Performance and Plume Characterization of a Helicon Hall Thruster

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Abstract: The Helicon Hall Thruster (HHT) development program seeks to achieve high efficiency operation of a high-thrust, low-specific impulse (Isp) Hall thruster by employing the efficient ionization mechanism of a helicon source while maintaining the efficient acceleration characteristics which have made Hall thrusters an attractive propulsion option. In the HHT, ions are created in an annular helicon first stage and then accelerated in a closed-drift, Hall-effect second stage. Helicon waves, which are cylindrically bounded whistler waves, are considered to be one of the most efficient methods of producing a high-density, low-temperature plasma. A decrease in the power required for propellant ionization will directly translate into an increase in thruster efficiency at non-nominal Hall thruster operating conditions. This paper will present the work done to date on the HHT performance evaluation for input power from 0.6 to 8 kW with discharge voltages ranging from 100 to 600 V and discharge currents ranging from 5 to 25 A. Results of the performance characterizations suggest that there is a possible advantage of a Hall thruster with RF ionization stage at increasing RF-power.

Nomenclature

AFRL	=	Air Force Research Laboratory
DoD	=	United States Department of Defense
EDA	=	ElectroDynamic Applications
EP	=	Electric Propulsion
GEO	=	Geosynchronous Earth Orbit
HHT	=	Helicon Hall Thruster
Isp	=	Specific Impulse
LEO	=	Low Earth Orbit
OTVs	=	Orbital Transfer Vehicles
SBIR	=	Small Business Innovation Research
SOA	=	State Of the Art
T/P	=	Thrust-to-Power
RF	=	Radio Frequency
mN	=	millinewtons

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I. Introduction

Electric Propulsion (EP) systems have found widespread use on modern spacecraft with more than 240 satellites in orbit using some form of EP (including resistojets, arcjets, pulsed plasma thrusters, ion engines, and Hall thrusters) as of 2008[4]. Mission studies performed by the Air Force have highlighted the ability of EP systems to greatly improve the capabilities of US Department of Defense (DoD) space assets by increasing the payload delivered to orbit or extending the life of a given spacecraft [1-3]. Of the multiple types of EP considered by the Air Force, the Hall thruster was found to provide the greatest benefit to the widest variety of DoD missions and was therefore given the highest technology rating of the available propulsion options [1]. A Hall thruster's combination of high specific impulse (compared to chemical systems), reasonably high efficiency, and high thrust density make it attractive for future missions such as LEO-to-GEO orbital transfer vehicles (OTVs), space tugs, and servicing/re-supply vehicles for large orbital assets [2, 4-6].

Recent Hall thruster research activities into high-T/P operation have demonstrated less than optimal high-T/P capabilities and are still in progress. As part of the Air Force's efforts to advance the state-of-the-art (SOA) in Hall thruster capabilities, ElectroDynamic Application, Inc. (EDA) was awarded a Phase I and II Small Business Innovation Research (SBIR) to develop a Hall thruster which can meet the high-T/P goals. The concept behind EDA's two-stage Hall thruster is to combine the ionization efficiency of a helicon ionization source, which has recently become a topic of interest in the EP community, with the established acceleration benefits of a Hall thruster. Previous two-stage Hall thruster research has primarily focused on the use of a two-stage thruster for high-Isp operation [7-20]. The new approach being pursued by EDA is to improve the operation of a Hall thruster at low-Isp (1000-1300 seconds), where a Hall thruster operate effectively if the propellant utilization efficiency of the thruster is maintained [6]. The incorporation of a helicon ionization source should help mitigate the propellant utilization efficiency losses from operating a Hall thruster at low-Isp.

The primary goal of the HHT project is to develop a Hall thruster with an augmented ionization source to improve thruster operation in a high-T/P operating regime. To understand the limits of current Hall thruster technologies, a brief discussion of the limitations of SOA Hall thruster capabilities is presented. This report will describe the some of the design principles of the HHT and the results of operating the HHT from input power of 0.6 to 8 kW with discharge voltages ranging from 100 to 600 V and discharge currents ranging from 5 to 25 A and RF powers ranging from 300 to 800 W. Finally this paper will summarize the results of testing the HHT in both single- and two-stage RF operation.

A. Limitations of a SOA Hall Thruster

To fully appreciate the benefits provided by the HHT, one must first consider the capabilities and limitations of the technology which the HHT seeks to improve upon. Hall thrusters are EP devices that are often favored over other forms of propulsion for near-Earth applications due to their ability to produce relatively high-Isp, ranging from 1500-5000 seconds, and moderate thrust levels, ranging 10-4000 millinewtons (mN), depending on thruster size and operating condition at thruster efficiencies ranging from 50-70%. A Hall thruster is defined as an electrostatic EP device and consists of an annular discharge channel with an upstream anode/propellant distributor, a downstream cathode, and a magnetic circuit with either electromagnetic coils or permanent magnets (Figure 1).

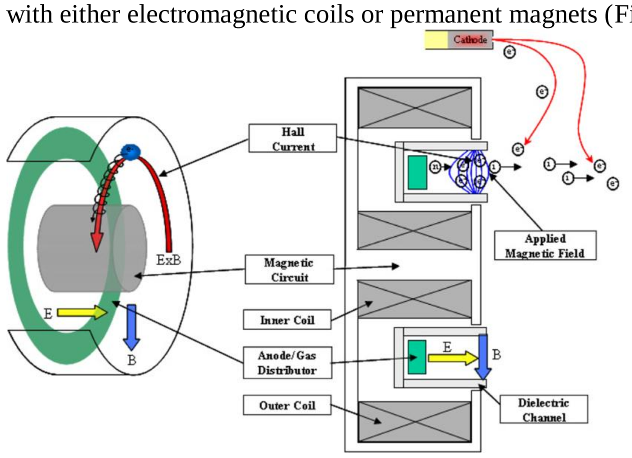

Figure 1, A sketch of a typical Hall thruster.

[4] Information obtained from Gunter's Space Page (space.skyrocket.de).

An important figure of merit used to characterize the performance of any EP device is its total electrical efficiency, which can be expressed as Equation 1 where η represents the device efficiency, P_{thrust}, represents the useful output thrust power, and P_{input}, depicts the input power supplied to the thruster.

$$\eta = \frac{P_{thrust}}{P_{input}} \tag{1}$$

For this discussion let us assume the input power supplied to a thruster can be divided into three parts as shown in Equation 2.

$$P_{input} = P_{thrust} + P_{ionization} + P_{lost} + P_{other} \approx P_{thrust} + P_{ionization} + P_{lost} \tag{2}$$

The powers listed in Equation 2 are defined as follows: P_{thrust} is the useful output thrust power, $P_{ionization}$ is the power that goes into ionizing propellant atoms, P_{lost} is the power lost to thruster inefficiencies, and P_{other} is power supplied to ancillary components of the device such as electromagnets, heaters, etc.. For modern electric propulsion devices, P_{other}, is generally small compared to P_{thrust}, $P_{ionization}$, and P_{lost} so the term will be dropped for this discussion[5]. We will also assume that P_{lost}/ P_{thrust} ratio will remain constant or behave the same as has been observed with other SOA and advanced SOA Hall thrusters. The total efficiency of an electric propulsion device can then be expressed as Equation 3.

