Diagnostics for In-space Testing of High Power Electric Thrusters on the International Space Station

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Abstract: High power electric propulsion provides the potential to move large cargo and manned spacecraft beyond low earth orbit. The maturing of these high power propulsion devices depends on the ability to test them in a relevant environment. Earth based vacuum facilities are capable of testing low power devices for extended test durations and even high power devices for very short test durations. However, these short test durations along with the boundary conditions imposed by the vacuum chamber wall can affect the operation and understanding of high power electric thrusters. A design study is underway to determine the design and feasibility of placing a high power electric propulsion test platform on the International Space Station (ISS). This paper discusses the diagnostic requirements and preliminary design of the diagnostics suite to be used on the high power test platform.

Nomenclature

a	=	acceleration
$\vec{B}(\vec{r})$	=	magnetic field
F	=	thrust

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g_0 = acceleration due to gravity at sea level
$I(\lambda)$ = intensity of line emission as a function of wavelength
I_{sp} = specific impulse
$\dot{m}$ = mass flow rate of propellant
$n_e(\vec{r})$ = electron number density
$n_j(\vec{r})$ = ion/neutral number density
p = operating back pressure
P_j = jet power
$\vec{r}$ = spatial position in plume
S = volume flow rate
$T_e(\vec{r})$ = electron temperature
$T_j(\vec{r})$ = ion temperature
v_{exh} = exhaust velocity of propellant
$\vec{v}_j(\vec{r})$ = ion velocity
$V(\vec{r})$ = plasma potential
w = molar mass
λ = wavelength of line radiation

I. Introduction (Motivation for test platform)

ABSOLUTE background pressures less than 10^{-4} to 10^{-5} torr are required to accurately characterize many electric propulsion devices. As electric propulsion technology matures and high power (>100 kW) thrusters become viable, electric propulsion becomes an enabling technology for moving large payloads around the solar system and reducing the duration of human exploration missions. Testing of high-power electric propulsion devices places extreme burdens on terrestrial vacuum facilities due to vacuum pumping required and physical size. The vacuum pumping capacity can be determined from the following definition for jet power relating jet power, mass flow, and exhaust velocity:

$$P_j \equiv \tfrac{1}{2}\dot{m}v_{exh}^2 \tag{1}$$

where the exhaust velocity can be replaced by $I_{sp}g_0$,

$$P_j = \frac{1}{2}\dot{m}(I_{sp}g_0)^2 \,. \tag{2}$$

Equation 2 can be rewritten to determine the mass flow rate for a desired jet power and specific impulse,

$$\dot{m} = \frac{2P_j}{I_{sp}g_0} \,. \tag{3}$$

The mass flow rate determined in equation (3) is independent of propellant species. For example, a thruster with a jet power of 100 kW operating at 4,000 s specific impulse will have mass flow rate of 0.13 grams/s. Vacuum pumping rates are usually given in units of volume per unit time, i.e. L / s, and are generally gas species dependent. Terrestrial vacuum facilities are limited to about 10^6 L/s for some of the largest chambers. Vacuum pumping may be increased for some gas species that are condensable on cryogenic panels but is limited by the surface area of the cryo-panels to capture the propellant. Volume flow rate, in units of L /s, is related to the mass flow rate at standard temperature but non-standard pressure (vacuum) by:

$$S = \dot{m}\left(\frac{760}{p}\right)\left(\frac{22.4}{w}\right) \tag{4}$$

where p is the desired pressure in torr and w is the molar mass of the gas species to be pumped in units of g/mol. For 0.13 g/s mass flow rate and 10^{-5} torr desired background pressure, the volume flow rate is 2.2×10^{8} L/s for hydrogen and 1.7×10^{6} L/s for xenon at steady state flow. Quasi-steady operations of order < 0.1 s may be performed by simply starting with a vacuum chamber at high vacuum and back filling to the maximum allowable background pressure. This technique provides a method for researching some of the high power plasma propulsion concepts but at very short test durations.

