, A_{exh}$). The total thrust force on aircraft level $F_{n,AC}$ follows from the BWB aircraft model for each point in the mission. This BWB aircraft model is described in the previous section for the SMR-0HEP configuration. The required ducted fan shaft power $P_{shaft,DF}$ shall be provided by the electric motor and the other electric components, for which the power values are summarized by the symbol for electric component power $P_{EC}$. The resulting power that is required by the electric generator is fed to the turboshaft model, together with the aircraft altitude and Mach number ($P_{shaft,TS}, h, M$). An overview of the turbo-electric propulsion system for SMR-RAD with all its components and variables is illustrated in Figure 8.

$$P_{shaft,TS}, h, M \qquad P_{EC} \qquad V_{TAS}, h, F_{n,DF} \qquad V_{TAS}, h, \gamma$$

Turboshaft model: Based on dedicated gas turbine cycle model and power-turbine

Electric components models: Electric generator model / ACDC converter model / Electric distribution model / DCAC inverter model / Electric motor model / Simplified models, based on specific powers, efficiencies

Ducted fan model: Based on polytropic pressure-duct equations

BWB a/c model: Based on masses and drag polars

Mission model — Source: Tan, 2018

Fuel consumption

$$m_{EC} = f_{EC}(P_{EC}, \eta_{EC},)$$

$$F_{n,AC} = f_{AC}(V_{TAS}, h, \gamma, m_{AC}, C_L, C_D)$$

$$\begin{pmatrix} \dot{m}_{fuel} \\ T_{T4} \end{pmatrix} = f_{TS}(P_{shaft,TS}, h, M)$$

$$\begin{pmatrix} FPR \\ P_{shaft,DF} \\ \eta_{prop} \end{pmatrix} = f_{DF}(V_{TAS}, h, F_{n,DF}, A_{exh})$$
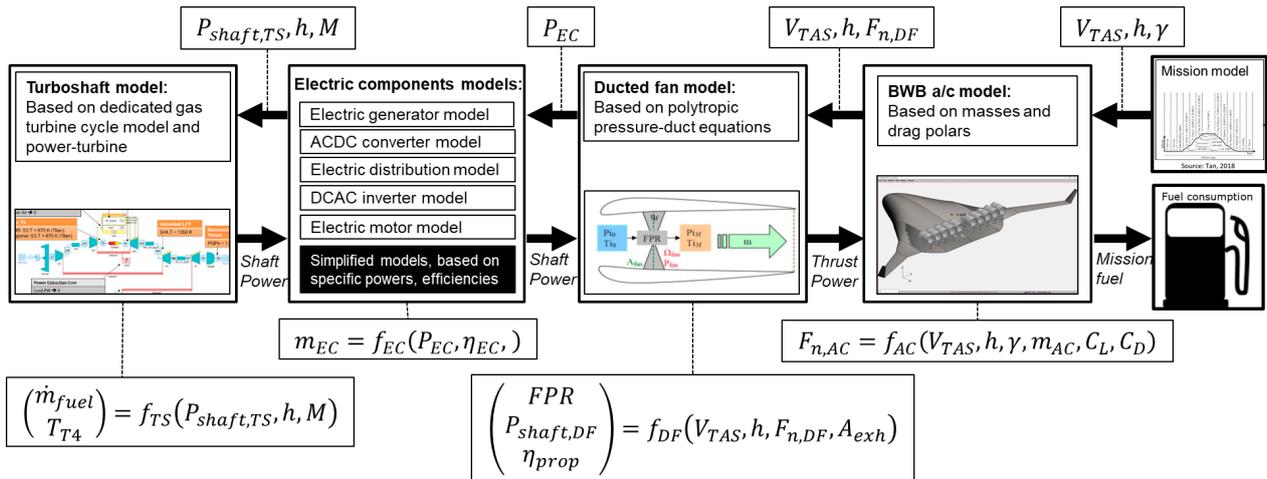
Figure 8: Illustration of the turbo-electric propulsion system for SMR-RAD, with the main powertrain components incorporated as more or less elaborate component models.