$$\eta \approx \frac{P_{thrust}}{P_{thrust} + P_{ionization} + P_{lost}} = \frac{1}{1 + \frac{P_{ionization}}{P_{thrust}} + \frac{P_{lost}}{P_{thrust}}} \approx \frac{1}{Const. + \frac{P_{ionization}}{P_{thrust}}} \tag{3}$$

Equation 3 shows clearly that the efficiency of a device is maximized when the power required for ionization is minimized. In typical single-stage Hall thrusters, the ionization process is coupled to the thrust-producing ion acceleration process. The result of this coupling is an inability to optimize both processes independently. Several researchers have attempted to overcome this limitation by developing so-called two-stage Hall thrusters [7-20]. The main premise behind these two-stage thrusters is to minimize the ionization cost by introducing a secondary ionization source. It is important to note that these previous efforts focused on two-stage configurations for high-Isp operation. Most of the previous two-stage thruster designs relied on DC electron bombardment to provide additional ionization of the propellant. The reliance of the conventional Hall thruster on DC electron bombardment is an important point to note because it limits the efficiency of the device at all operating conditions due to the power lost in ionizing the propellant. The typical ionization cost in a DC discharge is approximately a factor of 5-10 times higher than the theoretical minimum of 12.1 eV for xenon and typically requires more than 60-120 eV/ion of input energy to ionize a single xenon atom in a single-stage SOA Hall thruster [21-22]. This means that a large portion of the incident energy intended for ionization, which does not go into three-body ionization processes, is rejected as waste heat and/or light.

In addition to limiting the Hall thruster's ultimate efficiency at all operating conditions, the reliance on DC electron bombardment also limits the range of specific impulses over which reasonable efficient operation can be achieved. At high specific impulses, many Hall thrusters show a decrease in efficiency due to an increase in the electron leakage current and the creation of multiply-charged ions [18, 23]. This limitation has been largely mitigated by advanced magnetic field topologies that prevent significant increases in Hall thruster inefficiencies at high-Isp operating conditions. The limitation encountered at low-Isp and high-T/P operating conditions are more fundamental to the basic operation of a Hall thruster and are unlikely to be fully mitigated by improved magnetic field topology changes alone. Further improvements in thruster efficiency at low-Isp and high-T/P operation may be possible with an augmented ionization mechanism that is independent of accelerating voltage, such as a helicon/RF ionization source.

In a SOA Hall thrusters, ionization occurs when a neutral propellant atom is struck by an electron traveling with a kinetic energy in excess of the first ionization potential of the propellant atom or a succession of low energy electrons striking a propellant atom before the atom releases its excited energy as a photons (three-body ionization) [24]. While the factors determining the maximum electron temperature in a typical Hall thruster are complicated, this value can be approximated to be roughly 10% of the applied discharge voltage for most modern thrusters [25]. To achieve high-T/P at constant thruster power, a Hall thruster must operate at higher discharge currents which

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[5] P_{other} and P_{lost} could easily be retained in the following discussion, but doing so does not affect any of the conclusions or statements made below.

imply higher propellant mass flow rates. For the increased discharge currents at a fixed thruster size, the increased collision rate between electrons and neutrals due to increased pressure suggests that the electron temperature will be depressed further beyond the reduction associated with low-voltage operation. It follows that at the low discharge voltages required to provide operation at low-Isp, only a small fraction of the electron population will have sufficient energy to result in propellant ionization. This decreasing fraction of energetic electrons leads to an increase in the effective ionization cost and contributes to the dramatic decrease in overall efficiency exhibited by typical Hall thrusters at low-Isp.

Although Hall thrusters are an excellent solution for many missions, the discussion above shows that there are limitations to Hall thruster operational regimes, in particularly at the high-T/P operating conditions which are of increasing importance to future missions. It is important to note that while advances in Hall thruster technology may partially mitigate these shortcomings, the existence of the performance-limiting mechanisms are a fundamental result of the underlying Hall thruster physics and will most likely not be eliminated by straightforward design evolution. The factors leading to performance limitations in a conventional thruster can be summarized by the following three statements:

1. The DC electron bombardment employed in conventional Hall thrusters is inefficient and requires an input energy that is a factor of 5-10 times higher than the ionization potential of the propellant gas;
2. The direct relationship between discharge voltage and electron temperature in the Hall thruster leads to a dramatic increase in ionization cost and a resultant decrease in efficiency for low-Isp and high-T/P operating conditions; and
3. The reliance of the Hall thruster ionization mechanism on back-streaming electrons from a hollow cathode leads to an upper boundary on the thruster current utilization efficiency and therefore a maximum in the achievable thruster efficiency.

B. Helicon Hall Thruster Concept

The Helicon Hall Thruster seeks to achieve high efficiency operation by employing the efficient ionization mechanism of a helicon source while maintaining the efficient acceleration characteristics that make Hall thrusters an attractive propulsion option. In the HHT, ions will be created in an annular helicon first stage and then accelerated in a closed-drift, Hall-effect second stage. Helicon waves, which are cylindrically bounded whistler waves [26], are widely regarded as the most efficient method of producing a high-density, low-temperature plasma [22, 27]. For example, helicon sources have been reported to produce approximately an order of magnitude more plasma than a DC electron bombardment source for a given input power [22]. Recent data on various helicon sources from numerous laboratories have measured ionization costs of 60-90 eV/ion on argon [28-31], which still indicates the potential for a helicon source to significantly improve efficiency, assuming that other losses neglected above for clarity are the same. In addition, it should be noted that all other things being equal, the same discharge voltage with argon will still yield a lower thrust-to-power ratio due to its lighter atomic mass. In order to increase the maximum thrust-to-power, the reduction in ionization cost must also be demonstrated on a heavy propellant such as xenon. However, given the larger weight and size of the xenon atom relative to argon, it should have an electron-neutral ionization cross section larger than argon for the same electron population in addition to having a greater residence time in the helicon ionization region than argon at the same volumetric flow rate leading to a larger probability of an electron-neutral ionization collision. This points to a xenon fueled helicon source having a lower ionization cost then it would for argon. However, the ionization cost trend with Hall thruster discharge voltage points to ionization cost being proportional to discharge voltage [32]. Reference [32] presents ionization cost values for a high performing Hall thruster of 70 eV/ion at 105 V and increasing with voltage. At low voltages, the helicon ionization cost would need to be at the lower end of the range reported in the literature to provide some benefit where as at the higher voltage (higher specific impulse) end it may end up providing the largest impact on thruster performance.

An additional advantage of the HHT compared to conventional EP devices is its ability to efficiently operate on propellants other than xenon. Argon for example, is much less costly than xenon but is generally a poor choice as a propellant due to its relatively low atomic mass and high ionization potential which tend to make the ionization cost prohibitive for efficient thruster operation. The helicon source, however, has exhibited an affinity for operation on argon with ionization costs approaching the theoretical minimum. In addition to the noble gases, the HHT combined with a helicon cathode may be operable on additional propellants, including oxidizing materials, which would constitute poisons for conventional thermionic cathodes. Examples of such propellants are hydrazine and other non-propellants which, if found to be practical, would allow the HHT and chemical thrusters to share a common propellant.

As discussed in the previous section, significant decreases in the power required for propellant ionization in the Hall acceleration region will translate directly into a concomitant increase in thruster efficiency at off nominal operating conditions (low-Isp and high-T/P). The primary goal of the HHT project is to develop a Hall thruster with a separated and/or augmented ionization process to improve thruster operation in a low-Isp and high-T/P operating regime. Additionally, several design characteristics are incorporated in the HHT design that should improve single and two-stage operation at the low-Isp and high-T/P operating conditions. A simple schematic of the concept HHT discharge channel is shown in Figure 2.

