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Electrical Power System Design Aspects in the Development of the European Service Module

Arturo Fernandez*, Luca Fossati[a], Benoit Fleury[b], Ramon Lebron[c]

[a] *European Space Agency, Keplerlaan 1, Noordwijk 2201 AZ, The Netherlands*
[b] *Ariane Group, Les Mureaux*
[c] *NASA*

* Corresponding Author

Abstract

The European Service Module (ESM) is the part of the Orion spacecraft in charge of providing power, propulsion, thermal control, oxygen and water to the overall vehicle. Since it is a manned spacecraft, there were many interesting design decisions and issues along the development. This paper tries to summarise and highlight the most interesting ones, including the operations phase of Artemis I.

Keywords: Artemis I, ESM, electrical issues, design decisions, manned spacecraft

1. Introduction

Orion is a spacecraft built between ESA and NASA to bring back astronauts to the Moon and beyond. The Artemis programme aims at having again the means to explore the Moon and perform a variety of missions, including landing on the Moon, docking in the Gateway or even in the ISS. As a manned spacecraft, it should be as reliable as possible, including a very large failure tolerance to safeguard the astronauts on board.

In classical terms, manned spacecrafts were supposed to be double failure tolerant. This is a concept that is very difficult to design, test and verify. Thus, the ESM was not built with such formal requirement but instead, was built with a very large system level failure tolerance. The picture below shows the basic concept of the power system.

As can be seen, it has 4 independent 120 V buses and two independent 28 V buses. The physical allocation is done with two separate Power Conditioning and Distribution Units (PCDU). From a requirement perspective, the system can afford losing one full power chain (25% of the power) and still perform the mission nominally.

The system has four Solar Array Wings (SAW) with a total capacity of around 11 kW. Two PCDUs to deliver power to the Command Module Adapter (CMA) and ultimately to the Command Module (CM). The PCDU also delivers power to the ESM, both at 120 V and 28 V through current protected interfaces called Latching Current Limiters (LCL).

The connection of the ESM to CMA is made through 4 parallel interfaces, each delivering one quarter of the total power.

One very special feature of the ESM power system is the Solar Array Drive Mechanism (SADM). It has two rotating axis, one in the usual way along the longitudinal axis and another in the vertical axis. This is due to the fact that the mechanical loads on the SAW during the propulsion burns were too high if the SAW was deployed in the usual perpendicular position with respect to the spacecraft body. Thus, the additional axis can move the SAW up or down to minimise the loads.

The Artemis I mission launched in November 2022 went thorough 26 days of operations with a very successful outcome. It was also a very good test to put all units through the real operations environment and learn lessons that could be applied for the following missions.

2. Independent Power Buses for Manned Missions

The classic requirement for manned missions was always to be double failure tolerant. This is really difficult to meet in reality due to the large number of combinations of failures that are possible. Even more if we consider that the Failure Mode Effects Analysis (FMEA) that is done at unit level is actually a simplification of the real possible failures that can occur. Apart from the design process, the validation of such requirement is also extremely complicated, time consuming and costly. Thus, a very pragmatic approach has been taken in Orion: to implement a large redundancy policy across the various subsystems.