Terrestrial vacuum facilities also impose other conditions not representative of the space environment such as the closed boundary conditions from the vacuum chamber walls. Backflow, flow of propellant in the opposite direction of the thrust vector, can only be accurately quantified when sufficient vacuum chamber volume (on the order of several to tens of meters) allows the plasma to expand [1]. In addition, electric fields present in the thruster plume can be affected by the presence of the conducting vacuum chamber walls. These boundary conditions can provide a stabilizing effect on the plasma that is only present when the vacuum chamber walls are present. Some have used non-metallic chambers to avoid the effects from electrically conducting chamber walls but surface charging can still occur. Terrestrial vacuum facilities are important for development of low power electric propulsion and initial testing of high power devices, since they provide easy access to the thruster prototypes. However, high power electric thruster performance and operation will only be accurately quantified when tested in space. At low power, EP devices can be tested using ground based vacuum chambers but are still subject to the close boundary conditions of the vacuum chamber, which in some cases will interfere with attempts to accurately determine characteristics such as backflow rates.

The environment of space, with infinite vacuum pumping and open boundary conditions, provides the best, and in some cases the only possible, environment for accurate evaluation of high power EP devices. Also, a test platform designed for testing of high power EP devices can provide a unique test capability for low power EP devices where open boundary conditions allow accurate measurement of characteristics such as backflow. A design study has been initiated for a High Power Electric Propulsion Test Platform to be attached to the International Space Station [2, 3]. This paper addresses the requirements and initial design of the diagnostics suite associated with the test platform. The diagnostics must be well planned to provide the best use of the test platform. A brief description of the test platform is provided in section II. The requirements developed for the diagnostics suite will be listed in section III. An initial design concept of the diagnostics will be provided in section IV and a summary will be provided in section V.

II. Description of test platform

The design of the test platform must be based on well-defined requirements. The design study team for this effort, along with inputs from many in the electric propulsion community, established a set of design requirements and some of the top level requirements are listed here[3]. Ref. 3 provides more details. The thruster is referred to as the test unit.

- **Electrical Power**: The test platform will deliver 1 MW of electrical power to the test unit (an EP device) for 1 minute.
- **Propellants used**: For generality, various propellants are being considered ranging in molecular weight from hydrogen to xenon. The propellant used will affect some of the diagnostics and this must be taken into account during the final design to allow the most general use of the diagnostic suite.
- **Data Acquisition**: Up to 500 channels of data acquisition at data rates at or above 1 MS/s will be available for the diagnostics suite as well as user supplied diagnostics and instrumentation that are internal to the test unit (thruster).
- **Test unit (thruster) volume:** A 1 m X 1 m footprint with a height of several meters will be available for the test unit. The test unit must remain inside this volume to avoid interference with other test platform associated hardware. Each test unit will interface with the test platform via a standardized test

unit carrier. The test unit will be integrated into a carrier on the ground and then attached to the test platform at the ISS.

III. Requirements for diagnostics

The goal of the diagnostics suite is to provide information to evaluate the performance of the thruster as well as physical understanding of the thruster to facilitate optimization. The diagnostics are divided into three categories. 1) Performance measurement diagnostics, 2) Operational or investigative diagnostics, and 3) ISS monitoring diagnostics. It is required that the thrust, specific impulse, and electrical efficiency be accurately determined for high power EP. The operational diagnostics provide knowledge that can be used to optimize the thruster. For example, the information gained from the operational diagnostics will include beam divergence, backflow, acceleration mechanisms, neutral pressure effects, and ionization fraction.

A. Performance diagnostics

1. Thrust

The maximum thrust allowed by the thruster is dictated by the acceleration constraints of ISS and is being evaluated, but has not yet been determined. The maximum thrust achievable from a thruster is dependent primarily on the available power. The maximum power available to be delivered to a thruster is 1 MW with a test duration of one minute. Test units requiring less than 1 MW can operate for more than 1 minute. A thruster with input power of 1 MW, 50% electrical efficiency, and 4,000 s specific impulse will yield a thrust of 25 N. A minimum detectable thrust requirement of 0.5 N was placed on the diagnostics. The test platform can accommodate thrusters with less thrust but the facility thrust measurements may not be reliable. The 0.5 N thrust limit was derived assuming a jet power of 25 kW and a specific impulse of 10,000 s which also corresponds to 5 kW of jet power at 2,000 s specific impulse. Figure 1 shows the detectable ranges of the thrust with regard to jet power and specific impulse. The dashed line in Figure 2 represents 0.5 N.

