FEASIBILITY STUDY OF A POCKETQUBE PLATFORM TO HOST AN IONOSPHERIC IMPEDANCE PROBE

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ABSTRACT

Since the advent of CubeSat spacecraft, universities and private entities have been successfully designing and launching satellites at a fraction of the traditional cost. These satellites still accommodate useful scientific payloads. Another recently established satellite format is the PocketQube (PQ) - one eighth the size of a CubeSat – with the aim of further reducing launching costs. However, this brings with it the challenge of working with substantially smaller power, mass and volume budgets. Accurate ionospheric modelling requires the use of electron density measurements at the topside of the ionosphere which could be obtained via distributed in-situ sensing. This makes a low cost PQ constellation ideal for this application. In order to assess the feasibility of the PQ format, a preliminary study was conducted about the design of a PQ technology demonstrator capable of carrying a scientific payload. In this paper, the design approaches are discussed, keeping in mind the design budget restrictions as well as the constraints imposed by the ionospheric sensor.

1 INTRODUCTION

Throughout the last decade, the CubeSat has become the dominant standard for universities and private entities developing small satellites, with over 100 launched in 2015 alone [1]. The reason for the popularity of this particular nanosatellite format is attributable to: (1) a well-established and well-promoted standard, (2) readily accessible knowledge-base, (3) substantial flight legacy, (4) organised launch services with a deterministic cost structure, (5) a thriving ecosystem of off-the-shelf hardware, and finally (6) the availability of complete kits on the market which simplify entry for non-traditional space users.

The question that naturally arises is whether one can, or even should, go smaller. The main driving force for miniaturization (insofar as still being able to accommodate useful payloads) is the expected reduction of launch and development costs. This has effectively democratised space technology by allowing academic institutions and small businesses to participate. This has greatly increased the rate of development.

Secondary arguments for further miniaturisation are related to slowing the accumulation of space debris, by putting less matter into orbit per mission. However, this is somewhat counteracted by the proliferation of miniature satellites which is inherently brought about by the reduced cost barrier for entry. Finally, with such reductions in cost, it is quickly becoming feasible to launch larger constellations of satellites in missions where this would be necessary [2].

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2 DEFINING FEASIBILITY

The practicality of a miniature satellite project hinges on three aspects: (1) technical, (2) operational and (3) financial feasibility. We shall define technical feasibility on the grounds of the existence and availability of all the technology required to develop, test and launch a given miniature satellite. Operational feasibility shall be defined in terms of achieving mission objectives within the given operational constraints and time scales. Financial feasibility again means different things for enterprises with different budgets. However, we shall establish the threshold of financial feasibility when development costs equal launch costs. If development costs are significantly lower than the launch cost, then it may be surmised that insufficient effort is being expended on reduction of overall mission risk. The converse situation defeats the original objective of cost reduction. Therefore aiming for parity between development and launch cost seems logical.

With this in mind, a number of other standards have been developed for smaller and lighter spacecraft. These include the TubeSat [3], PocketQube [4] and most recently the SunCube [5]. Additionally, studies regarding 'satellite on chip' [6] have also been carried out. Whilst miniaturization inherently reduces the launch cost, this does not necessarily mean that the design costs will follow the same trend. The design costs are typically kept low by using non-space grade commercial off the shelf (COTS) components wherever possible. In the space environment, many of these COTS components (batteries [7], microcontrollers [8], solar cells [9] etc...) will be operated outside their design envelope and very often these have to be re-qualified by the user for the specific application. This frequently involves extensive testing and time consuming characterisation in order to augment the datasheet to the required level. Reliability engineering design techniques are required to achieve the desired reliability with components that were designed for terrestrial use. Fortunately, such information is frequently shared within the academic community, and over time the knowledge base of devices with a successful flight heritage grows to minimise the cost and effort incurred in future projects.

Table 1 – Economic comparison of different satellite classes, launched to ~550km LEO [10]

CLASS	MASS RANGE	EXAMPLES *	TYP. MASS	PREVAILING TECHNOLOGY*	LAUNCH COST	LAUNCH $/KG	DEV. COST	OVERALL COST
Satellite	1 – 10 T	Telescope	10T	Space Grade	€100M	€10k	> €1B	> €1B
MiniSat	100kg – 1T	Earth Obs.	1T	Space Grade	€10M	€10k	€300M	> €300M
MicroSat	10 – 100 kg	SSTL-150	50kg	Space Grade	€1M	€10k	€20M	> €20M
NanoSat	1 – 10 kg	CubeSat	1.3kg	COTS	€80k	€60k	€100k	~ €180k
PicoSat	100 – 1000 g	PocketQub	250g	COTS	€25k	€100k	€35k	~ €60k
FemtoSat	10 – 100 g	SunCube	35g	COTS, CoB	€3k	€85k	€40k	~ €43k
AttoSat	1 – 10 g	Sprites	~10g	COTS, CoB	€300	€30-90k	€5k	~ €5k
AttoSat	1 – 10 g	SoC	~1g	IC	€30	€30-90k	€200k	~ €200k