## SMR-RAD design evaluations

The SMR-RAD design approach is different from the design for all the previous configurations. In the REF, BAS and 0HEP configurations only the wing area was considered as the global design variable, which was applied only to update the configurations such that all TLARs are fulfilled. The SMR-RAD design aims at optimizing the turbo-electric propulsion system, such that all TLARs are fulfilled and the typical mission fuel consumption is minimized. The design variables considered in this study for SMR-RAD are the number, location and size of the ducted fans. In this way it is intended to maximize propulsive efficiency by maximizing total fan area. But the size and number of the propulsors will have limitations related to installation space and mass and drag of the fan casings, ducts and pylons. Another driver here may be the exploitation of boundary layer ingestion (BLI) benefits by installing the ducted fans extrados on the centre body. A low-fidelity modelling of BLI in ducted fans is also under investigation in the IMOTHEP project [17]. However, the modelling and analyses for incorporating the BLI effects are not yet completed and therefore the BLI benefits are out of scope for this paper.

An initial investigation of SMR-RAD is made for 8 ducted electric fans, as illustrated in Figure 9. This is a rather arbitrary choice, but it is made to start the investigations and make a first assessment of the feasibility of the chosen SMR-RAD concept.

Table 10: Design evaluations overview for the initial SMR-RAD with 8 ducted fans. Explanations of missions: 800NM@MP means: 800 NM (2222 km) mission at maximum payload of 20000 kg. OEI means: one engine inoperative mission including take-off + climb until 15 kft at climb rate > 100 ft/min. The yellow cells indicate the key variables that are set for each evaluation. The green cells indicate the key outcome of an evaluation, i.e. the typical mission fuel burn. The red cells indicate the violation of constraints in an evaluation. The orange cells indicate an active constraint in an evaluation. The grey cells indicate that the data are not relevant.

| RAD Variants and missions | Wing area [m2] | Range [km] | Cruise altitude [km] | Cruise TAS [m/s] | Reserve fuel [kg] | payload [kg] | mOE [kg] | average total PTO [kW] | average total bleed (kg/s) | fuel burn [kg] | mTO [kg] | TakeOff max(CL) | Landing max CL | max TT4 [K] | max Pshaft [MW] | TTC [min] 1.5kft - 35kft |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| **Rad1** | | | | | | | | | | | | | | | | |
| 0Hep1 airframe+8 dfans: 2750NM@41kft | 297 | 5093 | 12.6 | 230 | 3000 | 15900 | 40638 | 104 | 1.96 | 10087 | 69625 | 0.68 | 0.65 | 1713 | 19.0 | 35.0 |
| Rad1: 800NM@41kft, MP | 297 | 1482 | 12.6 | 230 | 3000 | 20000 | 40638 | 104 | 1.96 | 3827 | 67464 | 0.66 | 0.69 | 1694 | 18.4 | 35.0 |
| Rad1: 800NM@41kft, MP, resized wing area | 320 | 1482 | 12.6 | 230 | 3000 | 20000 | 42351 | 104 | 1.96 | 3955 | 69306 | 0.63 | 0.66 | 1720 | 19.1 | 35.0 |
| Rad1: 800NM@41kft@0.82M | 320 | 1482 | 12.6 | 242 | 3000 | 15900 | 42351 | 104 | 1.96 | 3851 | 65102 | 0.59 | 0.62 | 1719 | 18.1 | 34.9 |
| Rad1: 2750NM@41kft | 320 | 5093 | 12.6 | 230 | 3000 | 15900 | 42351 | 104 | 1.96 | 10425 | 71676 | 0.65 | 0.62 | 1731 | 19.6 | 35.0 |
| Rad1: OEI | 320 | 491 | n/a | n/a | 11600 | 15900 | 42351 | 52 | 0.98 | 1838 | 71689 | 0.67 | n/a | 1823 | 13.1 | n/a |
| Rad1: 800NM@41kft | 320 | 1482 | 12.6 | 230 | 3000 | 15900 | 42351 | 104 | 1.96 | 3798 | 65049 | 0.59 | 0.62 | 1701 | 18.1 | 35.0 |
| **Rad2** | | | | | | | | | | | | | | | | |

