BASIC ISSUES IN ELECTRIC PROPULSION TESTING AND THE NEED FOR INTERNATIONAL STANDARDS

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ABSTRACT

We present the main aspects related to effective and reliable ground testing of advanced Electric Propulsion Systems, with particular care at evidencing key issues which will need to be addressed in order to increase the reliability, confidence and actual application potential of such propulsion systems in space missions. The basic parameters which affect specific types of measurements and tests are addressed, and the many (often overlooked) pitfalls are evidenced: these clearly require the definition and enforcement of a series of standard test procedures and methods, both related to measurements to be performed on EPS's and to the unique identification of the test environment.

1. INTRODUCTION

Electric propulsion represents a key technology for present day space activities, and also among the most promising ones for future developments and space missions. Propulsion systems based on Hall Effect Thrusters (HET) and Gridded Ion Engines (GIE) are particularly interesting for spacecrafts with long operation time due to their relatively high thrust capability, coupled with a specific impulse which is often one order of magnitude higher than latest generation chemical systems and with a fairly long operating life. These systems are operating on the principle of electromagnetic and electrostatic acceleration, as opposed to gas dynamic acceleration occurring in chemical thrusters and arcjet or resistojet-type electric thrusters.

Presently, low and medium power HET and GIE in the 1-5 kW power range are available or under development. In the near term, considering the higher power levels which are expected to be available for propulsion purposes on large geostationary telecommunication satellites within the next decade, the focus will move to GTO (or MEO) to GEO orbit transfers by means of high power Electric Propulsion Systems (EPS) rated in the 10-15 kW power range.

On the opposite range of the size and thrusts scales, the Pulsed Plasma Thruster (PPT) and the Field Emission Electric Propulsion (FEEP) units offer a number of distinct and unique advantages for accurate attitude control, pointing and station keeping of small spacecraft, and are an attractive technology for scientific (drag-free, interferometry, etc.) and commercial (Earth observation) missions. PPT and FEEP units are rated in the 10-50 W power range and offer an high specific impulse coupled with a very small minimum impulse bit.

Although flight demonstration of EPS has occurred in the past, with some successful missions (including in the last few years NASA's Deep Space I, ESA's Artemis, commercial GEO satellites and scientific vehicles), EPS are still considered by many potential customers as a risky option, which is not worth the additional complexity.

EPS are more complicated than traditional chemical propulsion systems, at least due to presence of separated Power Supply Units that add to the cost and potentially reduce the reliability of the system. On the other hand, the additional capabilities that can be achieved by adopting EPS are often enabling a whole new range of missions and tasks to be performed by the spacecraft.

The key to future widespread adoption of advanced EPS lies in the possibility of increasing customer's awareness and confidence in this challenging technology, by fully characterizing and testing EPS in the most appropriate and representative conditions. Extensive tests will allow engine manufacturers to fully understand the propulsion system behaviour in the most diverse conditions, and to identify and eliminate potential failure modes. In turn this extensive test database will provide users with the full confidence that the selected EPS will perform in space as expected, and successfully fulfil the mission objectives.

In order to provide users with the maximum possible confidence in test results, all the instruments and techniques needed to fully characterize EPS should be uniformed and formalized by setting up a recognized standard. Possible sources of misunderstanding and inappropriate selection of test procedures will also have to be identified and clearly defined. This effort would save useful resources during the specification definition phase of future test activities, and would guarantee their quality and applicability towards the ground demonstration of specific aspects.

Because of the challenges of appropriate advanced EPS testing, only a small number of highly qualified and experienced test centres worldwide have the know-how and capability needed in order to provide propulsion system developers and users with the necessary high quality data: this experience should therefore be exploited in order to improve EPS test facilities and procedures. Furthermore the significant investment level required to set up high capability EPS test facilities requires a very efficient usage of available resources, in order to cover the whole range of different requirements.

There are different stages of testing, which any new EPS is undergoing during the development and manufacturing cycle:

- development tests,
- qualification tests,
- acceptance tests at the production stage.

From the customer's point of view, there is little doubt that qualification test conditions must be as close as possible to the actual space-operating environment, as the ultimate test objective is to determine and/or confirm operating parameters of the propulsion system. As will be shown below, the simulation of space environment on ground requires expensive facilities, which are only available in a limited number and which may quickly become the most critical aspect for the development of future high power EPS. For this reason, in parallel to the task of determining actual characteristics of EPS, there is also a strong desire to achieve qualification with minimal investments, time and labour expenses.