* COTS = commercial off the shelf, CoB = chip on Board, IC = integrated circuit, SoC = satellite on a chip

Flying a satellite at the least overall cost will be somewhere in between the CubeSat and the satellite on chip. With the final choice depending on the mission requirements such as expected lifetime, and number of satellites. One such possibility is the recently established PocketQube (PQ) [4] which is one eighth the size of a CubeSat at 5×5×5 cm. At the time of writing, only four PQ's have been launched (T-LogoQube, QubeScout-S1, WREN and Eagle-1) and no scientific payloads were placed on those in the smallest 1p format. The aim of this paper is to question the feasibility of hosting a scientific instrument on board a 1p PocketQube sized spacecraft, which we shall call – UoMBSat1.

3 SCIENTIFIC MISSION

The US National Space Weather Service defines space weather as the "conditions on the Sun and in the solar wind, magnetosphere, ionosphere, and the thermosphere that can influence the performance and reliability of space-borne and ground-based technological systems and can endanger human life or health" [11]. Of particular interest are the effects caused by the ionosphere. The ionosphere is a partially ionised region of the upper atmosphere, ranging from approximately 90 to 1500 km. The ionisation is mainly caused by solar radiation [12]. Variability in the ionosphere can degrade the performance of many radio systems; including satellite navigation systems (i.e. GNSS), satellite communications and space based radar. [12]. Because of this, the modelling, forecasting and monitoring of the ionosphere is very important. Measurements of the electron density in the ionosphere can be used to improve empirical ionospheric models or as inputs into assimilative ionospheric models. Such models can be used to help mitigate against the impact of the ionosphere on radio systems.

Measurements are often made below the height of the peak density of the ionosphere using ground based radar systems (i.e. ionosondes) [13]. Integrated measurements of the total electron content (TEC) can be made using dual frequency global navigation satellite systems (GNSS) [14]. However, for direct measurements above the ionospheric peak density, satellite based instrumentation in low earth orbit (LEO) is generally required. The ionosphere is a constantly changing environment and thus a constellation of spacecraft is required to provide good spatial and temporal measurement resolution. One way to achieve this is via the use of small, dedicated spacecraft, such as CubeSats or PocketQubes.

The three most widely used in-situ ionospheric instruments are the radio frequency impedance probe [15] [16] [17], the retardation potential analyser [18] [19] and the Langmuir probe [20] [21]. The latter two instruments have power consumption and volume requirements which are significantly larger than the available volume and power budgets for a PocketQube spacecraft. Therefore, these were omitted from further consideration to focus on the first technique.

The principle of operation of the radio frequency impedance probes is to determine a resonant point of the impedance of an antenna embedded in the ionospheric plasma surrounding the spacecraft. The impedance ($Z(\omega)$) is described by [17] [22]:

$$Z(\omega) = j\omega C_o \left[\frac{\omega^2 - {\omega_{SHR}}^2}{\omega^2 - {\omega_{UHR}}^2} \right], \tag{3.1}$$

where C_o is the capacitance of the probe in free space,
ω_{SHR} is the Sheath Seath Hybrid Resonant frequency, and
ω_{UHR} is the Upper Hybrid Resonant frequency.

The impedance has a pole at the plasma's Upper Hybrid Resonant (UHR) frequency (Figure 1). The UHR is an ionospheric plasma property dependant on the plasma frequency and its gyro-frequency (equation 3.2). Therefore the UHR frequency can be used to determine the electron density of the surrounding ionospheric plasma.

$$\omega_{UHR} = \sqrt{\frac{n_e q_e^2}{m_e \varepsilon_0} + \omega_c^2}, \tag{3.2}$$

$$\omega_c = \frac{q_e B}{m_e}, \tag{3.3}$$

where: n_e is the electron density; q_e is the electronic charge; m_e is the electronic mass; ε_0 is the permittivity of free space; ω_c is the plasma gyro-frequency; B is the Earth's magnetic field