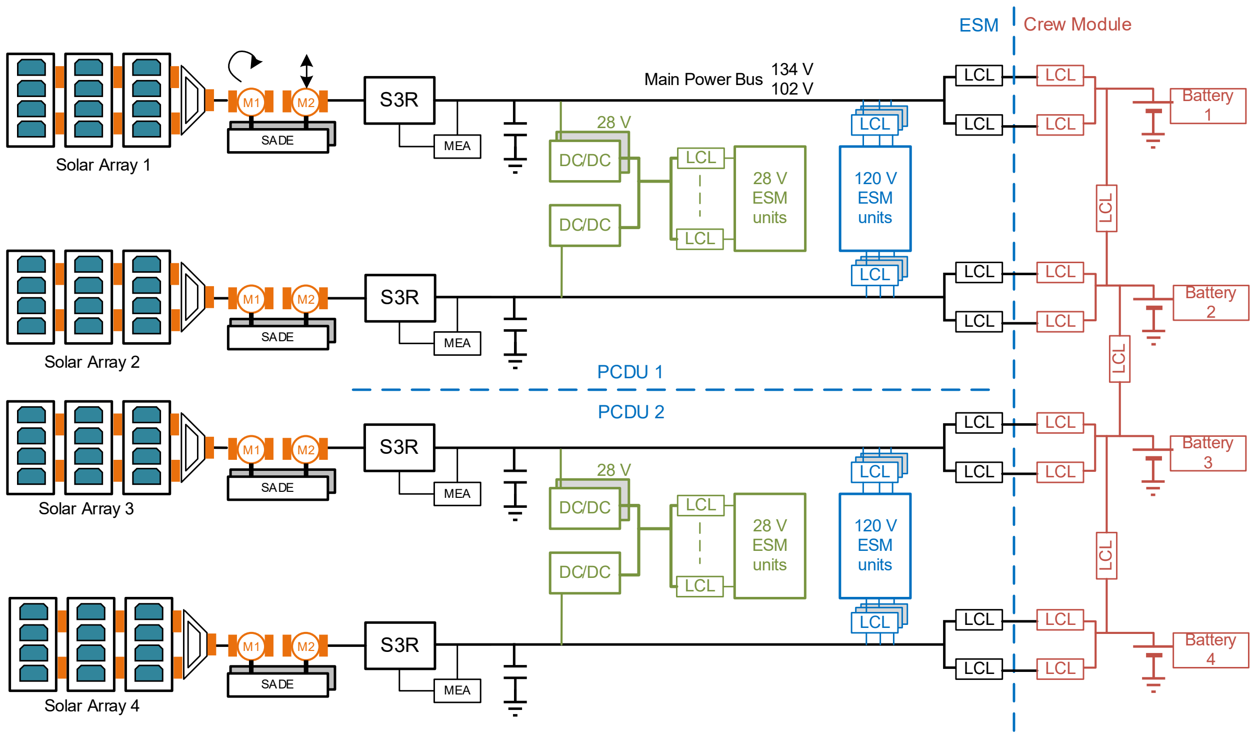

Figure 1: MPCV power system

The classic approach in the power subsystem for a normal spacecraft is to tolerate the failure of one section of the solar array. In general, that means losing less than 10% of the power, which is considered acceptable in terms of over design. In Orion, we can tolerate losing up to 25% of the power and still perform all operations nominally. Even though there could be specific failure modes leading to that situation, for instance jamming the deployment of one solar array wing, the majority of the usual failures considered in a normal spacecraft would have minimal impact in Orion.

Moreover, we would need to lose more than 50% of the power to have a Loss of Crew (LOC) scenario. That would entail having quite a dramatic combination of failure modes together to end up in that situation.

The design of the Orion power system is done with four completely independent power buses, all of them single point failure free (SPFF) as can be seen in Fig.1. The power conversion chain, the start-up system and the control system are completely independent from each other.

The main power conversion is performed with the classic European system based on the Serial Switching Shunt Regulator (S3R). This an extremely reliable system that is segregated in sections. Most of the failures in one section would lead to the loss of less than 3% of the power. The control system is Triple Majority Voted (TMV), which is also SPFF.

3. Double Bus Concepts in Long Development Time Missions

Orion implements a dual power bus system by deriving a 28 V bus from the main power bus as can be seen in Fig. 1. This concept allows easily re-using hardware developed for many other spacecrafts, thus giving flexibility in the development of the mission. It must be said that it also adds complexity to the system, so it must be considered carefully.

One of the not so obvious things to consider is the power capability of the dual bus. The decision is taken very early in the life of the project with a power budget that in most cases is not very mature. At that point, the development of the DC/DC converters of the secondary bus is launched and the maximum power capability will be fixed.

However, long and complex projects like Orion can have many turning points along its life. Multiple issues can arise, and the power of the 28 V bus could go up and down. Units that were thought to be re-sued in a given power bus could end up in the other one. An issue during the development could also affect the final location of the unit in the power system. Moreover, the power consumption of the units is not usually well known at the beginning of the project. Thus, the final power budget could potentially go in both directions.