The typical objective of development and acceptance tests may be defined as the comparison of actual hardware characteristics with those of a previous version or/and of some standard item. Differently form qualification tests, is may be more important to accurately determine deviation from previous results rather than the absolute value of a specific parameter. Potentially this allows simplifying ground test facility requirements.

The contrasting need of high accuracy and necessary investment is occurring for any technology, and considering the issue of EP standards development it results in the following main objectives:

- identification of test conditions, measurement methods and etc, which allow for the closest characterisation of actual hardware operating parameters in space;
- identification of quantitative dependency of the operating parameters on test conditions and equipment, and identification of acceptable ones for different types of test.

Both objectives are of similar importance from the practical point of view. Nevertheless, for facilitating present application of EPS, implementation of the first aspect seems more urgent, because without a proper measurement the application of any technology is questionable. For this reason the following material is mostly related to the first aspect.

2. BASICS OF EPS GROUND TESTING

Every new thruster or EPS is required (before space qualification is granted) to pass a series of ground tests in order to demonstrate its performance level and life time in operating conditions, compatibility of design solutions with environmental and mechanical loads during transportation, storage, assembling and launch. Although environmental and mechanical tests of EPS have little specific difference in comparison with other space system tests, performance and lifetime tests have many distinguishing features with respect to more conventional (e.g. chemical) propulsion systems. This specificity is determined by the some peculiar aspects of electric thrusters and systems:

- low and ultra low thrust level (from ~1 μN up to about ~10 N);
- high or ultra high vacuum level required for proper operation (for comparison, pressure near a spacecraft shortly after injection into GEO is lower than 10^{-6} mbar, according to Russian data);
- long operation time(up to 18,000 hours for deep space missions);
- specific issues related to the thruster plume, which is composed by high speed (up to 100 km/s), high energy plasma.

EPS testing therefore requires the use of very particular facilities[1,2], tools and measurement devices. These are mostly not available as off-the-shelf items, industrially produced and certified, especially if we consider devices used to measure low thrust and some plume plasma parameters: such devices are often custom designed and realized, offer poor traceability and cross comparison possibility, and their understanding and evaluation is one of the most challenging tasks facing the international electric propulsion community. A few examples clarifying this issue shall be considered below in more detail.

Another important point in EPS testing is the need to properly setting requirements for test vacuum conditions. Ideally these conditions should duplicate actual operation environment, both in terms of the surrounding gaseous medium (chemical composition, density, temperature, etc.) and of the structural elements and interfaces around the thruster (geometry, material, temperature, etc.).
Unfortunately this ideal situation is almost always impossible to achieve (or simply too expensive). Therefore a less restrictive interpretation of the above stated "ideal" requirements is needed, depending on the type of information or confirmation which is expected from a specific test: these "real" requirements are based on the experience of manufacturers and customers with previous thrusters or (more in general) with other space technologies: environmental simulation rather than duplication is achieved.

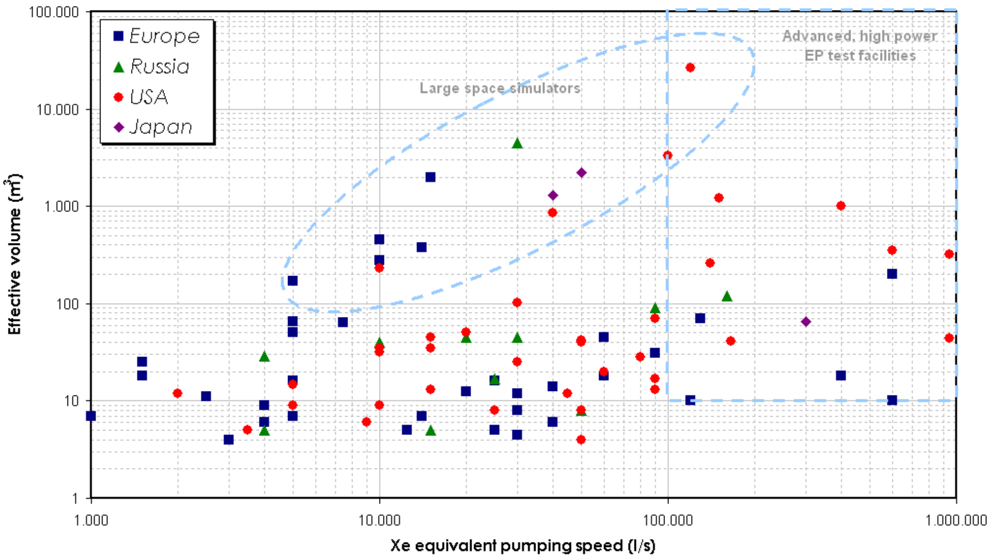

Fig. 1 Pumping performance vs. size graph of selected space simulation/EPS test facilities operating worldwide.