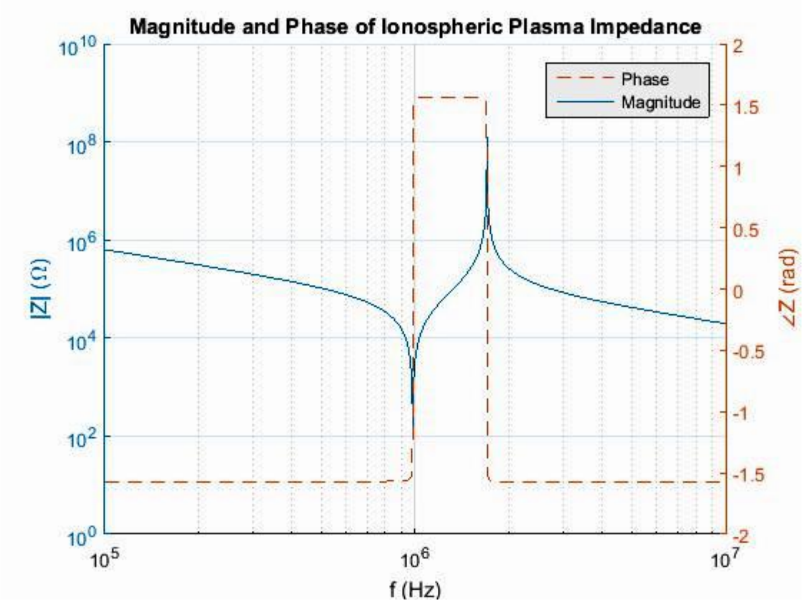

Figure 1- The magnitude (blue) and phase (orange) components of the Z(ω) - ω characteristic for ionospheric plasma, showing the two resonant points, the Sheath Hybrid Resonant (SHR) frequency and the Upper Hybrid Resonant (UHR) frequency.

4 MISSION REQUIREMENTS and CONSTRAINTS

The constraints in Table 2 have been applied for the initial system design, these constraints have been set by taking into consideration the PQ standard together with the required scientific mission requirements.

Table 2. Design Constraints

PARAMETER	CONSTRAINT	NOTE
Mass	250 g	PQ standard
Dimensions	5x5x5 cm	PQ standard
Attitude	Determination	Required to investigate the impact of attitude on the measurements
Attitude	Control	Not required
Magnetic Field	Clean spacecraft required	The electron density measurement relies on knowledge of the magnetic field. Therefore, the platform should be magnetically clean. Ideally magnetic field will be measured on board.
Position/Timing	Required	Location and time of measurements is required. Ideally GNSS on board
Altitude	300-500 km	Topside ionosphere

5 PROPOSED ARCHITECTURE

In CubeSats, individual subsystems are often designed independently, and then interconnected via a back bone such as the PC104. In the case of the PocketQube, the tighter design constraints necessitate an integrated approach, such that hardware is kept to a minimum so as to keep within the designed budgets. Thus the design approach would have to be significantly more integrated without compromising redundancy. The top-level architecture of UoMBSat1 is shown in Figure 2. The design will have to include distributed redundancy and fall back mechanisms such that the mission is not compromised. Since the payload will be designed independently of the platform, the PQ60 interconnectivity standard has been chosen to decouple payload design. Such modularity ensures reusability of both payload and platform for future missions.

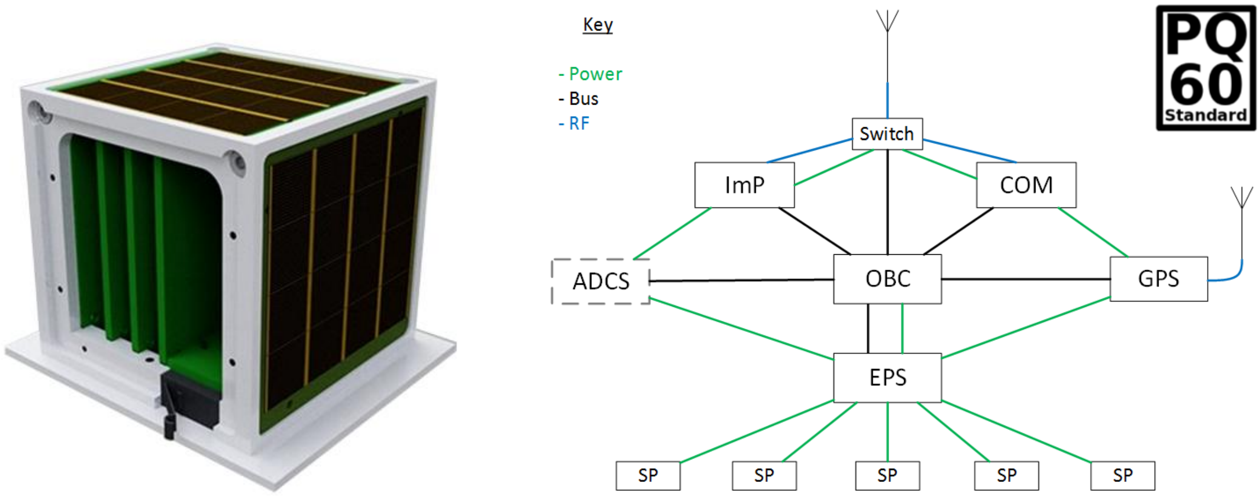

Figure 2 - Proposed UoMBSat1 Platform – Physical Architecture and System Architecture

5.1. Radio Frequency Impedance Probe – ImP

Various plasma impedance probes (ImP) have been designed and implemented; e.g. [20], [23] [24]. For this project, the ImP operates by exciting an antenna with a frequency sweep and measuring the impedance with a trans-impedance amplifier. Once the UHR frequency is determined, together with a measurement for the Earth's magnetic field, the electron density can be obtained.