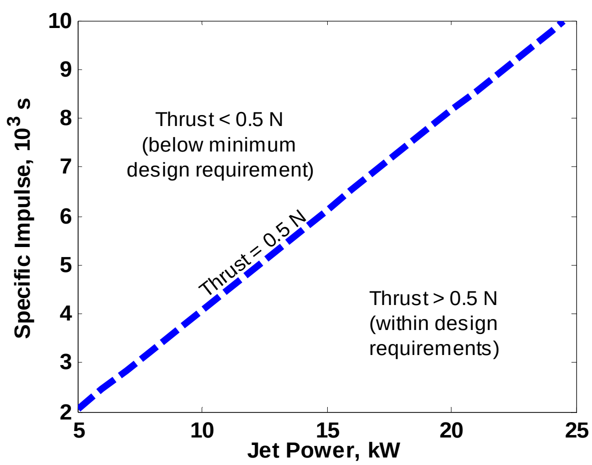

Figure 1. Graphical representation of detectable thrust

2. Specific Impulse

Another important performance parameter is the specific impulse itself. The specific impulse is determined from the relationship

$$I_{sp} = F/\dot{m}, \quad (5)$$

where F is the thrust and $\dot{m}$ is the mass flow rate of propellant. The specific impulse provides a measure of the propellant utilization efficiency. In particular, the requirement for a specific impulse measurement leads to a requirement to measure the propellant mass flow rate. A programmatic decision was made that each test unit will provide its own propellant supply and that the mass flow rate measurement will reside with the test unit. The accuracy of the mass flow rate measurement shall be sufficient to evaluate the performance of the thruster with better than 5% uncertainty.

3. Electrical Efficiency

Electrical efficiency will be determined by measuring the total power delivered to the thruster through the power interface at the test platform with accuracy sufficient to evaluate the performance of the thruster with better than 5%

uncertainty. Some power processing may be required if the test unit needs nonstandard power, and therefore some power management and distribution (PMAD) hardware must be included by the user in the test unit [3]. This PMAD hardware must also be encompassed in the volume allowed for each test unit.

B. Operational diagnostics

Generality of the test platform, the ability to test several types of EP devices, requires the diagnostics to interrogate plumes from a variety of propellants. Schematically, the identification of the properties to be measured precedes the selection of diagnostic techniques, but in practice the identification and selection are interrelated. Ideally, the targeted measurements would include the distribution function for all particles (charged and neutral) everywhere in the plume, the electric and magnetic fields in and around the plume, and the distribution of emitted radiation at all frequencies. However, it is not practical to measure everything, and some measurements, while feasible, would add prohibitively to the complexity and expense of the design. The nominal requirements placed on the diagnostics suite, then, represent a balance between the need for information to understand the plume processes and the practical demands of making the measurements. These nominal requirements, as currently determined, are listed below. The species are electrons, ions, and neutrals (and photons), where the ions (j = 1, J) may be from different gases and may also be of different charge states. The most important cases are:

(1) ion directed velocity $\vec{v}_j(\vec{r})$

(2) ion density $n_j(\vec{r})$

(3) electron density $n_e(\vec{r})$

(4) electron temperature $T_e(\vec{r})$

(5) ion temperature $T_j(\vec{r})$

(6) plasma potential $V(\vec{r})$

(7) magnetic field $\vec{B}(\vec{r})$

(8) line/continuum radiation spectra (visible/UV) $I(\lambda)$

Lifetime of a thruster is another important parameter to consider. Erosion of components and requirements regarding investigation of the component erosion will be addressed. High-resolution cameras with remote manipulation may be sufficient to provide elementary erosion studies. Some components may be retrieved and returned to the ground for evaluation if the ISS crew work loads permit.

C. ISS monitoring diagnostics

EP devices will be evaluated with respect to safety and risk to the ISS prior to being placed on the test platform. However, some diagnostics will be placed on or near the ISS to verify that any adverse effects from possible particle impingement, surface charging, and EMI are not present. These diagnostics may comprise a subset of the operational diagnostics and will have a high priority since they will be used to verify safe operations. The possible effects from various EP devices are being evaluated and the requirements for ISS monitoring have not yet been determined.

IV. Description of diagnostics to meet requirements (conceptual design)

The following are conceptual designs to meet the requirements listed in Section 3. Details of exact location and spacing of the diagnostics will be decided in the final design.