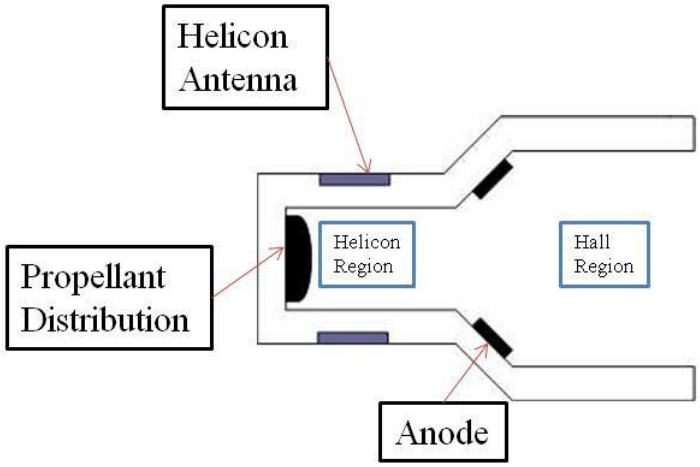

Figure 2, Schematic of the HHT discharge channel.

The ionization augmentation will be accomplished by having a discharge channel separated into two main regions, as shown in Figure 2. The upstream region will be the location of the helicon ionization source with an M=0 antenna configuration. Selection of helicon antenna configuration for the HHT has been discussed in greater detail by Beal et al. in Ref. [33]. The channel width was determined from annular helicon source theory with a purely axially applied magnetic field. The downstream portion of the channel will contain a Hall current ionization and acceleration region similar to what one would expect from a SOA Hall thruster.

The HHT discharge channel will contain additional features to improve operation at the high-T/P regime. The primary feature will be the incorporation of a small channel diameter-to-width ratio for the Hall region of the thruster. This feature has been shown to improve Hall thruster operation on both alternate propellants, such as krypton, and xenon at low and high power levels (from sub-kW to 100 kW) [6, 34-38]. Both the helicon and the modified Hall stage features should increase the likelihood of neutral particle ionization, compared to a SOA Hall thruster, thus benefiting the goal of low-Isp and high-T/P operation.

II. HHT Operation

C. Single Stage Operation

The HHT was installed in the University of Michigan Plasmadynamic and Electric Propulsion Laboratory (PEPL) Large Vacuum Test Facility (LVTF)[6] and testing of the thruster was begun in single stage configuration. Starting the HHT in the single-stage configuration was to bake out the thruster, burn-in the new discharge channel, and map out the single stage performance characteristics. Understanding the operational characteristics of the HHT in single-stage would help evaluate the two-stage helicon operation. A photograph of the initial operation of the HHT is shown in Figure 3.

[6] http://pepl.engin.umich.edu/

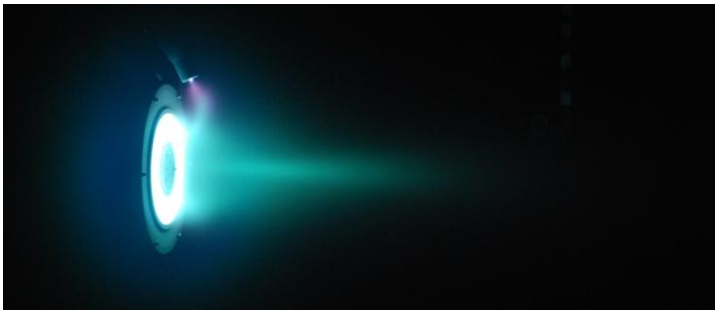

Figure 3, Single-stage operation of the HHT.

The HHT was operated in single-stage configuration at power levels ranging from 700 W to 8 kW of discharge power. Several plasma plume diagnostics were employed alongside the thrust stand measurements to map out the performance of the HHT in single-stage.

D. Two Stage Operation (RF Stage)

Before a long and detailed performance and plume characterization of the HHT was to be undertaken it was decided that the RF-stage of the HHT was to be started in case any modifications were needed in the RF antenna setup. It was found that due to the low pressures of the LVTF (mid 10^{-6} torr), when propellant was flowing through the RF-stage, that the hollow cathode of the HHT needed to be emitting electrons to initiate the RF discharge. A photograph of the HHT operating in RF mode is shown in Figure 4.

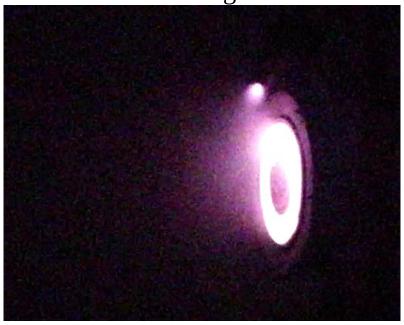

Figure 4 Initial operation of the HHT RF-stage with hollow cathode.

It was later discovered that once the RF-stage was lit the hollow cathode could be shut off without any issues in the operation of the RF-stage (Figure 5).

Figure 5, HHT RF-stage operating without the hollow cathode at a RF power of approximately 500 W.

III. Results

E. Single Stage Performance

The initially performance characterization of the HHT was performed with only the Hall-stage electromagnetic coils. It was determined that the RF-stage electromagnetic coils would provide additional modifications to the applied magnetic field topology of the HHT and therefore it was decided that the single-stage characterization of the thruster would be completed with the RF-stage coils engaged (Figure 6). The remainder of the single-stage performance data was taken for discharge voltages ranging from 100 to 400 V and propellant mass flow rates ranging from 5 to 22.5 mg/s. The remainder of the performance data shown in this report was taken with the RF-stage coils engaged at predetermined settings.

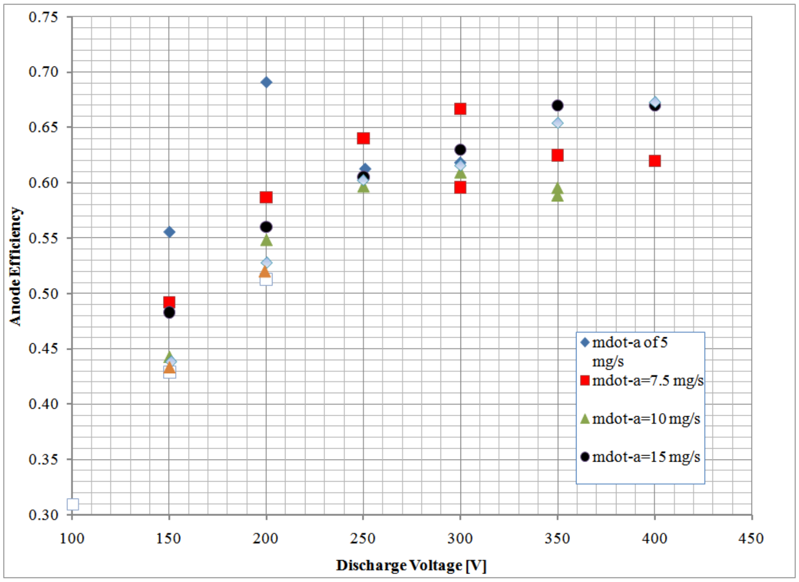

Figure 6, Initial performance results of the HHT with both the Hall-stage and RF-stage coils.