ImP requires three distinct stages (Figure 3):

1. An analogue front end, which includes the antenna and analogue electronics up to the ADCs.
2. A signal generation block.
3. A digital back end implemented on a microcontroller. This includes the required ADCs and the on-board data processing to obtain the electron density measurements from the impedance curve. This element is also responsible for communicating with the spacecraft OBC.

Both the PocketQube and ImP require an antenna, for communications and ionospheric measurements respectively. However, since volume is limited, a single antenna will be time-shared between the platform and the payload. This is possible as the performance of impedance probes is insensitive to the geometry of the antenna [25] [26]. Volume and power restrictions determine that the ImP sensor electronics have to be implemented in one PQ standard printed circuit board (42 mm by 42 mm) and with an initial target of 50 mW average orbital power. The measurement frequency, data storage and telemetry requirements for ImP are still being investigated in further studies with regards to the mission planning stage.

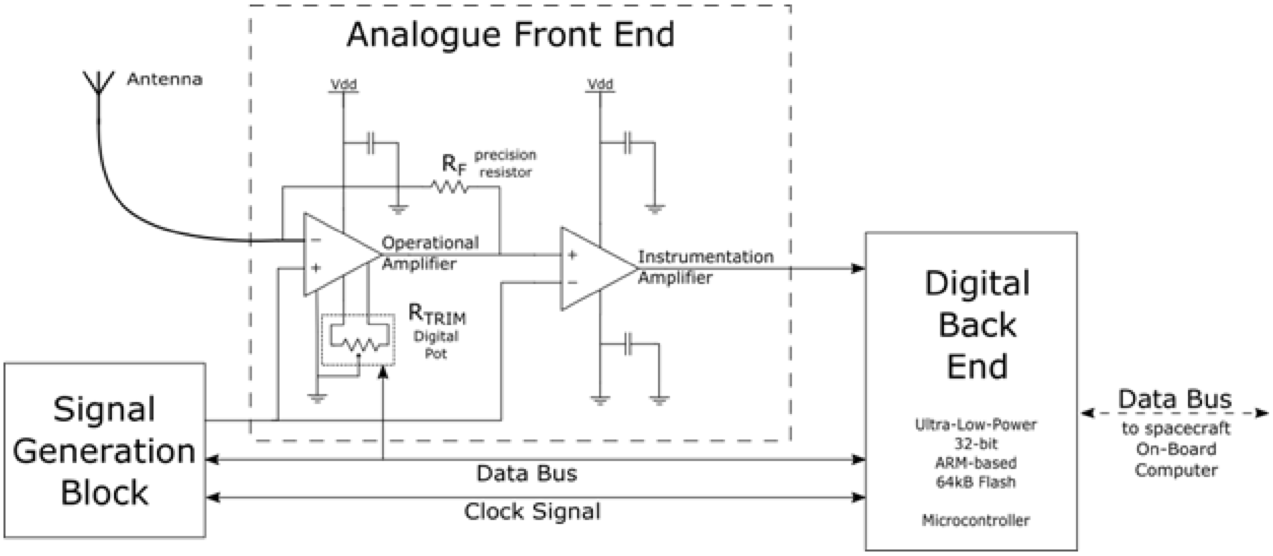

Figure 3 - Block Diagram of the proposed ImP architecture.

5.2. On-board Processing

The three main on-board processing functions are the electrical power system control, ADCS processing and the command and data handling. Ideally these three processes are implemented as part of independent systems. Power budget constraints will limit this possibility unless redundancy is compromised. Having a processor for each primary function, is therefore being considered. This provides the possibility of performing secondary functions in the case of a processor failure.

For this to be possible the processors will be operated below their nominal frequency. This would then be increased to perform the secondary functions. In this way the power consumption would increase when performing the secondary functions. This is possible since a malfunctioning processor can be isolated and its allocated power budget redistributed. In order to implement such a system digital buses will have to be used such that each processor can have access to all the subsystems. Thus a balance will have to be found between redundancy and increased complexity.

Single event upsets (SEU) have to be catered for, the use external window watchdog is one possibility to detect any anomalies in the code flow, together with a check pointing system. FRAM memory is also being considered for its radiation tolerance [27]. Error detection and correction techniques will also be used to regularly sweep the memory to prevent the accumulation of SEU corruption [28].

5.3. Communications Systems

The majority of deployed small satellites have used amateur satellite bands, with the most common frequency being the 435-438MHz (70cm) band [29]. This situation will most likely change in the near future since as the number of small satellites increases, it becomes proportionally more difficult for IARU to keep coordinating non-amateur satellites [30]. Currently, the 70cm band is still being considered for use in this project, mainly due to the vast knowledge available from previous projects and the availability of COTS transceivers with flight legacy. A half wavelength dipole antenna for a 70 cm band would be small enough to implement a simple antenna deployment mechanism and it falls within the requirement of being electrically short for the ImP. A COTS RF switch will be used to change the antenna operating mode, further research has to be carried out to analyse the effect of the switch insertion loss.