2.1. Vacuum pumping speed

The loose approach (simulation) is however not devoid of problems: depending on the specific tests, requirements should be properly set in order not to waste resources (manpower, funding, time). A typical example is the test background pressure requirement, which in turn (for a given thruster mass flow rate) implies a certain vacuum pumping speed. If we consider the case of new HET being developed, the usual requirement being given to the test people is as follows:
"The facility background pressure during the test will be maintained below $2.6 \cdot 10^{-5}$ mbar, with oxygen, nitrogen and water partial pressure in any case lower than $2 \cdot 10^{-7}$ mbar".
The pumping speed required in order to maintain this vacuum level, Q_p, can be determined from a simple equation, namely:

$$Q_p \approx \frac{m_t p_0}{\rho_0 p} \tag{1}$$

where m_t, p_0, p, ρ_0 are respectively the mass flow rate through the thruster, the propellant pressure at thruster inlet, the required pressure inside the vacuum chamber and the propellant density at thruster inlet. Considering, for example, the case of ground testing a high power HET, with a 10 kW input power and a 18 mg/s xenon propellant mass flow rate, from the presented pumping equation we can determine the minimum pumping speed which ensures the pressure inside the vacuum chamber equal to $2.6 \cdot 10^{-5}$ mbar, turning out to be in the order of 130,000 l/s [Xe], roughly equivalent to 260,000 l/s [N_2] (it is customary in facility performance sheets to quote N_2-calibrated pumping speed). A 10 kW GIE would operate with about 10 mg/s xenon, but because of the order of magnitude difference in required vacuum level (10^{-6} mbar rather than 10^{-5}), the minimum pumping speed should be several times higher. This pumping capability (coupled with a high power dissipation capability) is presently only achieved in 4 facilities in Europe, a similar number in the United States, one in Russia and one in Japan.
Therefore it is reasonable to identify test conditions, which are acceptable for different types of test, but also accounting for the fact that (by necessity) real tests are often performed in limited facilities. In case specific issues are raised by some end-user, such as the possible impact of the thruster plume with the spacecraft elements (for a specific spacecraft design) these are to be addressed with the specific user, by analysis, simulation or test under conditions which are representative of the specific spacecraft configuration. This

approach is not covering the entire thruster operating conditions, but in many cases is an acceptable simulation which allows to reduce the development time and cost.

2.2. Limit pressure for performance and lifetime tests.

In the specific case of HET, the selection of the limiting pressure for performance and life tests has been the object of much consideration over the past few years. Some results are reported below, similar considerations do also apply to other types of thruster such as GIE[3].

Impact of background gas flows on thruster operation and performance is small, if the corresponding mass flow rate, m_b, is small in comparison with the mass flow rate m_t being fed to the thruster. m_b is proportional to the gas number density inside the vacuum chamber and can be easily estimated, known the thruster cross-section area, A:

$$m_b = n_b A \sqrt{\frac{MkT_b}{2\pi}} \tag{2}$$

where n_b and T_b are the background number density and temperature, M is the mass of gas molecules, k is the Boltzmann constant. It is necessary to note that the effective thruster cross-section is higher than the geometrical one, because part of the ionisation and ion acceleration occurs outside of the accelerating channel. Therefore the cross-section area used in (2) must be increased at least by a factor two with respect to the geometric size.

Considering this rule of thumb, and assuming a background gas flow not exceeding 5% of thruster mass flow rate, xenon maximum dynamic pressure requirement for proper SPT-100 testing must be lower than about $6.5 \cdot 10^{-5}$ mbar. Indeed, comparison of performance values obtained during flight and those measured on ground satisfying this requirement shows that difference did not exceed 5%.

The good agreement is understandable, when one considers that ionisation and acceleration of ions inside HET channel have a volumetric nature, and that during thruster tests the most part of background in-flow is chemically and physically the same as freshly fed propellant. Thus, the presence of a small additional fraction of gas does not significantly alter the overall thruster operation. Furthermore, investigation of HET operating with gas mixtures[4] shows that thruster performance is (in the first approximation) linearly dependent on the gas composition.

Another source for determination of acceptable background test pressure is clearly direct data from experiments aimed at understanding the background pressure impact on thruster operation and performance. Results of these studies for SPT-100 show that for pressure higher than $4 \cdot 10^{-5}$ mbar changes in thruster operation are appearing, depending on parameters such as the cathode mass flow rate and the thruster age. However, for a nominally operating SPT-100, according to Fakel data[5], the pressure-induced effects are quite small up to a background pressure at least equal to $5.5 \cdot 10^{-5}$ mbar. Therefore it is acceptable to state that a less restrictive requirement may be defined for performance tests of SPT-100 and similar thrusters, i.e. lower than about $5 \cdot 10^{-5}$ mbar.