The anode efficiency of the HHT as a function of the anode Isp is shown in Figure 7. The anode efficiency and Isp includes only the anode propellant mass flow and discharge power (discharge current and voltage) and does not include the power and propellant usage of the cathode and electromagnetic coils. It is well known that once a Hall thruster is past its initial development stages (beyond laboratory/prototype model), careful attention will be placed on matching both the magnetic circuit and cathode requirements to the desired goals of a propulsion system. The HHT thruster presented in this report can be considered a laboratory/prototype model thruster and thus considering the cathode and magnetic circuit power and propellant requirements is not appropriate. The single-stage performance evaluation of the HHT showed that the thruster was operating as expected based on SOA and advanced SOA Hall thrusters published data.

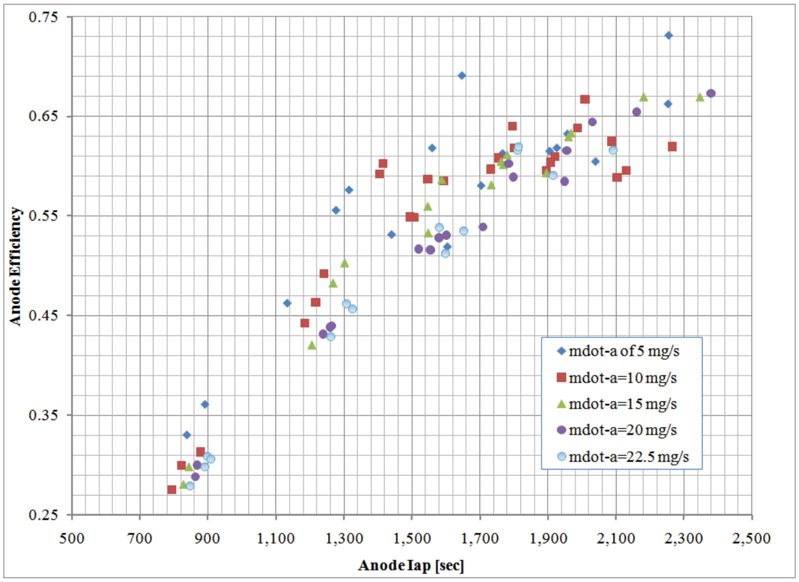

Figure 7, HHT single-stage anode efficiency as a function of anode Isp.

The ratio of the measured thrust to the discharge power (T/P) of the HHT in single-stage configuration is shown in Figure 8. The maximum T/P observed during testing of the HHT in single-stage was approximately 90 mN/kW at a discharge voltage of 150 V. The next higher propellant mass flow rate of 10 mg/s had T/P results between 80 and 90 mN/kW for discharge voltages ranging from 100 to 200 V.

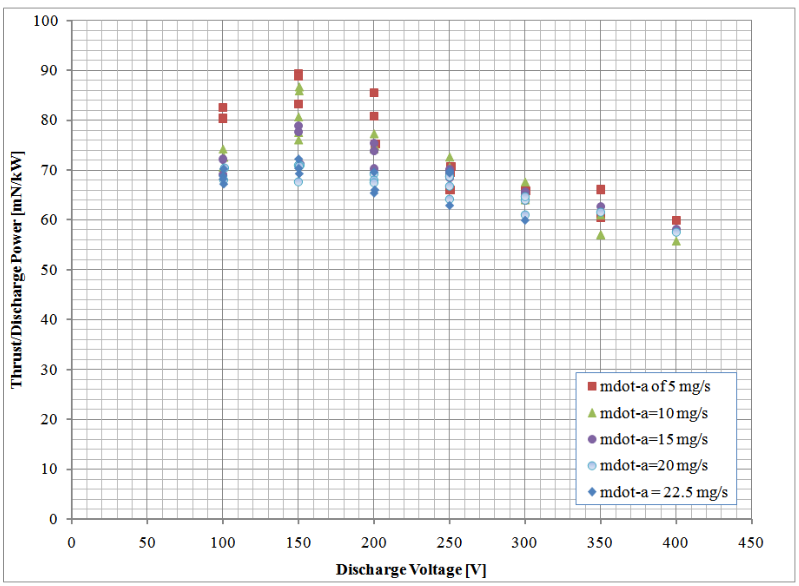

Figure 8, HHT single-stage ratio of thrust to discharge power as a function of discharge voltage.

F. Two Stage Operation Performance (RF Stage)

The two-stage performance characterization of the HHT was performed from discharge voltages ranging from 100 to 300 volts on the Hall-stage with propellant mass flow rates ranging from 10 to 22.5 mg/s and RF-stage powers ranging from 300 to 800 W. The lower propellant mass flow rate of 5 mg/s was not investigated because of the inability to achieve an RF discharge, even with employing the external hollow cathode to aid in startup. Limiting the performance characterization of the two-stage operation to a discharge voltage of 300 V was done to expedite the testing of the HHT due to a limited window of availability of LVTF.

Since the two-stage operation of the HHT includes the RF-stage with its associated components, the anode efficiency, anode Isp, and discharge power will be redefined to include the power of the RF-stage in addition to the

Hall-stage discharge power and propellant mass flow rate through the anode. The new definitions for the two-stage HHT operation will be thruster efficiency, thruster Isp, and thruster power. The HHT two-stage performance for RF power ranging from 300 to 800 W is illustrated in Figure 9 and Figure 10.

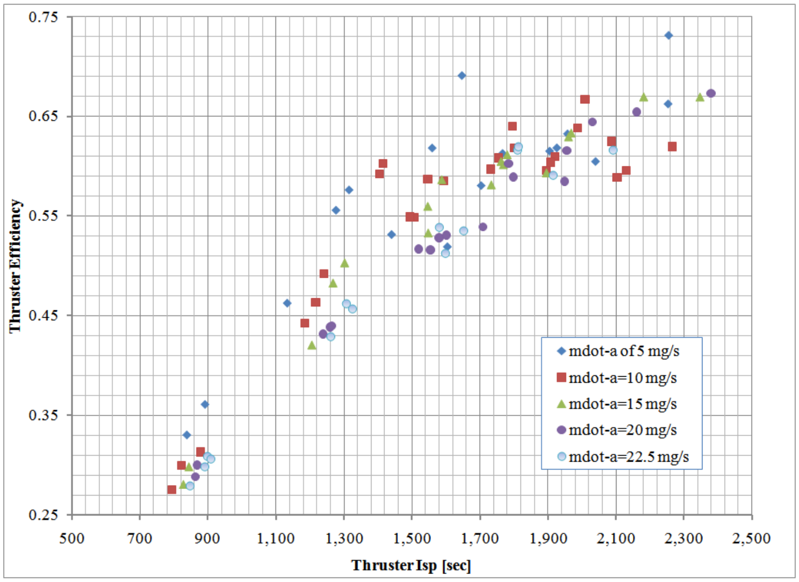

Figure 9, HHT two-stage thruster efficiency as a function of thruster Isp for RF power ranging from 300 to 800 W.