The maximum power available is usually a hard limitation with severe impact in cost and schedule. However, if too many loads are moved to the main power bus, the 28 V bus could end up heavily off loaded. That means that the DC/DC converters could be operating in a very deep Discontinuous Conduction Mode (DCM). Depending on the converter topology and the control system, several problems could appear:

- the voltage regulation could be worse than in nominal conditions
- the converter could operate in hiccup mode and the EMC could be compromised

It should also be noted that, in general, the focus of the test campaign is to ensure that the converters operate properly under the worst case conditions. Thus, more attention is paid to the maximum power and maximum temperature operating conditions since they are the sizing case. However, if the real operating conditions were very off loaded and cold, the test campaign could leave blind sides in a key power system item.

Thus, these issues should be considered from the beginning to avoid problems later in the development process. Moreover, a robust conversion system will give more flexibility to the placement of the units as needed by the project.

The worst case analysis (WCA) also led to some interesting conclusions in Artemis I. As is well known, space missions are developed considering margins in all subsystems since it is always possible to find unknown issues. The problem is that it is very complicated to keep all the margins under control and avoid the interaction between them. This can lead to overdesigning some systems. In Artemis I, the solar array produced 22% more power than predicted. In terms of solar array design, this means that a smaller solar array would have been possible. Furthermore, the overall spacecraft power consumption was also lower than anticipated. Thus, there was a relatively large power margin in this mission. Considering this is a manned spacecraft, from a mission perspective this was considered as a positive aspect because it gives more room to accommodate failures. As mentioned before, the reliability concept is based on a large redundancy level. In this case, it means that the power system would still have nominal behaviour after a larger number of failures.

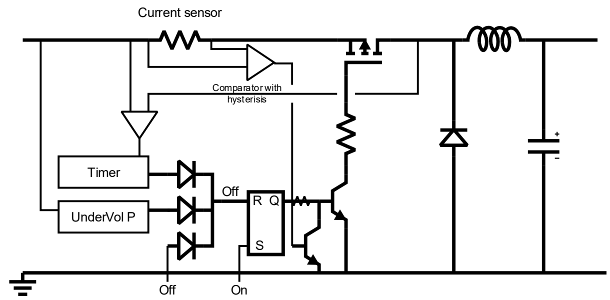

Figure 2: Basic scheme of Latching Current Limiter

4. High Current LCL

An interesting aspect of high power spacecrafts as Orion is the current protection devices needed to isolate the main power bus from failures in the various units. As is well known, the most usual way to protect the main power bus is to use current limiters to feed all units connected to the main bus. In general, those current limiters operate in linear mode while they are in limitation mode. Figure 2 shows a basic scheme of an LCL. The main drawback of that concept is that there is a very high dissipation during that period of time, typically a few milliseconds (ms). If the outlet must deliver very high current, the thermal design of the outlets has to be designed very carefully. Moreover, in complex spacecrafts like Orion it is very likely that there are multiple devices in series in each power line. This is because it might be needed to isolate different parts of the system.

Thus, the compatibility between the various current limiting devices becomes complicated and it has to be ensured that compatibility tests are performed.

One option to deal with a very high current situation is to implement a variable trip off time, so that the limitation time is shorter when the current is higher. This is simple to implement but also has some impact on the verification campaign since several conditions must be tested.

If the current is very high, it opens the question about the best technical choice for this application. If the dissipation during the limiting sequence is too high, the thermal design could be a limiting factor. Thus, other less dissipative options might be needed for future designs.

At the moment, ESA is investigating the use of a switched LCL based on GaN FETS. The operation is not linear and hence, the dissipation level in limitation mode is much lower than in the classic design.