| 0Hep2 airframe+8 dfans: 2750NM@41kft | 297 | 2222 | 12.6 | 230 | 3000 | 15900 | 40292 | 104 | 1.96 | 4908 | 64100 | 0.62 | 0.65 | 1678 | 17.7 | 35.0 |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| Rad2: 800NM@41kft, MP | 297 | 1482 | 12.6 | 230 | 3000 | 20000 | 40292 | 104 | 1.96 | 3811 | 67104 | 0.65 | 0.69 | 1692 | 18.4 | 35.0 |
| Rad2: 800NM@41kft, MP, resized wing area | 317 | 1482 | 12.6 | 230 | 3000 | 20000 | 41798 | 104 | 1.96 | 3925 | 68723 | 0.63 | 0.66 | 1715 | 18.9 | 35.0 |
| Rad2: 800NM@41kft@0.82M | 317 | 1482 | 12.6 | 242 | 3000 | 15900 | 41798 | 104 | 1.96 | 3821 | 64519 | 0.59 | 0.62 | 1713 | 18.0 | 34.9 |
| Rad2: 1200NM@41kft | 317 | 2222 | 12.6 | 230 | 3000 | 15900 | 41798 | 104 | 1.96 | 5067 | 65764 | 0.60 | 0.62 | 1702 | 18.2 | 35.0 |
| Rad2: OEI | 317 | 491 | n/a | n/a | 9300 | 15900 | 41798 | 52 | 0.98 | 1784 | 68782 | 0.65 | n/a | 1807 | 12.6 | n/a |
| Rad2: 800NM@41kft | 317 | 1482 | 12.6 | 230 | 3000 | 15900 | 41798 | 104 | 1.96 | 3769 | 64467 | 0.59 | 0.62 | 1696 | 17.9 | 35.0 |

For the Rad1 configuration the results given in Table 10 represent:

- The first row presents the mission evaluation results for the long design range (2750 NM (5093 km), 15900 kg design payload). The sizing of the TE powertrain leads to a mOE of 40638 kg.
- The 2nd row presents the mission evaluation results for the TLARs of maximum payload (20000 kg). It is found that the maximum $C_L$ in the mission occurs during landing and is infeasible ($C_L$=0.69).
- In the 3rd row the wing area is increased from 297 m$^2$ to 320 m$^2$ and the operating empty mass (mOE) is increased accordingly by 1713 kg. Again, the mission evaluation for the TLARs of maximum payload (20000 kg) is evaluated. Now it is found that the maximum $C_L$ during landing is just feasible ($C_L$=0.66).
- In the following 3 rows the mission evaluations for the high cruise speed (Mach 0.82), for the long design range (2750 NM (5093 km), 15900 kg design payload), and for the "OEI at TO with MTOM" mission are evaluated, and no violation of any criteria occurs.
- The last row for Rad1 presents the results for the typical mission, where no violation of any criteria occurs and total mission fuel burn of 3798 kg is found.

For the Rad2 configuration the results given in Table 10 represent:

- The first row presents the mission evaluation results for the short design range (1200 NM (2222 km), 15900 kg design payload). The sizing of the TE powertrain leads to a mOE of 40292 kg.
- The 2nd row presents the mission evaluation results for the TLARs of maximum payload (20000 kg). It is found that the maximum $C_L$ in the mission occurs during landing and is infeasible ($C_L$=0.69).
- In the 3rd row the wing area is increased from 297 m$^2$ to 317 m$^2$ and the operating empty mass (mOE) is increased accordingly by 1506 kg. Again, the mission evaluation for the TLARs of maximum payload (20000 kg) is evaluated. Now it is found that the maximum $C_L$ during landing is just feasible ($C_L$=0.66).
- In the following 3 rows the mission evaluations for the high cruise speed (Mach 0.82), for the short design range (1200 NM (2222 km), 15900 kg design payload), and for the "OEI at TO with MTOM" mission are evaluated, and no violation of any criteria occurs.
- The last row for Rad2 presents the results for the typical mission, where no violation of any criteria occurs and total mission fuel burn of 3769 kg is found.