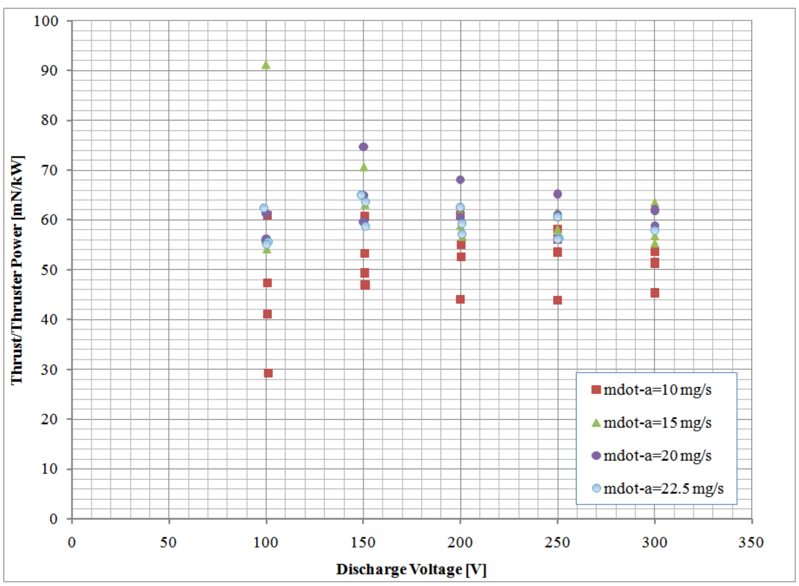

Figure 10, HHT two-stage thruster T/P as a function of discharge voltage for RF power ranging from 300 to 800 W.

The performance of the HHT in two-stage configuration is on par or lower than compared to the HHT in single-stage operation (Figure 11 - Figure 13 and Figure 15) with one exception, 800 W RF power and 20 mg/s (Figure 14). The raw data for the 800 W RF power and a propellant mass flow rate of 20 mg/s appears to be legitimate, however we could not completely rule out RF interference with the thrust stand. It is expected that if the RF-stage was improving the performance of the HHT, the results of the 22.5 mg/s data and 800 W RF power would have shown a similar trend as other data points. Only continued testing of the HHT at the same RF power levels and greater, with the addition of tangential thrust stand measurements (one way to confirm the RF power is not interfering with the performance measurements), would validate this improved performance with 800 W RF power.

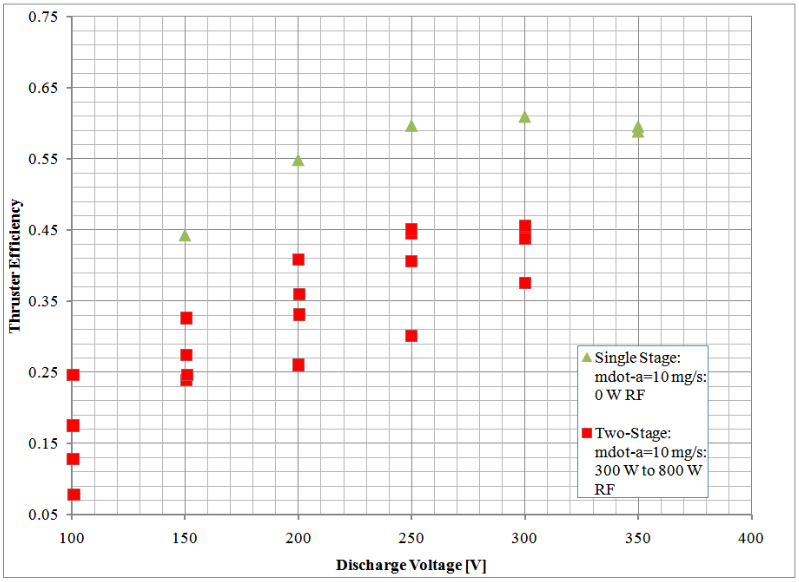

Figure 11, Comparison of the HHT anode/thruster efficiency as a function of discharge voltage for RF-stage power ranging from 300 to 800 W at 10 mg/s propellant mass flow rate.

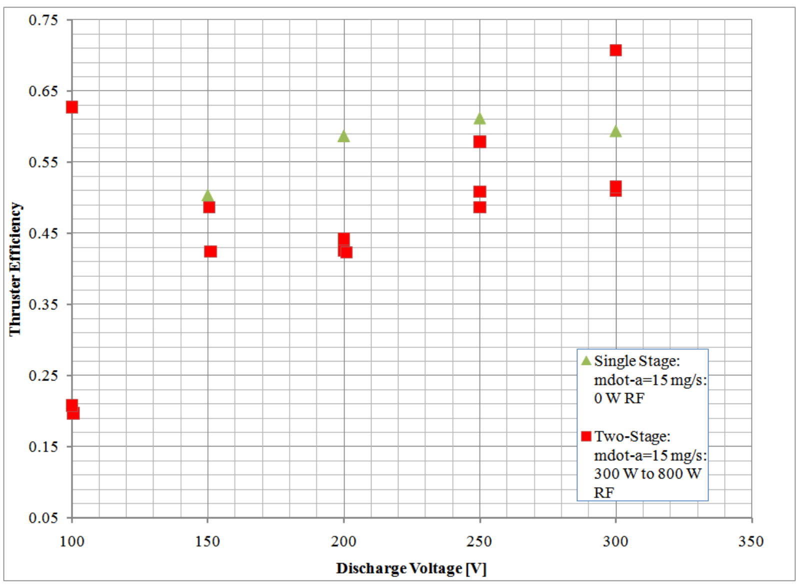

Figure 12, Comparison of the HHT anode/thruster efficiency as a function of discharge voltage for RF-stage power ranging from 300 to 800 W at 15 mg/s propellant mass flow rate.

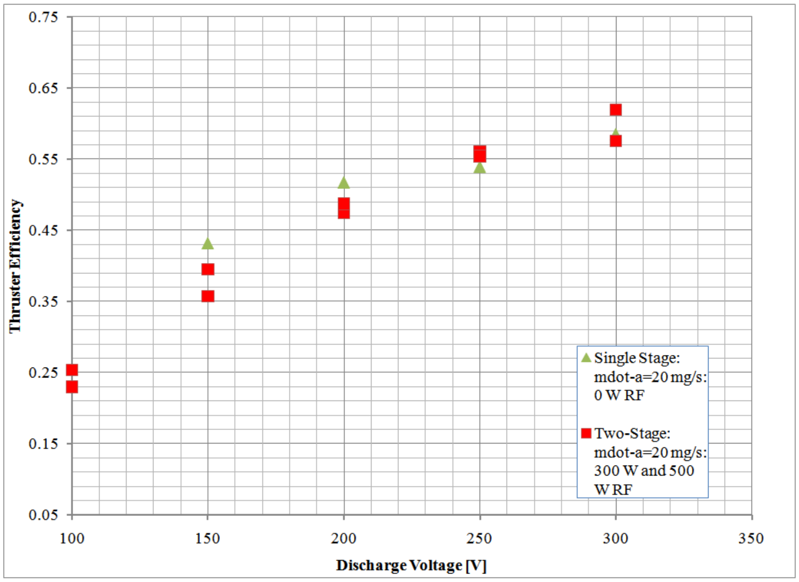

Figure 13, Comparison of the HHT anode/thruster efficiency as a function of discharge voltage for RF-stage power of 300 W and 500 W at 20 mg/s propellant mass flow rate.