A half-duplex approach has been chosen so as to work on a single frequency, keeping the hardware to a minimum. A radio transceiver integrated circuit (IC) is the best solution for such small systems as it provides all the functionality in the least amount of space.

For a sun synchronous orbit it is assumed that from the 15 orbits per day, the satellite will pass reasonably overhead twice a day. The satellite contact time will be 6.5 minutes per pass, assuming a minimum elevation angle of 15° [31]. If half duplex communications is used it is assumed that 75% of the contact time will be in downlink, and if AX.25 is used then there will be an overhead of 11% [32]. Now assuming that 100 telemetry signals are required with floating point accuracy per orbit and these are sampled at 16 times per orbit, then the total of 94 KB are required for telemetry. Assuming that the Payload will require 50 KB as an initial estimate, then a total of 144 KB of data has to be transmitted. With these assumptions a minimum baud rate of 2000 is required, a baud rate of 2400 is therefore being assumed.

The Amsat link budget spreadsheet was used to calculate the downlink and uplink budgets [33]. For the downlink budget if it is assumed that a 0.5W output power is obtained from the transceiver then the Effective Isotropic Radiated Power (EIRP) will be -2.6dbW. The worst case slant range is 1518Km, which results in a free space path loss of -148.9dB [34]. A spacecraft antenna pointing loss of -1dB is also considered and this translates to a lenient pointing accuracy requirement of ±23.6°. If it is assumed that a circularly polarized antenna is used at the ground station then the worst case polarization mismatch will be 45°, giving a polarization loss of -3dB [34]. For the ground station a 7 element crossed Yagi antenna is assumed, having a gain of 13.5dBi. The system link margin was calculated assuming an FSK modulation scheme and the AX.25 protocol. With a maximum packet size of 332 bytes, and a 1% packet error rate (PER) then a bit error rate (BER) of around 10^{-6} is required. For the Uplink budget, the ground station transmission power was taken to be 30W.

For uncoded FSK, the uplink system link margin was calculated to be 10.2dB whilst for downlink this was just 6dB, when typically a margin of 10db is recommended [35]. For this link margin the baud rate must be reduced to 1000 Bd. Alternatively forward error correction (FEC) techniques can be used to increase the link margin by 4dB, so as to achieve the estimated value of 2400 [36].

5.4. Orbital Determination

One of the Payload requirements for Electron Density measurements is to have a measurement of orbital data, specifically the PQ’s location. In order to implement this, two possibilities are being explored, the first one being the use of NORAD Two-Line Element (TLE) sets to obtain orbital parameters, which would then be used in an orbit propagator. The accuracy achievable using this approach is in the order of 10-30 km (1σ) track error and this assuming a propagation of 24 hours. The error, of course, worsens with longer propagation times [37].

The second option being considered is to use a GNSS receiver to obtain positional information. COTS receivers designed for terrestrial applications cannot operate in LEO, specifically due to the fact that the high velocities generate large frequency shifts in the signal, which standard receivers aren’t designed to lock onto. Additionally, COTS GNSS receivers are limited in operation due to the International Traffic in Arms Regulations (ITAR) which limits the maximum velocity (1000kts) and altitude (60000ft) at which these receivers are allowed to provide a position fix. Along with the above mentioned difficulties, a GNSS receiver will have an impact on the PQ’s power budget. In fact, continuous operation of a GNSS receiver may not be possible within the power constraints. A study is therefore required on the possibility of periodically using GNSS to obtain orbital parameters, which would then be used by an adequate orbital propagator to establish an estimate of the PQs position.

5.5. Attitude Determination System - ADS

As discussed in section 4, ImP electron density measurements may be attitude dependent. However, the accuracy requirement has not yet been determined. Therefore $\pm 10^{\circ}$ has been assumed. The ESTCube-1 flight results [38] have validated an attitude determination system (ADS) by comparing measurements to the attitude calculated from the on-board camera image. A peak discrepancy of 1.43° was found. Whilst this is not a measure of absolute error, it gives an indication of the achievable accuracies. The ESTCube-1 used a combination of gyroscopic sensors, magnetometers and sun sensors combined using an Unscented Kalman Filter. This approach has been widely adopted for CubeSat ADS.

For the PQ, a similar approach will be adopted but regardless of the method used, a magnetometer is still required for the Imp compensation, and a gyroscopic sensor is needed if active detumbling is to be performed. In the PQ, some solar cells (1 per face) will also be used for sun sensing. This avoids dedicated sun sensors at the expense of some ADS accuracy, but saves on PQ surface area.