The key results and conclusions from the evaluations of the SMR-RAD configuration are the following:

1. For the SMR-Rad1 configuration, with maximum design range of 2750 NM@41 kft (5093 km@12.6 km), the wing area must be increased from 297 m$^2$ to 320 m$^2$. The critical case is the mission evaluation for the TLARs of maximum payload (20000 kg).
2. For the SMR-Rad1 configuration the total mission fuel burn for the typical mission is **3798 kg**.
3. For the SMR-Rad2 configuration, with maximum design range of 1200 NM@41 kft (2222 km@12.6 km), the wing area must be increased from 297 m$^2$ to 317 m$^2$. This is slightly lower than for Rad1 because of the slightly lower mOE for the initial design range mission. The critical case is the mission evaluation for the TLARs of maximum payload (20000 kg).
4. For the SMR-Rad2 configuration the total mission fuel burn for the typical mission is **3769 kg**.

## 3.7 Evaluation of the design variations and optimization of SMR-RAD

**Design variation of on-board system: More-electric aircraft (MEA) update of SMR-RAD**

The initial SMR-RAD configuration includes the same bleed air off-takes and pneumatic- and hydraulic systems that were also assumed in all previous configurations (REFX, REF, BAS, 0HEP). In the SMR-RAD the bleed air is extracted from the HPC of the turboshaft engine. However, this turboshaft engine is not optimized for bleed extraction and therefore the energetic efficiency is relatively low. It is more efficient to extract all energy from the turboshaft's power turbine shaft, which is connected to the electric generator. Therefore a so-called MEA update is applied to SMR-RAD, which avoids the bleed air extraction from the HPC. Consequently the pneumatic systems that are the consumers of the bleed air, in particular the ice protection system (IPS) and environmental control system (ECS), must also be

replaced by non-pneumatic systems. The pneumatic IPS can be replaced by electric heating systems and the bleed air supply to the pneumatic ECS can be replaced by electric air compressors. Besides the pneumatic systems, also the hydraulic systems, like flight control and landing gear actuators, can be replaced by electric alternatives. The changes in system masses and the power requirements for such a MEA architecture for an A320neo category aircraft has been previously investigated [18], and the following values are adopted in this study:

- The resulting total change in system mass on aircraft level is estimated at -980 kg, i.e. a reduction of 980 kg.
- The total electric power requirement for all the non-propulsive systems is estimated at 350 kW during the whole flight.

As indicated in the results of SMR-RAD in Table 11, this non-propulsive power off-take of 350 kW has been included in the power demand from the main electric generators. The results for the MEA update of the initial SMR-RAD are listed in the summary Table 11 in the results paragraph below.