In the case of life time tests (long duration in vacuum operation) the same type of statement is however non substantiated yet, as both long duration firing tests on SPT-100 (at JPL[6] and Fakel[7]) were performed at dynamic pressure not exceeding $2 \cdot 10^{-5}$ mbar. In this context the results of PPS-1350 lifetime test at SNECMA[8] (with pressure in the order of $6.5 \cdot 10^{-5}$ mbar) confirmed the possibility of reducing this requirement. Furthermore it must be reminded that early Fakel's SPT life time test yielded results very close to more recent ones, even though those were obtained at pressure of about $6.5 \cdot 10^{-5}$ mbar.

Concerning the partial pressure of residual gases (other than propellant), one can state that these must be significantly lower than pressure of the propellant gas. But the acceptable level was so far not consistently verified. Presence of residual gases inside the vacuum chamber causes the absorption of background gas by thruster components and chemical interactions of this gas with the surfaces. These interactions can lead to the generation of a modified surface layer, with change in electric conductivity, work function and secondary emission factors, sputtering yield, etc.: impact of these parameters is particularly important for life time tests.

It is clear from past evidence that oil free facilities are absolutely essential in order to perform long duration life tests. Early life test of the SPT-100, performed in an oil-diffusion pumped facility, showed a monotonic deteriorating of thruster performance, and mechanical cleaning of the thruster had to be performed. Cryogenically pumped facilities appear to be the best solution in this respect. Furthermore, in order to correctly evaluate the thruster lifetime, it is very important to operate the thruster until the end of the life test without exposing it to the atmosphere, in order to prevent the active surfaces from saturating with water vapour and air, affecting the thruster performance at the beginning of each subsequent cycle.

2.3. Vacuum chamber size for lifetime tests.

The minimum vacuum chamber size for EPS testing depends on the space, which is required for plume characterization, and (more relevantly) on the level of thruster contamination by particles sputtered from vacuum chamber walls, which is considered to be acceptable.

The long (1,000-18,000 hours) operating life capability of new EPS is supposed to be demonstrated on ground, during specific in-vacuum long duration firing. The logistics and technology required for maintaining a complex ion or plasma device (often in the multi-kW power range) in uninterrupted high vacuum operation during more than one year is a formidable task, which is often underestimated during the thruster development stages, and so far only a limited number of systems have been fully life tested on ground. Reducing the interaction of sputtered particle with thruster surfaces is one of the most challenging tasks to be addressed in this respect. Energetic particle sputter erosion is largely more severe in the case of inert gas-propelled (HET and GIE) thrusters than in the case of more conventional arcjets, because of the large difference in sputter yields. As an example, the sputter yield of argon ions accelerated at 1500 eV and impacting on a molybdenum target is nearly $2 \cdot 10^4$ times higher than the yield of hydrogen at typical arcjet energies[9].

Concerning the backflow of particles sputtered from chamber walls, it is clear that this must be small in comparison with the self-contamination due to evaporation and sputtering of thruster elements. The sputter backflow depends on vacuum chamber geometry and size, therefore its upper limit determines the requirement for the minimum vacuum chamber size for a given thruster. Contamination by sputtered particles can have definite impact on the insulators, electrodes (anode, cathode, neutralizer) and (for HET) discharge chamber wall surface properties.

For typical SPT-type HET, the erosion rate in the plasma channel ranges between 5 and 30 Å/s, while the re-deposition rate of this eroded material is about 1 Å/s[3]. Based on these results a requirement of less than 0.1 Å/s sputtered material return rates has been stated for SPT-100 life testing[10]. In the case of graphite-lined test facilities the issue of loss of insulation on active surfaces is also open.

For a cylindrical facility, a theoretical sputter return rate model was developed, accounting for the decay of ion current density along the vacuum chamber and for the experimentally measured sputter yield of wall materials. Based on these results, a facility diameter equal to 1.5 m and a length equal to 2.5 m was suggested for SPT-100 testing; however experimental tests performed on SPT-100 in both JPL and Front Range Research facilities showed a lower value (about one half) of the sputter return rate than predicted by theory, suggesting that dimensional requirements may be relaxed from that usually suggested by simple analytical models. It must also be noted that a large facility diameter is also required to minimize wall thermal loading on the facility. The above conclusions could be verified by results of SPT-100 and PPS-1350 lifetime tests, which can be used for the preliminary formulation of a requirement on the vacuum chamber sizes for HET lifetime tests (the three tests were performed using different facilities at JPL, Fakel and SNECMA). Considering that SNECMA test chamber was the smallest, and that during the life test no contamination effect was evidenced on thruster, this chamber can be used as the reference case.