Another interesting technique for attitude determination is through the use of carrier phase measurements of GNSS signals across single or multiple antennas [39]. Although this technique has been successfully tested on large aircraft and small unmanned air vehicles [40], the challenges associated with small spacecraft at high-speed are very different and must be explored carefully.

5.6. Attitude Control System - ACS

For the purposes of communications and ImP, an attitude control system (ACS) is not strictly necessary. However, the implications of having one on-board are being investigated nonetheless, as it would improve the link budget, improve GNSS communications, allow better distribution of solar radiation on the panels, and reduce radio fading at the expense of complexity and power consumption. The pointing accuracy requirement is largely defined by the directional antennas used for communications, and for a dipole antenna this requirement is not particularly stringent. In any case, the limiting factor of the control algorithm will be the ADS accuracy, so an initial pointing accuracy of 10° was set as well.

Due to the magnetic cleanliness requirement, traditional passive control techniques such as hysteresis rods and permanent magnets have to be avoided and gravity booms and solar sails were considered too mechanically complex to fit in a 1p PQ. So two approaches are being considered:

1. 3MT: 3-axis coreless copper winding magnetorquers (MT) [41], or
2. 3RW+3MT: 3-axis reaction wheels (RW) coupled with smaller MTs for de-saturation [42].

For the PQ, the disturbance torques, in descending order of importance, are of four types:

1. Magnetic Disturbance, due to magnetisation or Lorentz forces created by currents in the PQ
2. Aerodynamic Disturbance, due to residual atmospheric pressure at LEO
3. Gravity Gradient Disturbance, reacting to the uneven distribution of mass in the PQ
4. Solar Radiation Pressure effects, acting unevenly on the faces of the PQ

The above were calculated and compared for a PQ sized object [43] scaled from CubeSats. A worst case residual magnetic moment of 4.16×10^{-4} Am^2 has been calculated, yielding a disturbance torque of 2.1×10^{-8} Nm, by assuming the largest contribution coming from a badly-designed solar panel current, looping round the panel's perimeter. An aerodynamic disturbance of 1.56×10^{-8} Nm, was estimated from the residual atmosphere at 550km acting on a total incident area of $40cm^2$ at 7586m/s, at a moment arm taken at 1/3 the size of the 70cm band antennas. Gravity gradient and solar pressure effects were deemed negligible at 9.0×10^{-10} Nm and 4.1×10^{-10} Nm respectively.

A 3MT solution is the most magnetically clean, given that these would only be energised when ImP is inactive. However, the actuation would be rank deficient since the coil whose axis is parallel to the earth's magnetic field cannot produce any torque, and any coil close to parallel would need to be substantially overdriven to generate any useful torque. This means that reliance on just magnetorquers requires them to be sized and rated to provide at least 10 times the dominant disturbance torque. This results in large and heavy coils that consume substantial power.

The alternative method of actuation is to use the 3RW+3MT setup, based on DC micro-motors and smaller MT coils. This allows the MT to be opportunistically activated only when the B-field is at ~ 90º to the coil axis, which lends itself for much greater power and mass efficiency. The RWs also lend themselves for simpler and more accurate control.

The problem here is that the motors carry a tiny annular NdFeB permanent magnet which although screened with a mild steel canister, does leak some flux. The measured magnetic flux density at the motor surfaces is quite low and comparable to the geomagnetic flux (~50uT), but it has yet to be seen to what extent this will distort the field in proximity to the ImP antenna. That said, removal of all magnetic disturbances will be impossible, because even the current in the panels and the batteries will generate some disturbance. Therefore, some method of compensation would have to be devised anyway.

Finally, a viscous damper consisting of a hermetically sealed fluid-filled chamber is additionally being considered as a passive de-tumbling / de-oscillating system of last resort. However, this needs to be modelled and would in any case only work if there is oscillatory motion, due to gravity gradient effects. For this, the PQ's centre of mass would need to be significantly off centre [44].

5.7. Electrical Power System – EPS

For a PQ a reasonable solar cell area will be 6 sides with 16 cm^2 of active area and cells being considered are 28% efficient triple junction solar cells. The worst case sun synchronous orbit in terms of power is when the Sun is in the orbital plane (midnight/noon), since the PQ will have the longest shadow time [45]. With these parameters the PQ will have an orbit time of 96 minutes with a shadow time of 37%, assuming a Nadir pointing design [46], the average orbit power available is 300mW, when considering typical efficiencies of the Electrical Power System.

In order to achieve this estimate, the EPS must make use of Maximum Power Point (MPPT) tracking. An integrated chip solution is being considered as the most viable approach. Another concern when assuming 300mW is the solar cell degradation, for bare cells the End of Life(EOL) was predicted to be 42% for bare TASC cells whilst Spectrolab UTJ cells with cover glass were estimated to have an EL of 92% after two years [47].