**Design variations of the ducted electric fans configuration**

The SMR-RAD design aims at optimizing the turbo-electric propulsion system, where the design variables considered in this study for SMR-RAD are the number, location and size of the ducted fans. Because of the limited installation space on the rear centre body of the BWB, the sizing of the fan area shall be such that the ducted fans fit, i.e. that their size does not exceed approximately 16 m in span direction. Variations of the number of ducted fans are considered: besides the initial 8 fans configuration, also 10 slightly smaller fans and 6 slightly larger fans are considered. The 3 different configurations with the 8, 10 and 6 ducted fans are illustrated in the Figure 9.
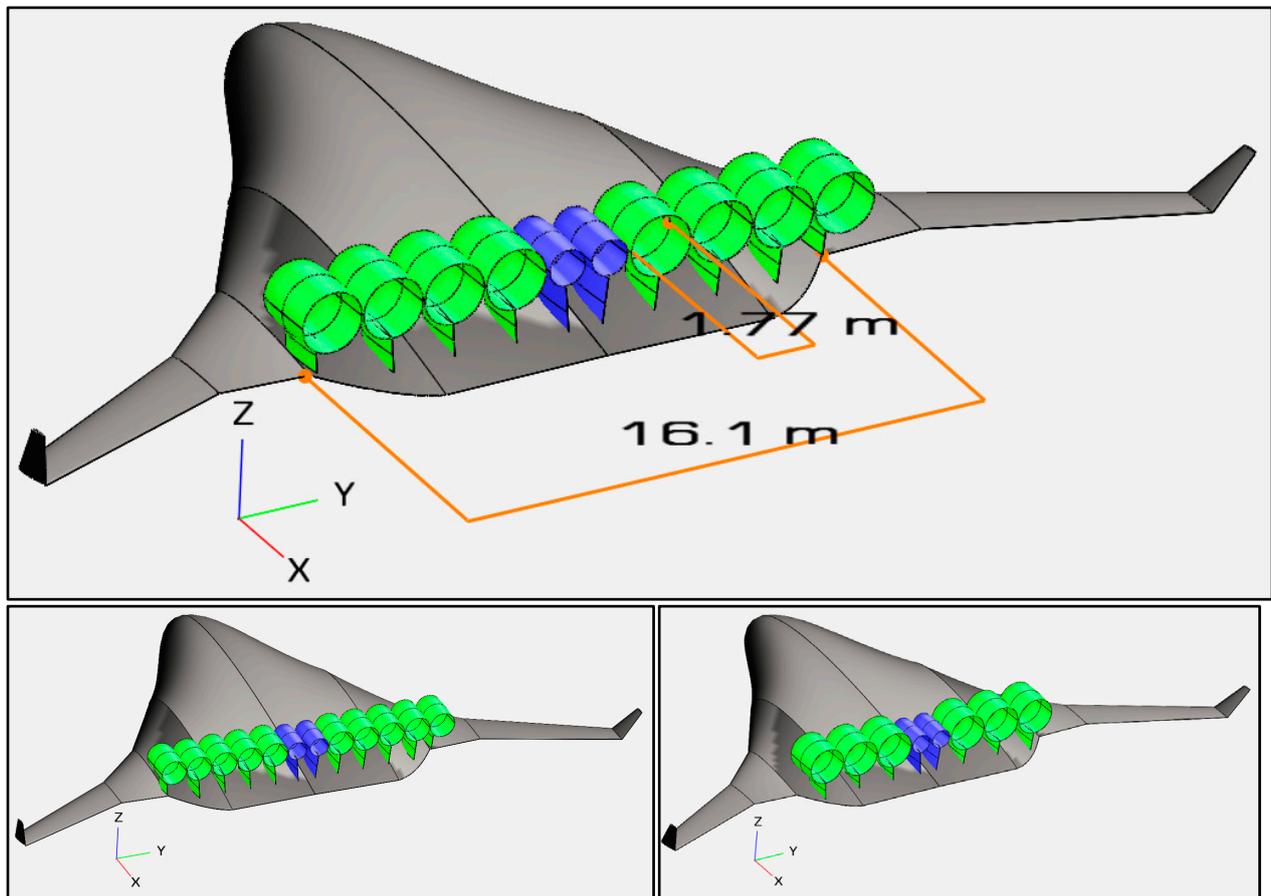


Figure 9: Illustration of the SMR-RAD with the various propulsion configurations: the initial configuration with 8 ducted fans each with 1.77 m fan diameter (upper picture), the variant with 10 ducted fans each with 1.41 m fan diameter (lower left picture), the variant with 6 ducted fans each with 2.06 m fan diameter (lower right picture). It must be noted that all the ducted fans and the two turbo-generators are all installed on the rear center body and the turbo-generators have a fixed diameter of 1.0 m. The ducted fans are indicated by the green nacelle+pylon and the turbo-generators are indicated by the blue nacelle+pylon.

The results for the 3 different SMR-RAD configurations with the 8, 10 and 6 ducted fans are listed in the summary Table 11 in the results paragraph below.

**Design variation of turbo-electric power train technology assumptions for SMR-RAD**

The technology assumptions for the turbo-electric power train components for EIS in 2035 that are used so far for SMR-RAD are the pessimistic values listed in Table 9. Instead, more optimistic values may be considered, as also listed in Table 9. The effects of these optimistic assumptions are evaluated for the best performing SMR-RAD ducted fan configuration as found in the previous section. The results for SMR-RAD configuration with the optimistic technology assumptions are listed in the summary Table 11 in the results paragraph below.