We assume that in the case of a new HET, all the significant parameters (plume distributions, dependence of sputter yield on thruster operation mode, etc.) scale with the discharge power, and we want to define the vacuum chamber size which is producing a contaminating flow during the new thruster test which is not worse than what obtained during the reference test. It can be found that the vacuum chamber diameter and length must be scaled as follows:

$$D \approx D_o \sqrt{N / N_o} \tag{3a}$$

$$L \approx L_o \sqrt{N / N_o} \tag{3b}$$

where D, L, D_o, L_o are the diameter and length of new and reference chambers, N, N_o are the power for new and reference thrusters respectively. Equations (3a) and (3b) were obtained assuming that the sputtered particle deposition rate is inversely proportional to the characteristic size of the vacuum chamber.

As mentioned above, estimations based on simplified analytical models can give only provide a rough approximation of the sputtered particle backflow. Furthermore, under high vacuum conditions, the free path of particles is comparable or larger than the vacuum chamber size, and equations such as (3a) and (3b) cannot provide adequate results. In order to overcome these issues, application of direct simulation codes (Particle In Cell coupled with Direct Simulation Monte Carlo, PIC/DSMC) to the full configuration of the thruster inside the vacuum chamber[11] (with representative geometry and pumping distribution) has been attempted with good results[12]. Development of such codes is underway in Europe and the USA.

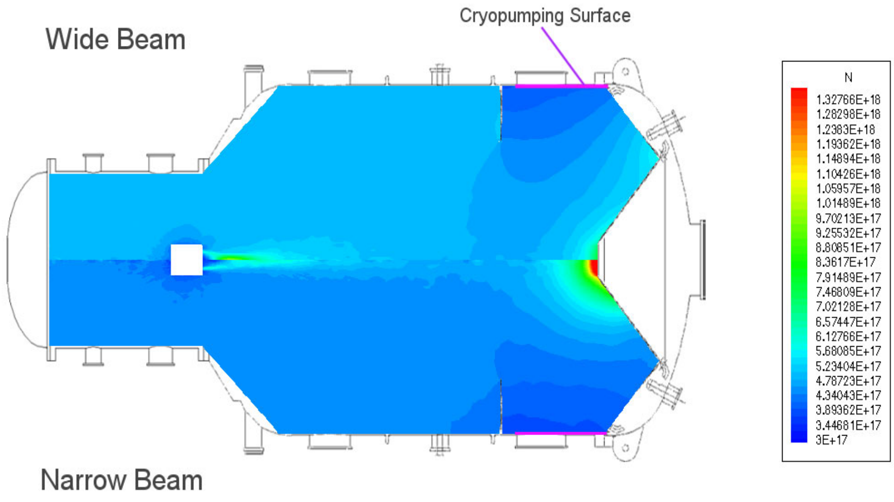

Fig. 2 PIC/DSMC simulation of Alta IV-4 facility, performed for two 3 kW EPS: a wide beam HET (top) and a narrow beam GIE (bottom). Same chamber yields different effective pumping speed: 130,000 l/s for HET, 150,000 l/s for GIE.

Although the above discussion was mostly related to HET, the same considerations apply to GIE. However GIE generate a more collimated ion beam, with higher ion energy, therefore for the same power level the facility diameter requirement may be reduced and the length increased with respect to what is necessary for HET.

2.4. Limit pressure and chamber size for plume measurements.

Both the limit test pressure and minimum vacuum chamber size are influenced by the required plume characterization method. In general, two approaches can be used for plume characterization: either measuring all the plume parameters of interest across the maximum possible volume[13] (which requires a large vacuum chamber and a very high vacuum in order to reduce interaction of plume particles with background gas) or measuring particle flows on a relatively small reference surface across the plume and calculating plume particle dynamics further downstream using a mathematical model of the plume expansion[14]. The second approach is typically used nowadays for HET plume characterization, because it allows a fast and relatively inexpensive plume characterization: the required dynamic pressure level can be increased proportionally to the ratio between the chamber size needed for the former method and the size of the reference surface used for the latter.

More in detail, the second plume measurement strategy needs two assumptions to be satisfied:

- the background gas should not significantly disturb the plume processes along the path between the thruster and the reference surface;
- the characteristic size of the reference surface must be significantly larger than thruster size.