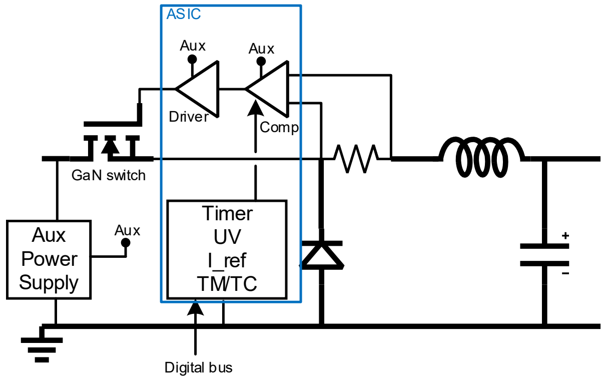

Figure 3: Concept of Switched LCL

However, there are also important technical challenges that must be carefully considered. Note that for instance, the switched LCL needs a filter that has an impact on the main power bus impedance and in the power quality.

In Artemis I there was a flight anomaly in the high current LCLs. There were several glitches during the mission that interrupted the power flow in some lines. Given the high redundancy level of the spacecraft, this was not a real problem. However, it was an unwanted behaviour that led to a thorough investigation of the root cause. The conclusion was that a single event effect (SEE) in an optocoupler forced an unwanted OFF command in the LCL. The troubleshooting was simple since it only required an ON command to turn it back to nominal operation. At hardware level, it was found that insufficient filtering in the optocoupler line allowed the SEE to generate a false signal. The issue has been solved by improving the filter on that line.

5. Battery Bus With a Battery Far Away From The Bus Capacitance

The usual power system in most of the spacecrafts has the battery connected very close to the power conversion system. Moreover, the power distribution happens downstream the battery connection. Thus, the main power bus is electrically very clean from a power quality perspective. However, if the battery is physically located far away from the conversion point and multiple units are connected to the power line, between the battery and the power converter, the line impedance will be much higher than in the usual arrangement. As a result, the current ripple can be much higher than usual. It should be noted that the current ripple will affect all electrical interfaces connected to the main power bus and must be considered in the power quality requirements.

The grounding scheme is also another topic to consider. It could be located in the regulation point of the main power conversion circuit or closer to the users, far away from the regulation point. Common mode emissions must be carefully analysed in this case.

6. Slow Start Up After Eclipse

One interesting aspect of human spacecrafts is the fact that the transitions between eclipse and daylight could happen in very diverse attitudes. Thus, the illumination of the solar array could have very shallow angles and very slow transitions. This is always an issue to be carefully considered in power systems. As soon as some light hits the solar array, the voltage comes up very quickly while the current is still very low. Thus, the power system will try to deliver power based on the voltage measured on the SA interface. However, there is very little power available and in general, the main power bus would not be sustainable. In such case, the system could enter into a sort of hiccup mode with multiple re-start events until the power availability is enough to maintain the main power bus.

Two very important things have to be considered:

- Ensure that the timing of the start-up sequence is adequate and completely deterministic
- Ensure that the FDIR counters don't block the system assuming that there is a failure

Specifying the worst-case illumination scenario when coming out of eclipse is a key requirement, as well as the relevant test campaign to verify the correct behaviour in this scenario.

If the EMC requirements are limiting, the impact at design level would be to have a larger main bus capacitance or even to have a local battery close to the solar array regulator output. This aspect must be considered as early as possible because the impact in the system is significant.

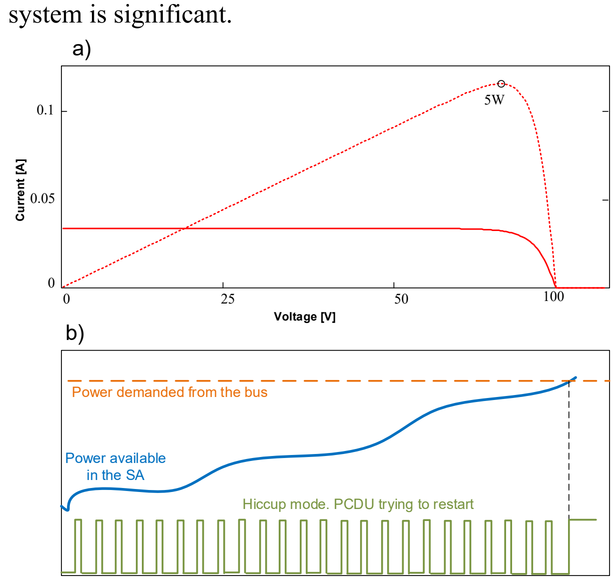

Figure 4:a) IV curve with low illumination; b) hiccup mode when power on SA is below the demand