The EPS is also responsible for Energy storage, the most common approach in CubeSats has been the use of Lithium-ion polymer batteries (Li-Po). Their disadvantage is the narrow temperature operating region and the limited number of charge/discharge cycles as well as the possibility of capacity degradation under vacuum, if these are not physically restrained [48]. In order to increase the number of charge/discharge cycles it is recommended to keep the depth of discharge (DOD) and charge/discharge rates to a minimum [7]. It is estimated that with a DOD of 30% the Li-Po will have a loss of capacity of 50% over one year [7] [49]. The worst case depth of discharge will occur when the transmitter is active during eclipse time. Assuming a 500mW output RF power for the communications and a 30% efficiency (estimate from commercial transmitters) [50] then 1.66W is required for 6.5

minutes, and assuming that an additional average of 200 mW is being consumed throughout the eclipse time, then a 1000mAH battery will have a DOD of 8%.

Since the energy storage is the limiting factor to a mission's lifetime, super capacitors will be considered as a secondary storage so that when the Li-Po degrades, the mission can continue at a reduced performance. Super capacitors have the advantage of being less sensitive to charging cycles and temperature. They have higher power densities, but they also have a smaller energy density.

The EPS design has to focus on the design of switchable supplies for power distribution and circuit protection such that sub systems are protected from single event latch-ups (SEL). The use of switchable supplies is necessary so that subsystems can be powered down and isolated if needed. Additionally telemetry will be collected to examine the performance of the EPS as well as the solar cell and battery degradation.

5.8. The Power Budget

For the Power Budget to be met, operations have to be scheduled such that the average orbital power is kept to a minimum whilst ensuring that each subsystem's power peak occurs at different times to keep the battery DOD as low as possible. Table 3 shows the preliminary power budget, with estimates as discussed throughout the paper, a 25% margin has been set to cater for any parameters that may not have been factored in.

Table 3 - Preliminary power budget

SUB-SYSTEMS	COMPONENTS	ORBIT TIME [%]	FREQUENCY [PER DAY]	POWER		
				INSTANTANEOUS [mW]	PEAK [mW]	ORBITAL AVERAGE [mW]
EPS	EPS processor*	Continuous	n/a	10	10	10[1]
COMMS	Operation(Beacon)	10%	Each orbit	100	1666	10[2]
	Operation (TX)	7%	2	1666		15
	Operation(RX)	Continuous	n/a	60		60
ADCS	3RW+3MT	TBD	Each orbit	25	TBD	25[3]
	ADS	TBD	Each orbit	30		15[4]
	ADCS Processor*	TBD	Each orbit	10		10[5]
GNSS	GNSS Module	Whole orbit	2	120	120	16[6]
OBC*	µController	Continuous	n/a	15	15	15
ImP	µController	TBD	TBD	TBD	TBD	50
					Total:	**226**

*processors assumed to be identical, power consumption assumes processors are put in low power modes when not in use.

1. The EPS power assumes a processor with lower operating frequency
2. The beacon is assumed to have 13dBm output power, at 100bps CW
3. Initially allocated budget for a tentative ACS, for worst case 100% duty.
4. Attitude determination assuming 2 gyroscopic sensors and 2 magnetometers
5. Initially allocated budget for ADCS processor, for worst case 100% duty.
6. Power consumption from commercially available GNSS module

5.9. The Mass Budget

With the design criteria discussed in this paper, an initial mass budget has been allocated as shown in Table 4, keeping within the 250g limit. In order to work within this budget, 1mm or thinner PCBs have to be considered and a chassis will be designed so as to reduce the mass to a minimum. It has yet to be established whether the remaining budget will be sufficient for an eventual ACS.

Table 4 - Estimate of Mass Budget

SUBSYSTEM	COMPONENTS	MASS [g]	MASS [g]	NOTES
Structure	AL7075 chassis	25	**95.7**	Custom designed and chassis machined in-house
	PCB's	62		Assuming 5 PCBs and 6 PCBs for solar cell faces, 1mm thick
	Connectors	5.7		Data Connectors, RF Connectors
	Fasteners	3		Screws or latches
EPS	Solar cells	11.2	**33.2**	Triple Junction cells, 16cm^2 per face including cover glass
	Battery	17		Measured for a typical 1000mAH Li-Po
	EPS components	5		MPPT, Processor, Sundry passives
COMMS	RF Components	5	**7.3**	Transceiver, RF Amplifier, RF switch
	Antenna	2.3		Beryllium Copper antenna sections of 175 × 0.8 × 0.1mm
GNSS	GNSS Module	10	**10**	Module includes GNSS receiver and patch antenna
ADCS 3RW+3MT	Magnetorquers	8.1	**93.8**	Assuming 3 magnetorquers of 2.7g each for RW de-spinning
	Motors	4.8		Assuming 3 DC micro motors of 1.6g each
	Reaction Wheels	24.6		Assuming 3 reaction wheels of 8.2g each
	Drivers, Sensors	5		Accelerometer, Gyroscope, Magnetometer, 3 Axis
	Contingency	51.3		Any un-utilised mass budget may be distributed elsewhere.
OBC	Microcontroller	5	**5**	Including voltage supervisor, watchdog, external memory
ImP	µController	2	**5**	Including voltage supervisor, watchdog
	Analogue	3		Acquisition signal chain
		Total:	**250.0**	