The first assumption states that plume parameter measured on the reference surface are representative of correct operation and can thus be used as an input data for the further calculation of plume dynamics. The critical pressure to satisfy this assumption can be estimated considering the density ratio between the background gas and the neutrals being released by the thruster. For HET the neutral flow typically does not exceed 15% of the total mass flow rate through the thruster m_t. For SPT-100 the total mass flow rate is about 5 mg/s, therefore the neutral flow does not exceed 0.75 mg/s and the main source of neutrals is the cathode/neutralizer. If we assume that the neutral mass flow rate released by the cathode, m_n, has a distribution according to a cosine law and that neutrals are only significantly released at the cathode, we can obtain the following expression for the maximum thruster-released neutral flow, m_{nsp}, through a reference surface at a distance R from the cathode exhaust:

$$m_{nsp} \approx \frac{m_n}{\pi R^2} \tag{4}$$

We can also estimate the maximum thruster-released neutral density n_n, by assuming that these have thermal velocity in equilibrium with the cathode emitter temperature. Again, in the case of SPT-100 with LaB_6 emitter at a temperature of about 1,500 C we have $v_n \approx 5.33 \cdot 10^4$ cm/s, and for R=1 m we have $n_n \approx 2 \cdot 10^{11}$

cm^{-3}. Therefore we can define the requirement on the background density to be lower than $2 \cdot 10^{10}$ cm^{-3}, which in the case of a room temperature chamber results in a pressure limit of $1 \cdot 10^{-6}$ mbar.
Another approach for identifying a critical dynamic pressure for plume measurements, is directly considering the plume interaction with the background gas. It is known that the most relevant type of interaction between accelerated ions and neutral atoms of the same nature is the charge exchange (CEX) process. The cross-section for this process is increasing when the ion energy (velocity) is decreasing, and is about $5 \cdot 10^{-15}$ cm^2 for Xe ions with energy in the order of 100 eV [15]. Therefore in order to have a CEX ion/background ratio below 0.01 (i.e. not more than 1% of the fast ions are colliding with background neutrals) over a distance of 1 m, the background pressure should not exceed about $1 \cdot 10^{-6}$ mbar, which is in agreement with the previous estimate.
As a whole, the above data show that the dynamic pressure requirements for plume measurements are more severe than those for performance tests. This conclusion is confirmed by NASA GRC measurements of the ion current distribution in SPT-100 plume[16], showing that at background pressures between $8 \cdot 10^{-6}$ and $3 \cdot 10^{-5}$ mbar the ion current distribution in the peripheral part of the plume is significantly different from what obtained at $3 \cdot 10^{-6}$ mbar. If plasma density distributions are of interest, these (being significantly affected by CEX) need to be performed at low background pressure (in the order of $1 \cdot 10^{-6}$ mbar). The same applies to slow ion backflow towards the spacecraft surface, as this is basically determined by CEX process.
On the other hand, for the part of the plume within 30 degrees from the axis, NASA GRC results show tha measurements retain an acceptable accuracy up to a background pressure of about $3 \cdot 10^{-5}$ mbar, which fact (considering that the near axis plume region is comprising the most energetic ions) implies that indirect thrust vector measurements and other measurements only concerned with ions above 20-30 eV can be reasonably be performed at pressure up to $5 \cdot 10^{-5}$ mbar.
Depending on the main goal of the specific test, different test requirements can be specified for plume measurements. Careful consideration is necessary when planning a test, in order not to over-restrict the environmental conditions (with unnecessary costs and large accessibility issues) while preserving the most important features of the studies phenomena.

3. INSTRUMENTS FOR MEASUREMENT

3.1. Thrust and plume measurement.

EPS testing requires the use of very carefully designed measurement and diagnostic devices, especially when these devices should operate for the whole duration of thousands-hour tests, exposed to the aggressive action of the plasma/ion beam and in a critical thermal environment. The test environment may pose different requirements from what is preferred for performance testing or life time qualification.
Although usually not an issue from the facility point of view, the accurate and possibly high bandwidth measurement of thrust is a key aspect in the characterization of a new thruster. In this respect advanced EPS's have an intrinsic disadvantage, i.e. the very low thrust to mass ratio, which turns a conceptually simple force measurement task into a challenging engineering problem. In fact, the high sensitivity, which is required in order to resolve thrust variation to 1% or better, poorly matches the need for time resolved measurements with a meaningful bandwidth (well in excess of 100 Hz). Thrust balances are often designed and built for a specific thruster, and their performance is highly variable and not always consistent. This aspect is particularly critical in FEEP thrusters due to the extremely low force (micronewton level thrust and sub-micronewton level thrust noise) to be measured over an extended low frequency range (10^{-4} Hz to 1 Hz). Isolation from seismic and other noise sources, and the need of having determined and repeatable results are the main issues[17].
Considering plume measurements, CEX induced backflow of slow ion is one of the most difficult parameters to be properly measured (i.e. within 5-10%). The difficulty is connected with the necessity to discriminate between fast and slow ions across the same control surface and to determine the directed velocity of slow ions: this is presently performed by a clever use of intrusive electrostatic probes (e.g. Retarding Potential Analyser, RPA), however the level of uncertainty connected with this type of instrument is still high. LIF methods may be helpful in this respect[18], but they still have to undergo extensive development and testing in order to be fully understood.