A. Performance diagnostics

1. Thrust

The thrust will be determined using accelerometers to measure the acceleration and using the known mass of ISS. The thrust will then be determined by the simple relation:

$$F = ma \qquad (1)$$

where a is the acceleration caused by the thruster only and m is the mass of ISS. The sum of all other components of acceleration such as atmospheric drag in the direction of the thrust vector must be determined to accurately quantify the thrust produced by the thruster. One possible solution may be accomplished by simply turning the thruster on and off to observe the change in acceleration due to the thruster. The specific means for resolving this issue will be determined in the final design phase of this effort. However, it is important during early phases to identify such issues and to determine if adequate resolutions are feasible

The mass of the ISS at completion is expected to be 454,000 kg. A 0.5 N force directed through the CG of the ISS will provide an acceleration of $0.11\mu g$, where $1g$ is $9.81\,\mathrm{m/s^2}$. This is the minimum acceleration requirement for the thrust measurement.

A flight-qualified unit already exists to sense accelerations as small as $0.1\,\mu\mathrm{g}$ and is part of the Microgravity Acceleration Measurement System (MAMS)[§§] in the ISS. Although a MAMS unit is present on ISS, a separate MAMS unit should be placed on the HPEP test platform to help reduce uncertainties with regard to dynamic effects in the ISS structures and to permit the alignment of the thrust vector with the most sensitive axis of the MAMS. There are other accelerometer systems that may also be feasible.

Load cells placed between the test unit and test platform may also provide an accurate determination of thrust but the sensitivity and accuracy of this method are subject to tare loads. These tare loads are a source of error in ground based thrust stands, and careful engineering of the thrust stand is required to reduce or quantify these errors [4]. The tare loads result from deflections of cables and other hardware connected between the thruster (or carrier) and the test platform. Although this method introduces many of the difficulties in ground-based thrust stand measurements, these difficulties have been addressed and most have been resolved. The thrust measurement systems mentioned here are not mutually exclusive, nor exhaustive, and a particular method (or methods) will be selected during the initial design phase.

2. Specific Impulse

A calibration scheme similar to that used in terrestrial EP laboratories should be used for in-situ calibration of the mass flow measurement system. This technique requires a calibrated volume and pressure gauges and has been used routinely in ground-based laboratories. A calibrated mass flow meter will be required with each test unit and the unit must have appropriate electrical connections (power and signal) to connect to the test platform data acquisition system. Again, the accuracy and resolution of the mass flow measurement system shall permit performance evaluation of the thruster with better than 5% uncertainty. Distribution of the propellant inside the carrier to various parts of the thruster will be the responsibility of the user.

3. Electrical Efficiency

The current and voltage will be measured using transducers at the test platform-to-test unit interface. These measurements will provide the total power delivered to the test unit. Data acquisition channels will be provided for transducers in the test unit PMAD system to allow the user to monitor the power distribution within the test unit.

B. Operational Diagnostics

The thruster plume region is divided into two regions, near field plume and far field plume, defined below. The diagnostics requirements for the two regions will differ, in part because the plasma density will drop by several orders of magnitude through the plume. Candidate diagnostic techniques to measure the properties identified in Section 3 are listed below.

(1) retarding potential analysis (RPA);	species indiscriminate; intrusive
(2) laser induced fluorescence (LIF);	species restrictive; non-intrusive
(3) Langmuir probe measurements (LP);	electrons; intrusive
(4) microwave interferometry (MI);	electrons; non-intrusive
(5) emission spectroscopy;	species ID; line-of-sight
(6) Thomson scattering;	electrons; difficult

The application of these techniques to the target quantities is summarized in Table 1.

[§§] http://microgravity.grc.nasa.gov/MSD/MSD_htmls/mams.html

Table 1. Summary of diagnostic applications.

	RPA	LIF	Langmuir probes	microwave interferometer	emission spectroscopy	Thomson scattering
$\vec{v}_j(\vec{r})$	yes	yes			some	
$n_j(\vec{r})$					species ID	
$n_e(\vec{r})$			yes	yes		high-density
$T_e(\vec{r})$			yes			high-density
$T_j(\vec{r})$	yes	yes			some	
$V(\vec{r})$			yes			
$\vec{B}(\vec{r})$		yes				
$I(\lambda)$					yes	

1. Near Field Plume

The near field plume is defined as the region beginning at the plasma source region of the thruster and extending downstream for a distance of (approximately) one meter. This region will have the highest plasma density and provide the most data for understanding the physics operation of the thruster. Figure 2 shows the near field plume region of an EP device in a ground laboratory with a Langmuir probe parked just outside the plume [5]. Some areas may not be accessible near the plasma source if it is embedded inside the thruster. The near field plume region extends radially from the centerline of the thruster or thrusters to a radius of 0.5 m.