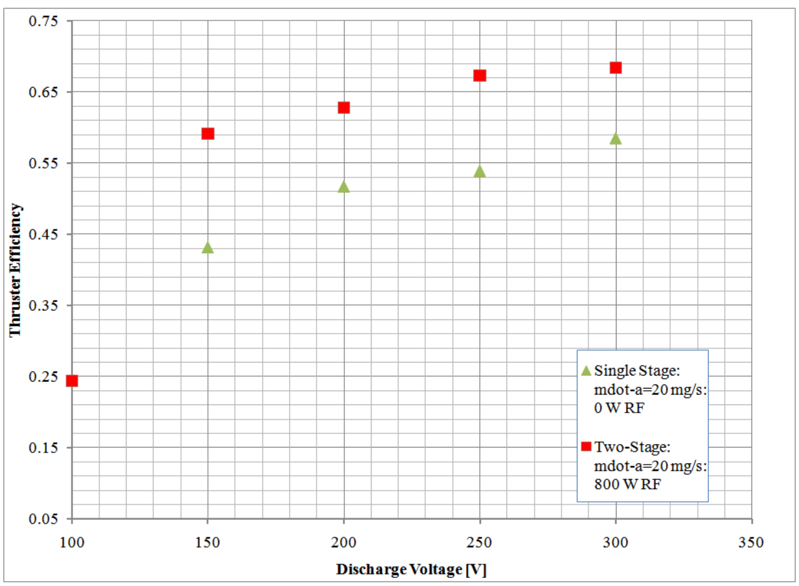

Figure 14, Comparison of the HHT anode/thruster efficiency as a function of discharge voltage for RF-stage power of 800 W at 20 mg/s propellant mass flow rate.

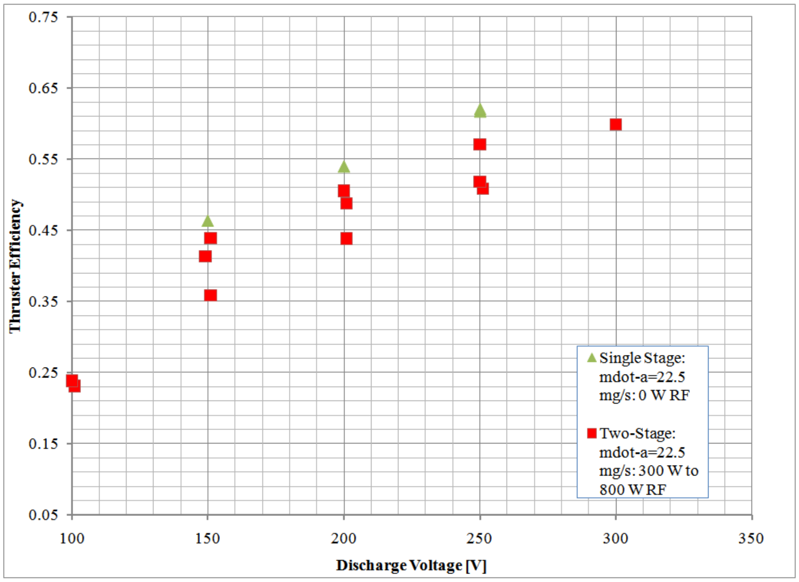

Figure 15, Comparison of the HHT anode/thruster efficiency as a function of discharge voltage for RF-stage power ranging from 300 to 800 W at 22.5 mg/s propellant mass flow rate.

An additional method of examining the performance results is to look at the thruster efficiency as a function of RF power at a give propellant mass flow rate and comparing these results with the data obtained in the single-stage configuration at similar thruster operating parameters. Figure 16 through Figure 19 is an attempt to show that as one increases the RF-stage power the efficiency decreases accordingly, except for the 800 W RF power cases. As discussed above, this 800 W RF power exception in improvement with the RF-stage could be a result of either RF interference in the thrust stand or the initial indication that at higher RF powers the performance of the thruster may improve over what was observed in single-stage operations. Previous attempts to develop a two stage Hall thruster have shown similar trends as observed in Figure 16 as the additional stage's power is increased [12, 15]. The goal of this program was to examine the operation of a two-stage RF device at operating regimes not fully explored in the previous two-stage Hall thruster studies.

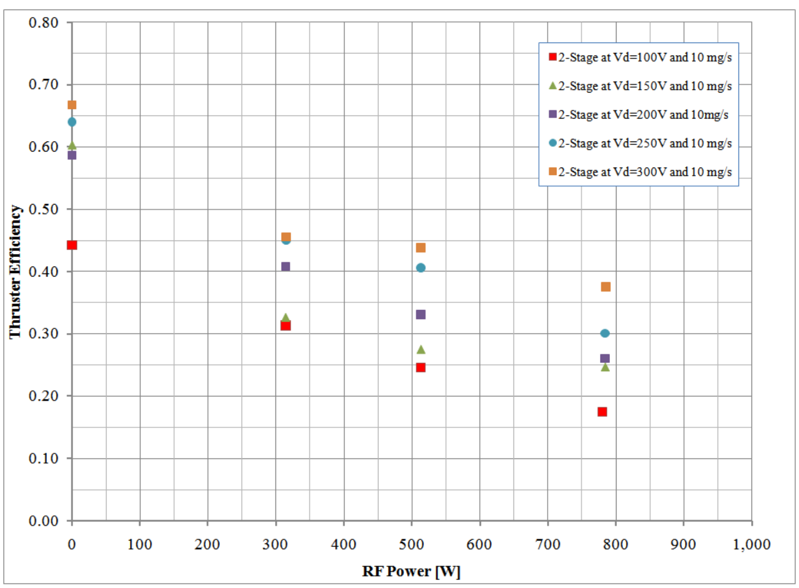

Figure 16, Comparison of HHT thruster efficiency as a function of RF power from 0 to 800 W and discharge voltage at a propellant mass flow rate of 10 mg/s.

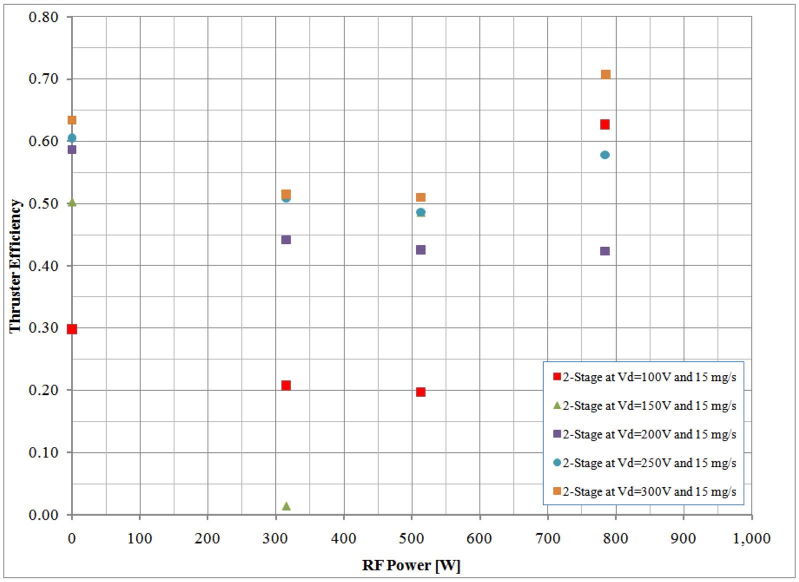

Figure 17, Comparison of HHT thruster efficiency as a function of RF power from 0 to 800 W and discharge voltage at a propellant mass flow rate of 15 mg/s.