**Design variation studies: Summary of results**

The results from the design variation studies of SMR-RAD are summarized in Table 11. Here only the typical mission results for the Rad1 configurations are given. This is because the Rad2 configurations appear to have hardly any benefit in the typical mission fuel consumption. In addition, the Rad2 configurations have the disadvantage that they are only capable of fulfilling the short design mission (1200 NM (2222 km)), whereas the Rad1 configurations is capable of fulfilling the long design mission (2750 NM (5093 km)). Consequently, the lower design mission range requirement from 2750 NM (5093 km) to 1200 NM (2222 km) leads to hardly any reduction of the typical mission fuel consumption. Moreover, for each of the design variations of the Rad1 configurations, all TLARs are always fulfilled. These TLARs evaluations are not included in the Table 11 to limit the size of the table. Moreover the compliance with all TLARs was already shown for the initial SMR-RAD (Table 10) and the design variation studies have decreased power requirements due to the higher energetic efficiency and the lower take-off mass (mTO).

Table 11: Results overview for all the SMR-RAD design variation evaluations. Only the Rad1 results for the typical mission are shown.

| RAD Variants and missions | Wing area [m2] | Range [km] | Cruise altitude [km] | Cruise TAS [m/s] | Reserve fuel [kg] | payload [kg] | mOE [kg] | Aver-age total PTO [kW] | Aver-Age total bleed (kg/s) | fuel burn [kg] | mTO [kg] | Take-Off max (CL) | Landing max CL | max TT4 [K] | max Pshaft [MW] | TTC [min] 1.5kft - 35kft |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| **Rad1** | | | | | | | | | | | | | | | | |
| Rad1 initial (8 Dfans) | 320 | 1482 | 12.6 | 230 | 3000 | 15900 | 42351 | 104 | 1.96 | 3798 | 65049 | 0.59 | 0.62 | 1701 | 18.1 | 35.0 |
| Rad1 MEA (8 Dfans) | 312 | 1482 | 12.6 | 230 | 3000 | 15900 | 40640 | 350 | 0 | 3448 | 62988 | 0.58 | 0.62 | 1695 | 17.8 | 35.0 |
| Rad1 MEA (10 Dfans) | 309 | 1482 | 12.6 | 230 | 3000 | 15900 | 40327 | 350 | 0 | 3463 | 62689 | 0.59 | 0.62 | 1700 | 17.7 | 35.0 |
| Rad1 MEA (6 Dfans) | 309 | 1482 | 12.6 | 230 | 3000 | 15900 | 40066 | 350 | 0 | 3420 | 62386 | 0.58 | 0.62 | 1690 | 17.6 | 35.0 |
| Rad1 MEA (6 Dfans) Optimistic assumptions | 297 | 1482 | 12.6 | 230 | 3000 | 15900 | 38027 | 350 | 0 | 3263 | 60190 | 0.59 | 0.62 | 1662 | 16.7 | 35.0 |

It is found that the introduction of MEA systems yields a substantial saving in typical mission fuel consumption. The variations of the numbers of ducted fans yields only small changes in fuel consumption. The configuration with 6 ducted fans has the lowest fuel consumption. Therefore this configuration is further evaluated for the optimistic technology assumptions for the electric components, which yields a further substantial reduction in typical mission fuel consumption.

### 4. Assessments of potential benefits of SMR-RAD

The results from the initial SMR-RAD and all the design variation studies of SMR-RAD are summarized and comparison is made to all the other configurations (REF, BAS, 0HEP) in Table 12. Here only the key summary results of the configurations for the long design mission (2750 NM (5093 km)) are given. This is because the configurations for the short design mission (1200 NM (2222 km)), appear to have hardly any benefit in typical mission fuel burn.

Table 12: Results overview for all the SMR-RAD design variation evaluations and comparison to all the other configurations (REF, BAS, 0HEP). Only the results for the long design mission (2750 NM (5093 km)) configurations are shown. All values are given on aircraft level, i.e. based on summation of all engines and propulsors.