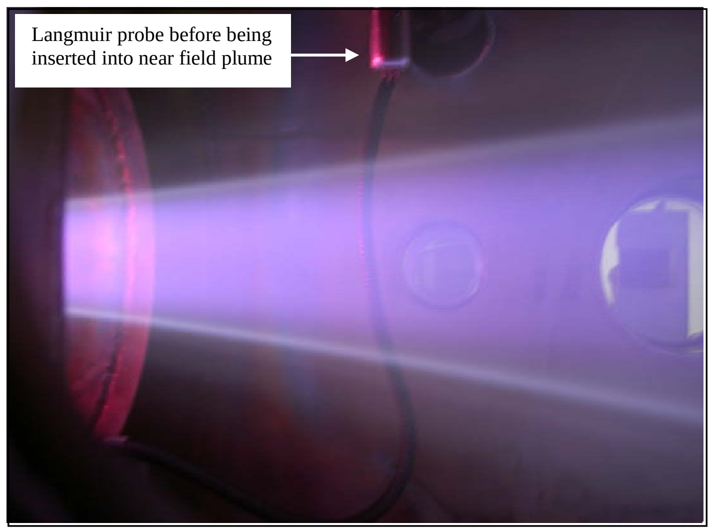

Figure 2. Near field plume of a thruster with a diagnostics parked near plume

Figure 3 shows a conceptual sketch of the near field plume diagnostics. The diagnostic instruments translate / rotate to give a 3-D map of the plasma properties in the near field plume region. The layout shown is preliminary and does not necessarily represent the latest test platform design.