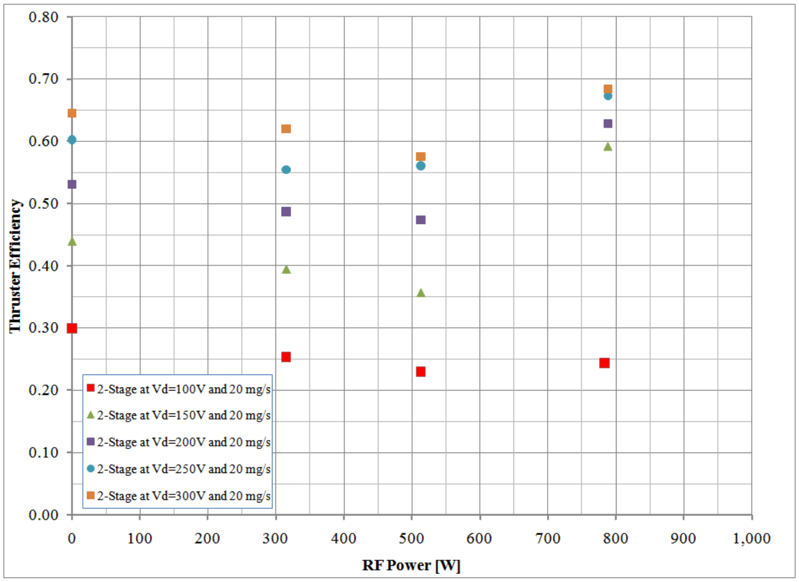

Figure 18, Comparison of HHT thruster efficiency as a function of RF power from 0 to 800 W and discharge voltage at a propellant mass flow rate of 20 mg/s.

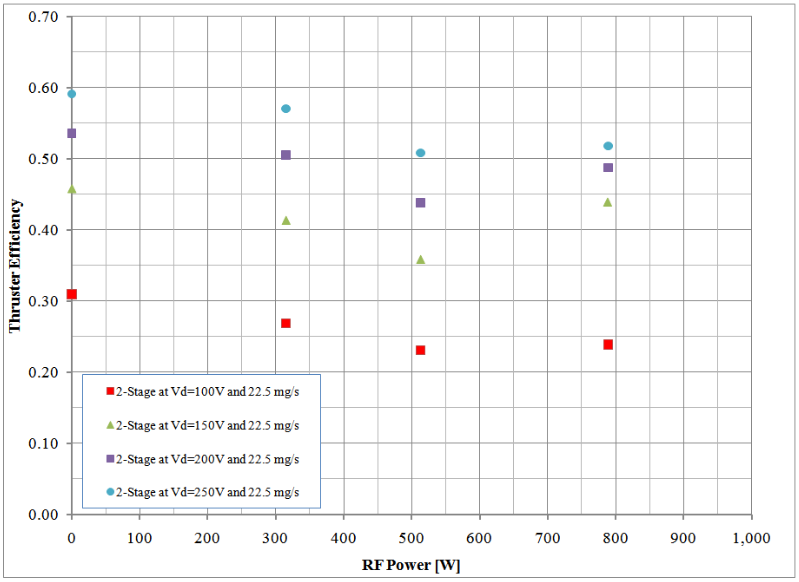

Figure 19, Comparison of HHT thruster efficiency as a function of RF power from 0 to 800 W and discharge voltage at a propellant mass flow rate of 22.5 mg/s.

IV. Plume Diagnostics

The plasma plume of the HHT was investigated with several different diagnostic devices available at PEPL. The plume of the HHT was mapped with a concentric Faraday probe similar to the probe used in reference [32], a **E**x**B** probe [39-40], a Langmuir probe [41], and a Retarding Potential Analyzer (RPA) [41]. It has been shown in previous studies [18, 25, 42] that several thruster utilization efficiencies can be gathered from the acquired plume data and performance results of the thruster. Examining the utilization efficiencies of the HHT will provide some valuable insight into the operation of a two-stage Hall thruster at low-Isp operating regimes. Please note that not all of the HHT operating conditions were fully mapped with the plume diagnostics due to various reasons such as thermal issues with the RPA when employed too often, interference of the Faraday probe swipes with thrust stand, and need to measure the plasma potential when the RPA was used.

G. Single Stage

The HHT plasma plume was investigated in the single-stage configuration for multiple throttle points. As mentioned above, not all of the diagnostic probes were used on every throttle point and only a few points had sufficient plume data to complete an efficiency analysis of the thruster. Two such points at the 10 mg/s propellant mass flow rate in single-stage configuration utilization efficiencies are detailed in

Table 1. Sample traces of the ExB, Faraday, and RPA data for the 200 V 10 mg/s single-stage case are shown in Figure 20 through Figure 22, respectively.

Table 1, Comparison of the plume properties and utilization efficiencies of the HHT in single-stage.

	V_d = 150 V = 10 mg/s	V_d = 200 V = 10 mg/s
I_d [A]	10.1	9.8
Ω_1	0.9595	0.9705
Ω_2	0.0405	0.0225
Ω_3	0.0	0.007
V_a [V]	122	173
I_{Beam} [A]	7.16	7.26
λ [°]	22	17
η_v	0.813	0.867
η_c	0.709	0.741
η_m	0.966	0.975
η_q	0.997	0.997
η_d	0.858	0.912
η_a Calculated	0.476	0.569
η_a Measured Thrust	0.49	0.55

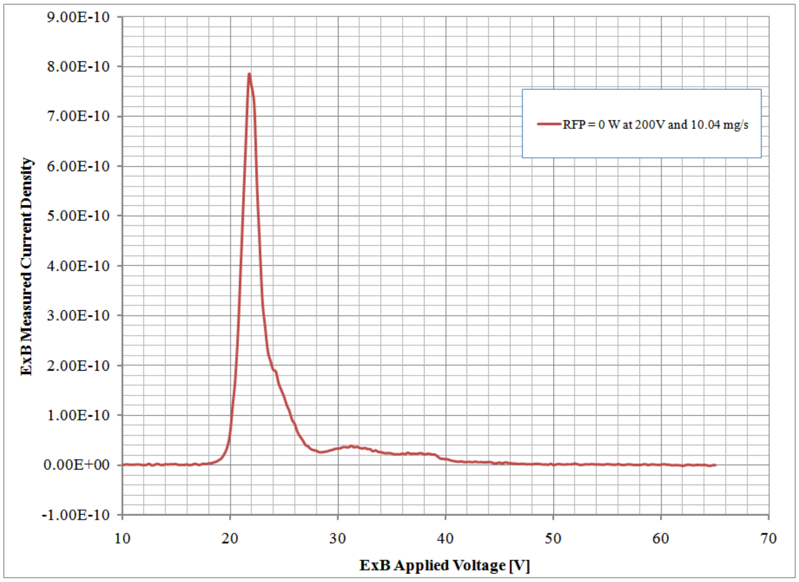

Figure 20, ExB trace for the 200 V 10.04 mg/s single-stage operation of the HHT.