| | REF | BAS | 0HEP | RAD initial 8 Dfans | RAD MEA 8 Dfans | RAD MEA 10 Dfans | RAD MEA 6 Dfans | RAD MEA, 6 Dfans Optimistic assumptions |
|---|---|---|---|---|---|---|---|---|
| *Design 2750 NM* | | | | | | | | |
| mOE [t] | 44.3 | 40.7 | 37.9 | 42.4 | 40.6 | 40.3 | 40.1 | 38.0 |
| mTO [t] | 78.5 | 71.1 | 67.2 | 71.7 | 69.0 | 68.7 | 68.3 | 65.8 |
| max Pshaft [MW] | 33.6 | 29.6 | 24.9 | 19.6 | 19.2 | 19.1 | 19.0 | 17.9 |
| *Typical mission 800 NM* | | | | | | | | |
| Fn [kN] (mid flight) | 39.7 | 32.1 | 28.0 | 30.2 | 29.3 | 29.1 | 29.0 | 28.0 |
| TSFC [g/kNs] (mid flight) | 15.46 | 14.39 | 14.18 | 13.06 | 12.23 | 12.40 | 12.24 | 12.04 |
| PSFC [kg/kWh] (mid flight) | 0.197 | 0.175 | 0.181 | 0.171 | 0.155 | 0.155 | 0.156 | 0.156 |
| Fuel burn [kg] | 4855 | 3773 | 3737 | 3798 | 3448 | 3463 | 3420 | 3263 |
| Fuel burn relative to REF fuel burn [%] | 100% | 78% | 77% | 78% | 71% | 71% | 70% | 67% |

## 5. Conclusions and further work

As part of the IMOTHEP project, the conceptual design investigations for the SMR-RAD configuration are done using the NLR tools for aircraft design and mission evaluation. The current findings show some potential for the considered project targets (-40% fuel consumption for SMR in comparison with 2014 State of Art). Reductions in Typical range mission fuel burn of up to 33% are predicted for the SMR-RAD configuration in comparison to the REF configuration. The project target of -40% fuel consumption is therefore assumed to be feasible. However, a major part of the reduced fuel burn results from the expected technology developments up to 2035, as can be concluded from the BAS configuration, for which a fuel burn reduction of 22% is found in comparison to the REF configuration. Only a small portion of 1% fuel reduction is achieved by the radical configuration of the BWB airframe. Another 10% fuel reduction is achieved actually due to the application of HEP and MEA technologies, of which about 7% is due to the MEA architecture..

The conceptual design investigations for the SMR-RAD configuration have shown that the BWB with DEP in combination with the TE power train architecture is a feasible approach for the reduction of fuel consumption. For the 800 NM Typical range mission, fuel burn reductions up to 30% with respect to SMR-REF are found. In case of optimistic technology assumptions (such as the optimistic values in Table 9) this reduction can be even extended to 33%. More advanced technology assumptions, such as the exploitation of drag reduction by BLI, may yield even further reductions beyond 33%.

Furthermore, it was found that reducing the design range from 2750 NM to 1200 NM has only very small impact on the Typical range fuel burn of all the SMR configurations; for SMR-RAD for example only about 1% reduction in typical mission fuel burn can be achieved by the reduced design range requirement.

The resulting SMR-RAD configuration still has limitations and uncertainties that shall be further investigated. The ducted fan analyses are based on simplified modelling. For instance, the additional drag for the ducted fan was estimated by simplified modelling. Also the potential BLI benefits are out of scope for this paper. It is expected that improved aerodynamic and ducted fan models provided by higher fidelity analyses will decrease these uncertainties. Further aerodynamic analysis of the BWB airframe shall be improved to optimize the configuration. Simplified models were used for the power train sizing, based on the specific power and efficiency values. It is expected that more detailed electric component models (e.g. with shaft speed, power, voltage or temperature dependencies) will provide more insight into the feasibility of the current power train sizing. The current turboshaft model was dedicated for the considered mission requirements in terms of thrust and power. Possibly updated mission requirements may imply a further adapted turboshaft model, which would make the SMR-RAD analysis more accurate. Such an improved model is important as it directly impacts the Typical range fuel burn.

The application of batteries for energy storage has not been analyzed during this study. It is not expected currently that batteries will bring benefit to SMR-RAD in combination with TE, because batteries have much lower specific power than generators. However, alternative HEP architectures for SMR-RAD could still be further investigated during the ongoing studies in the IMOTHEP project.

In the ongoing and following more detailed design studies in the IMOTHEP project the SMR-RAD design will be further refined, taking advantage of technological HEP design studies and including increasing levels of fidelity. The ultimate goal in IMOTHEP is – together with the REG and CON aircraft configurations under study - to identify for HEP the key enablers and technology gaps that future research will have to bridge.

**Acknowledgements**

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