3.2. EMI/EMC characterization.

EMI/EMC characterization is an important step in the flight qualification of a new EPS, as this experimental characterization is capable of providing useful information with respect to the EPS successful integration on an operating spacecraft.

EMI characterization tests are quantitative. Electromagnetic emissions measured in volt, ampere, tesla, or volt per meter are compared to specification values. Susceptibility to, or immunity from, specification values of volt, ampere, tesla, or volt per meter is also measured. On the other hand EMC tests are qualitative. The term denotes the electro-magnetically compatible simultaneous operation of different equipment. EMC can be defined by the absence of EMI, but EMC is more than that. An EMC test is performed at some level of system integration. EMC is ascertained by energizing equipment A, determining proper operation, energizing equipment B, and noting whether or not equipment A continues to operate as before without any degradation. The EMC test results can be summarized as a square matrix of victims and culprits.

The successful conclusion of the EMI quantitative tests is a reassuring indication that the final EMC test will also have the desired outcome. Failure to meet EMI requirements may indicate a need for redesign, but typically further analysis is first performed to determine if the particular failure is likely to cause an EMC problem. For example, an equipment emission that exceeds the radiated emission (RE) limit by 20 dB at 100 kHz may not be serious if the overall system for which the equipment is destined does not utilize the spectrum below 2 MHz.

In order to allow for EMI measurements of an operating EPS, with the receiving equipment positioned outside of the vacuum vessel, an electromagnetically transparent vacuum chamber would be needed. The most commonly employed solution for such applications is using a composite construction, with a fibre reinforced polymeric matrix (glass vacuum jars have been used in the past but their applicability is limited to very small thrusters operating for a short time duration). The amount of power dissipated by a medium penetrated by EM waves is a function of two parameters: the dielectric constant and the loss tangent. The relevant equation is:

$$W = 0.5 f \cdot E^2 \cdot \varepsilon \left(\tan\delta \right) \tag{5}$$

where W is the dissipated power, f is the wave frequency, E is the intensity of the electric field fluctuation, ε is the dielectric constant and tan δ is the loss tangent. For this reason it is important to use a material with the lowest possible dielectric constant and loss tangent, especially in the high frequency range. In general, both ε and tan δ are functions of the EM wave frequency, therefore materials have different transparency characteristics in the different frequency bands. Relatively inexpensive solutions, such as a E-glass fibres in a polyester resin matrix are useless for measurements at frequencies exceeding 10 GHz (X-band), on the other hand quartz fibres in a cyanester resin matrix maintain their applicability well beyond the Ka-band and are therefore the preferred choice for space applications.

All the elements inside the vacuum vessel used to support thruster and its equipment should be manufactured using epoxy/S-Glass or Plexiglas or another EM transparent material. Also, all the elements contained inside the vessel, with the exception of the thruster under test, should be constructed in such a way not to be source of electric interferences. Plexiglas flanges should be used in order to build suitable permanent feedthroughs for power and fluid connections to the thruster. Steel bolts used for mounting the flanges should be replaced by dielectric plastic fixtures after a sufficiently high vacuum level is achieved (bolt compression being replaced by pressure difference), in order to remove any perturbation in the measured EMI field.

Clearly some conducting components may not be avoided (for instance the vacuum pumps and valves) therefore the facility should be characterized for absorption and reverberating effects over the considered frequency range before the thruster is installed. This is particularly the case when a dielectric small chamber is connected to a larger steel facility for taking advantage of an existing vacuum installation: reverberation from the metallic walls through the open connection should be measured and taken into account in data post-processing and interpretation. The usual non-reverberating type enclosures shall in any case be needed around the dielectric vacuum vessel, fully enclosing the facility and the receiving/transmitting equipment (as per the relevant EMI/EMC standards).

A second possible approach to EMI measurements, applicable to most facilities with conductive chamber walls, is the pre-determination of the chamber RF reverberation characteristics using standard emitting antennas, which, in turn, can be easily calibrated. This approach was used for several programs in Russia, and also in the USA for testing SPT-100 and D-55 thrusters.