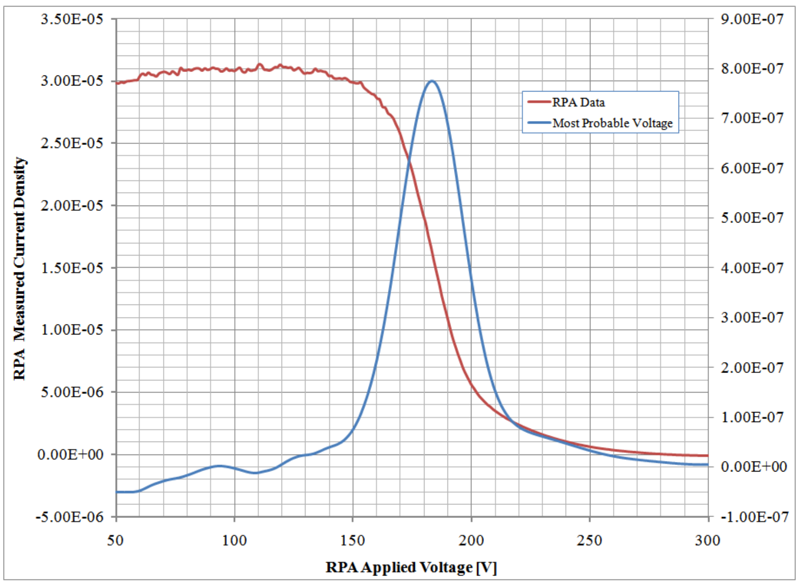

Figure 21, RPA data and most probable ion voltage for the 200 V 10.04 mg/s single-stage operation of the HHT.

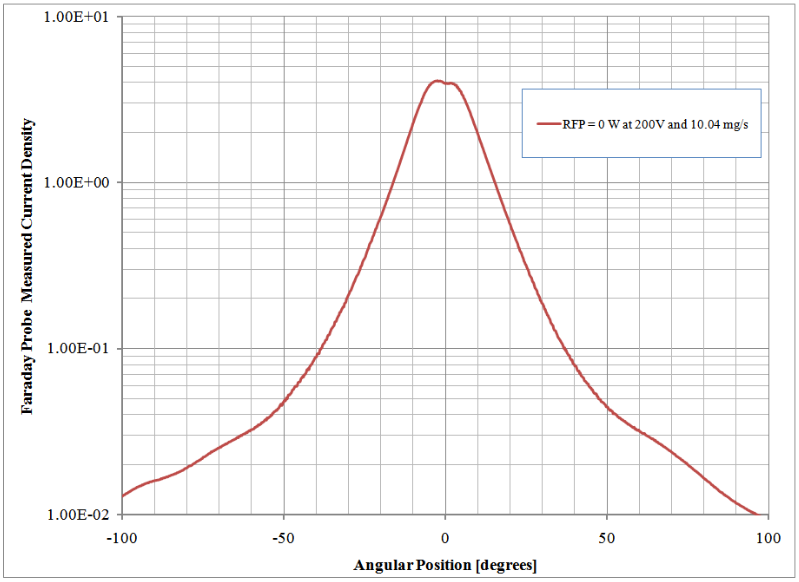

Figure 22, Faraday probe data for the 200 V 10.04 mg/s single-stage operation of the HHT.

H. Two Stage Operation (RF Stage)

For the two-stage HHT testing most of the plume data was taken during each throttle point or at similar operating conditions. A comparison of the thruster utilization efficiencies, plume data, and ion production cost for a single discharge voltage and propellant mass flow rate for both single- and two-stage HHT operation as a function of the RF-stage electromagnetic coil and power are shown in

Table **2** and Figure 23. The results shown in

Table **2** and Figure 23 do not show a performance gain with the operation of the RF-stage. However, the selection of the 100 V and 20 mg/s throttle point for comparing the influence of the RF-stage on the operation of the HHT was chosen during the testing of the thruster and before the results shown in Figure 23 were known. If future testing of HHT prove that the two-stage operation of the HHT above 800 W does improve the thruster at high T/P conditions then a similar analysis of RF-stage applied magnetic field and RF-power should be able to provide further insight into the characteristics of the HHT.

Table 2, Comparison of the plume properties, utilization efficiencies, and ion production costs of the HHT in single-stage and various two-stage RF-powers.

V_d = 100 V = 20 mg/s	RF Power 0 W	RF Power 300 W	RF Power 500 W	RF Power 800 W
I_d [A]	24.4	24.4	24.2	24.7
Ω_1	0.9512	0.9512	0.9512	0.9512
Ω_2	0.0488	0.0488	0.0488	0.0488
Ω_3	0.0	0.0	0.0	0.0
V_a [V]	80	80	80	80
I_{Beam} [A]	15.89	15.96	15.89	16.05
λ [°]	32	33	32	33
ε [eV/ion]	75.39	74.7	74.1	75.75
η_v	0.8	0.8	0.8	0.8
η_c	0.651	0.654	0.656	0.65
η_m	1.064	1.069	1.064	1.075
η_q	0.996	0.996	0.996	0.996
η_d	0.721	0.71	0.714	0.701
η_a Calculated	0.398	0.396	0.398	0.39

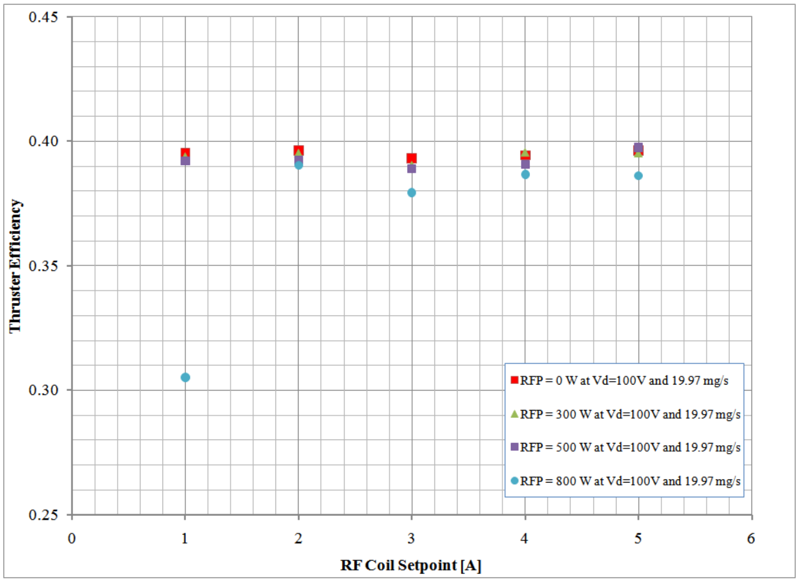

Figure 23, Comparison of the thruster efficiencies as a function of the RF-stage coil current for single- and two-stage operation.

V. Conclusions

I. HHT Summary

A helicon/RF based two-stage Hall thruster was developed and characterized during the course of HHT program. Results of the performance characterizations suggest that there is a possible advantage of a Hall thruster with RF ionization stage at increasing RF-power. However, conclusive results were not obtained during the Phase II effort due to possible RF interference with the thrust stand. The HHT performance characterization resulted in a maximum T/P of approximately 90 mN/kW at a discharge voltage of 150 V and low propellant mass flow rates (low natural propellant densities). It was thought that the small channel diameter-to-width ratio for the Hall region would help improve the low-Isp operation, but no indications were observed that support this concept. However, the small channel diameter-to-width ratio could still prove useful for harder to ionize and/or lighter propellants, as has been seen with other Hall thrusters. Another characteristic that was designed into the HHT was the ability to separate the propellant distribution from the anode of the system. Performance results of the HHT have shown no benefit of separating the anode and gas distribution.

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