7. Telemetry Rates and Accuracy for Failure Investigation

Telemetry (TM) is basically the only way that we have to understand the healthiness of the spacecraft during flight. In nominal conditions its importance might be overlooked but, during flight, telemetry reveals itself as one of the key features of the system. Any failure investigation or troubleshooting procedure relies mostly on the telemetry available. The obvious point is to have the right parameters giving visibility to the status of the spacecraft. However, there are some more interesting issues that might be underestimated.

As has been explained before, a manned spacecraft might be subject to a higher variability in configuration compared to a conventional satellite. Thus, the operation points of the units could be well apart, from very low power consumption in standby to high current demand in other operation scenarios. It is then required to have a good accuracy along the whole value range, from low to high. Otherwise, the values read could be highly misleading. If the telemetry is optimised for the high value range, at low power consumption the values could be very inaccurate. Thus, if troubleshooting is needed in that scenario, it could be complicated to understand the real situation.

Another key aspect is the telemetry rate. Today we still fly with low TM rates, in the other of a few Hertzs or tens of Hertzs. In nominal conditions this is good enough but again, in troubleshooting scenarios this is very low. Electrical signals have much shorter transient behaviours and low-rate TM is not able to catch them. In manned spacecrafts this is particularly important since the impact could be very serious. Today we have the technology to push the TM rate much higher and, even though it might not look like an attractive feature, it is paramount in flight. There are examples of this concept already flying, for example in the deployment of solar arrays. The FPGA in the PCDU can take samples of specific signals at very high rate (kHz), store them temporarily and then download them on request. In essence, this is a sort of digital oscilloscope concept that is technically feasible at a larger scale. However, TM is usually seen as a secondary feature and not high in the priority list or the budget allocation. The lesson learned is that in complicated missions and especially in manned missions, it is worth allocating more resources to the TM performance.

In terms of TM behaviour, it is also very important to ensure that the systems show univocal signals to avoid confusion during flight. In Artemis I, a couple of systems showed glitching behaviour. That triggered investigations to assess the cause of such glitches. A classic example is the signal coming from the solar array deployment microswitches. These are very sensitive devices, and the closing contact can give some false signal sometimes. The problem is that it is key to ensure that the solar array is correctly deployed to avoid any problem during manoeuvres and the microswitches are the ones in charge of assessing the completeness of the deployment. Fortunately, there are other means on board to verify that and hence, it was possible to conclude in short time. In any case, this should not prevent from improving those systems and make them more robust and less sensitive.

Another classic example is the presence of contamination particles in the Solar Array Drive Mechanism (SADM) slip rings. In Artemis I, some signals coming from the SADM had glitches at specific rotation angles. Every time the sweep has passing through a given position, there was noise in some signals. In this case, it was less critical than the deployment case and the fact that the glitch occurred always at the same point led to the conclusion that it was a contamination issue. This is a critical aspect in slip rings but in general, it can be solved by a drive in process commanding the solar array to rotate several times.

8. Conclusions

The development of power systems for human spacecrafts is a complicated task that faces multiple hurdles during its lifespan. The long duration of the development can impact significantly the power budget and the variable nature of the configurations in flight makes this issue even more complex. Moreover, the high redundancy needed to achieve the required reliability has an important impact since oversizing a system is automatically multiplied several times. Thus, simple systems that are easily scalable and are flexible are the best option for this type of missions.

Telemetry and commandability are paramount aspects during flight operations and, even though they might not look as very fancy features, improvements in the telemetry availability and telemetry rates are key technologies to be implemented in future manned missions.

Finally, high system level redundancy is seen as a more feasible concept than the classic double failure tolerance concept. The verification effort associated with the latter is very significant and more important, drives the reconfiguration options in flight in case of troubleshooting. Thus, power systems with multiple independent buses that can be interconnected are a very interesting option for very reliable missions.

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