5.10. Operational Aspects

The initial operational model adopted plans to use two ground stations, one in Malta and another in Birmingham. However, since these two locations are relatively close, this doesn't provide a significant advantage over a single station. An alternative, is to engage the amateur radio community to widen the ground station network which could be set up for data access. For this, the beacon could be used to broadcast the more important data. If necessary, the main transceiver could be scheduled for more frequent data transmissions at the cost of reducing the transmission power and data rates.

As for command telemetry and data analysis, two teams would have to be set up. One team would be responsible for the platform commands and data analysis such as battery voltage, and general system health, so that any faults can be simulated on the flight spares. Commands can then be uplinked to the PQ to modify its behaviour. The second team would be responsible for the scientific mission operations, including payload faults, ImP calibration, and analysis of the collected scientific data.

5.11. The Cost Budget

Development of one-off, high-value devices such as satellites undergoes several phases and several complete systems are built throughout the process. These models are typically termed as follows: Structural and Thermal (STM), Engineering (EM), Qualification (QM), Flight (FM), at least 2x Flight Spares (FS). This means that the development costs must include the development of around 6 complete spacecraft. Although the testing aspects do not contribute the cost of construction of each device, these have been included in the overall development cost figure as shown in Table 5.

Table 5 – Estimated Development/Construction Costs for STM, EM, QM, FM, 2×FS

SUBSYSTEM	COMPONENTS	COST [€]	COST ×6 [€]	NOTES
Structure	AL7075 chassis	100	**3,300**	Custom designed and chassis machined in-house
	PCB's	330		5 PCBs and 6 PCBs for solar cell faces, 1mm thick
	Connectors	20		Data Connectors, RF Connectors
	Other	100		Vacuum-rated adhesives, greases and elastomers
EPS	Solar cells	1,500	**10,050**	Triple Junction cells,16cm^2 per face including cover glass
	Energy Storage	125		1000mAH Li-Po, Ultra-capacitors
	EPS components	50		MPPT, Processor, Sundry passives
COMMS	RF Components	50	**450**	Transceiver, RF Amplifier, RF switch
	Antenna	25		Beryllium Copper Antenna
GNSS	GNSS Module	1,250	**7,500**	Module includes GNSS receiver and patch antenna
ADCS 3RW+3MT	Magnetorquers	10	**1,050**	3 coils of 2.7g each for RW de-spinning
	Motors	15		3 DC micro motors
	Reaction Wheels	50		3 reaction wheels of a composite material
	Drivers, Sensors	100		Accelerometer, Gyroscope, Magnetometer, 3 Axis
OBC	Microcontroller	50	**300**	Including voltage supervisor, watchdog, external memory
ImP	µController	100	**1,200**	Including voltage supervisor, watchdog
	Analogue, RF	100		Acquisition signal chain
Component Acceptance Testing			**2,500**	In house testing, using purpose-built equipment
PQ Pre - Qualification testing			**2,500**	In house testing, using adapted equipment
Pre-Launch Qualification Testing			**5,000**	Certified laboratory testing
		Total:	**33,850**	**Therefore Incremental Replication Cost = €3975**

6 CONCLUSION

In this paper the challenges of designing a PocketQube within the constraints imposed by an Ionospheric impedance probe have been highlighted. A suitable instrument which meets the overall mission requirements has been selected, and the implications of this choice is reflected in the solutions being considered for a feasible mission. The requirements have been discussed in relation to a preliminary top-level design, including a reasonably stable estimate of the mass, power, cost and communications link budgets. The resulting tallies provide an encouraging portrayal of the objectives. No insurmountable technical difficulties were found, and several viable design choices were identified. For these reasons, the project is deemed technical feasible.

Operational feasibility is less evident. However, the sub-divisible nature of the project allows the universities involved to engage a good number of undergraduate and postgraduate students to develop several systems in parallel. Each team gains expertise that will be helpful for debugging after launch. The possibility of reprogramming the PQ in orbit also simplifies the process of recovery, in the case of faults or unexpected behaviour. After a few months, mission execution is also expected to be largely automated.

The financial feasibility of the project is also in line with what was established as acceptable at the outset, where the cost of development nearly equates with the cost of the launch. Therefore, the feasibility of developing a sophisticated PQ spacecraft with a scientific payload has been established in this sense. Moreover, the incremental cost of the device is estimated to be under €4000 which makes a constellation of such devices, a practical proposition.

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