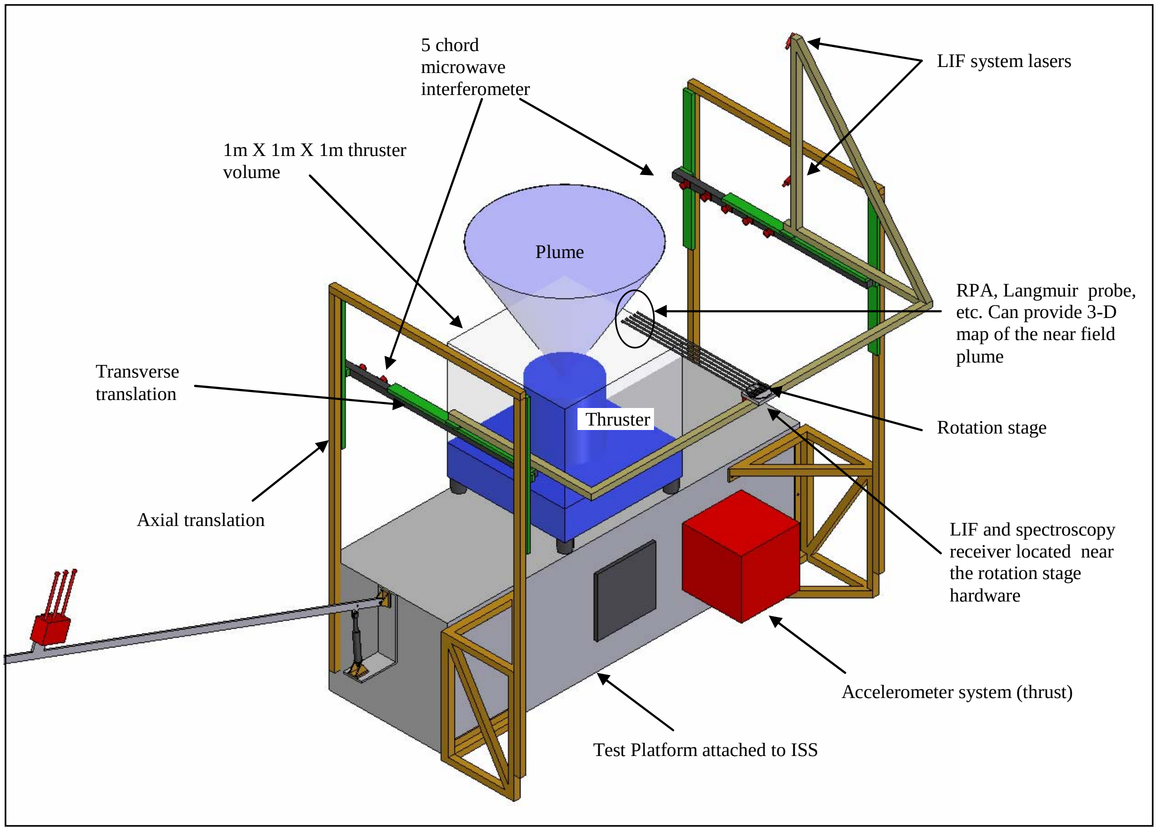

Figure 3. Conceptual layout of near field plume diagnostics

2. Far plume

The far field plume comprises all regions outside the near plume region. shows a conceptual layout of diagnostics in the far field plume. An eight meter boom is shown with several RPAs, LPs, and other intrusive diagnostics placed in groups along the boom. The boom will be anchored near the thruster on the test platform and swept through the plume. Other RPAs and LPs will be located on the space station truss for monitoring. The Space Station Remote Manipulator System (SSRMS) may also be used to place diagnostics in the far field plume of the thruster and is being evaluated for this purpose.

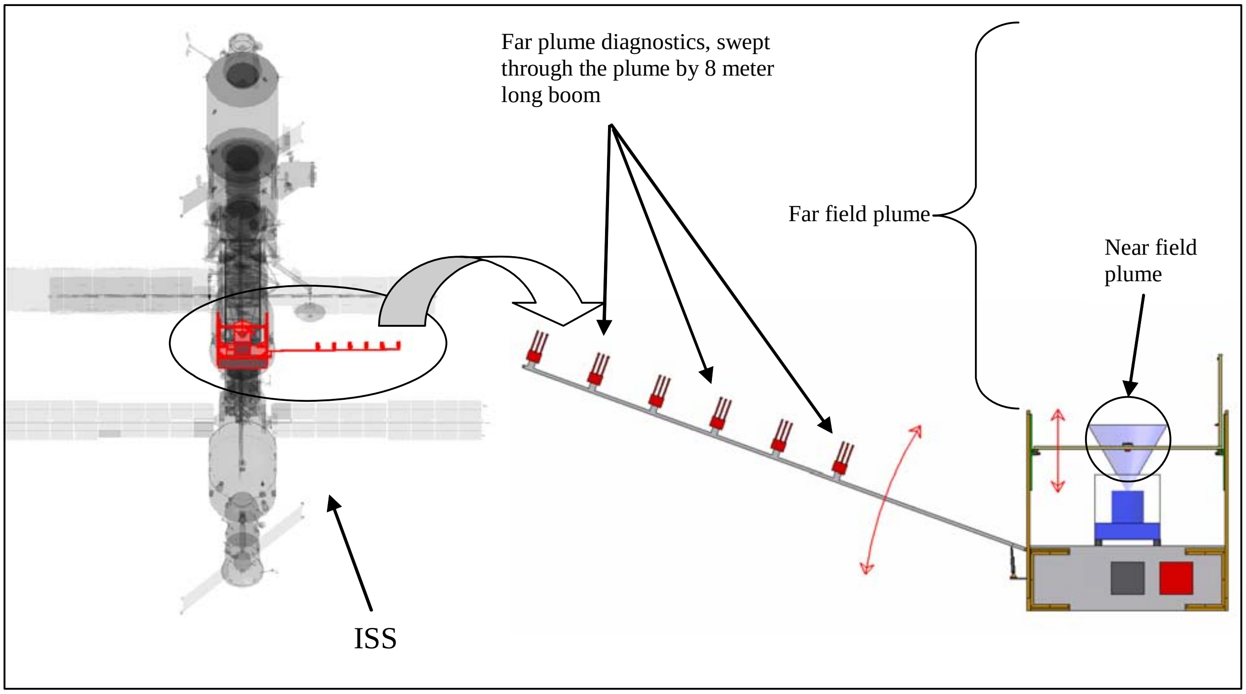

Figure 4. Conceptual sketch of far field plume diagnostics

V. Conclusion

The new vision for space exploration provides an opportunity to make the ISS more accessible. Using the ISS as it was intended, a space based laboratory for testing and research, will make it a more valuable asset by creating new capabilities for NASA and others. Rapid human exploration missions and the transport of cargo around the solar system will require advances in high power electric propulsion and hence require testing that can not currently be performed in ground-based laboratories. For all of these reasons, the proposed HPEP test platform for the ISS is an attractive prospect. The current design study will provide the information which would be required to begin development of an actual test platform for in-space use on the ISS. The diagnostics suite discussed in this paper is one of the essential components of the HPEP test platform, and its proper design will contribute substantially to the effective utilization of the platform

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