Implementation of both approaches (dielectric walls and reverberation calibration) still requires significant research and verification effort, and presently there is not enough data to give strong preference to one of them (although commercial customers seem to prefer the first type of EMI test data).

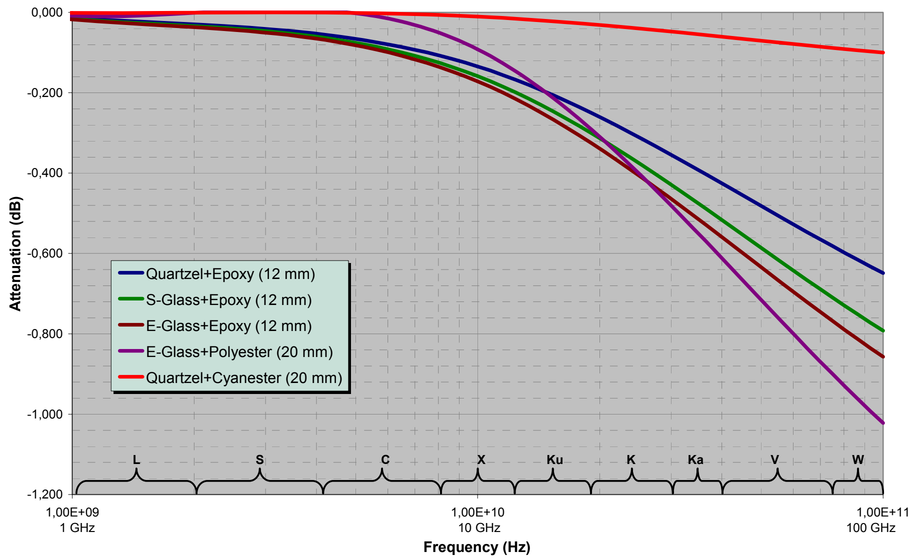

Fig. 3 RF transparency of various dielectric vacuum chamber walls (thickness sized for 1 m dia. vessel).

4. THE NEED FOR INTERNATIONAL EPS GROUND TESTING STANDARDS

Although only the most frequently occurring sources of misunderstanding and controversy in EPS tests were above addressed, many other aspects exist which require a careful consideration and proper understanding in order to sort out any possible source of uncertainty.

It seems today that EPS have reached the level of maturity requiring the implementation of appropriate standards: testing standards are probably one of the most important aspects to be addressed in order to provide a firm ground to evaluate the performance and characteristics of advanced EPS.

The scope of an EPS ground testing standard would be wide, and would significantly affect future EPS development. In particular it may:

- help EPS developers and manufacturers to better understand the interaction of their systems with ground facilities, and to better predict (or extrapolate) the in-space performance;
- help users/customers to better understand the test data which the EPS supplier provide them, in order to increase their awareness and confidence in the system capability;
- help both users/customers and developer/manufacturers to choose the best possible facility and measurement method in order to properly characterize a specific EPS aspect;
- help Agencies and space policy bodies to better understand the need of advanced test facilities for future EPS, and to direct funding and support appropriately;
- help test service providers to focus on a relatively small number of diagnostic and test methods, being fully confident that the customers will rely on those (standardized) methods for assessing the performance and characteristics of their systems.

Limiting our attention to the few tricky aspects reported in the above paragraphs, the EPS ground-testing standard may cover such topics as:

- how to properly define the background environment for a test and to uniquely identify this requirement;
- how to define and measure the effective pumping speed for a facility/EPS combination, and possibly defining and implementing a standard way to measure a nominal effective pumping speed which is relevant to EPS testing;
- how to design, realize, calibrate and operate a standard thrust balance, which may be operated in the same way in a large number of different facilities therefore providing a comparable database of thrust measurements in traceable conditions;
- how to implement the standard plume measurement techniques in such a way to have measurements which could be compared from facility to facility, from team to team, and to have measurements which accuracy could be traced back to a common, fully characterized and understood reference case.
- how to define a standard post-processing algorithm for defining thrust vector direction from plume measurements, possibly developing a standard integrated probe with the capability of simultaneously measuring all the parameters which will then be used for post-processing.

From a historical point of view, we note that the definition and implementation of standards have (in the past) always been an essential prerequisite for the wide acceptance of a specific technology. Taking the "black-magic" feeling away from test data, reducing the unexplained scatter in data from different facilities, providing a unique measurement reference when comparing data from different thrusters and/or manufacturers will increase user/customer confidence and support in advanced EPS and ultimately help towards their application.

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