Recent Development Activities in Hollow Cathode Technology*†

Michael J. Patterson and Matthew T. Domonkos
National Aeronautics and Space Administration
Glenn Research Center
MS 301-3
21000 Brookpark Road
Cleveland, Ohio 44135
216-977-7481
Michael.J.Patterson@grc.nasa.gov

Christian Carpenter and Scott D. Kovaleski
QSS Group, Inc.
Cleveland, Ohio

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A number of hollow cathodes have been recently developed at NASA Glenn Research Center for a variety of electric propulsion applications. These include 3.2 mm, 6.4 mm, and 12.7 mm diameter hollow cathodes for emission currents ranging from 100 mA to 100 Amperes. Engineering model assemblies have been fabricated for 10 A, 25 A, and 40 A applications, and flight units have been manufactured for 10 A and 25 A operation. Cathode components, as well as complete assemblies, have been delivered and flown for space missions, including the International Space Station, Deep Space One, and STEX. On-going activities at NASA GRC include development of engineering model low-current (<2 A) hollow cathodes, low-flow hollow cathodes, and high-current (40- and 100-A class) hollow cathodes. Tests to reduce environmental and interface requirements for long-life operation have also been conducted. Commercialization activities, to transfer cathode technology to aerospace and non-aerospace users, are also being pursued.

Introduction

With the success of the NASA Solar Electric Propulsion Technology Applications Readiness program ion propulsion system on the Deep Space One spacecraft,[1] the future for this propulsion technology for other NASA missions appears promising. For most of these ion propulsion applications hollow cathodes assemblies (HCAs) are required for discharge plasma generation and beam neutralization functions. Near-term applications for HCAs for both ion and Hall propulsion call for emission currents ranging from as low a 100 mA to 100 Amperes. Hollow cathode assemblies may also be used for spacecraft charge control, low-electron temperature space plasma simulation, and ground-based materials processing.

While considerable work has been conducted in the

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development of hollow cathodes for electric propulsion, some of which was supported by the International Space Station plasma contactor development activity,[2] additional efforts are required. These activities include improving efficiency, understanding life-limiting mechanisms and extending life, developing compact cathode assemblies for low-power electric propulsion, and developing high-current cathode designs for high-power engines.

NASA Glenn Research Center (GRC) is conducting a fairly comprehensive experimental and analytical program in cathode technology under the Space-Base ion propulsion program, including work in both hollow cathodes and propellant-less cathode technology. This paper discusses recent developments in hollow cathodes.

Electric Propulsion Applications

The following section describes various development activities associated with hollow cathodes for electric propulsion applications requiring emission currents in the range of about 0.1 to 100 A.

Low (< 2 Ampere) Current

Small HCAs (SHCAs) are appropriate for electron emission current requirements in the range of about one hundred milli-amperes to approximately 2 Amperes. They operate at xenon propellant flow rates in the range of 0.02-0.20 milligrams per second (approx. 0.2-2.0 sccm). Applications for SHCAs include the discharge and neutralizer cathode of the NASA 8 cm diameter 300 W ion engine,[3] the hollow cathode micro-thruster (HCMT),[4] as well as small low-power Hall thrusters. The hollow cathode is constructed of a 3.2 mm diameter refractory alloy tube.

Several prototype versions of this cathode have been manufactured and tested on xenon propellant. They include both keeper-less, and open-keeper electrode geometries, used in parametric investigations of cathode design, as well as for the discharge and neutralizer cathodes for the NASA 8 cm thruster, and for the HCMT. Data for an intermediate-size design are shown in Figures 1 and 2.

In Figure 1, the minimum xenon flow rate to maintain stable spot-mode operation (as indicated by <200 mV peak-to-peak AC noise measured on the keeper voltage) is plotted as a function of cathode emission current. The data were obtained with the SHCA operating in a 'diode-configuration'; all emission current was collected to the open-keeper. As noted, the minimum flow rate at about 0.48 A was about 0.084 mg/sec, and decayed to about 0.031 mg/sec at 1.53 Amperes.

Cathode tip temperature data were obtained for a range of emission currents (I_e) for this geometry, and these data are shown in Figure 2. The tip temperature, in degrees Celsius, for this assembly can be approximated by the equation:

$$T_{tip} = 405 \cdot \ln(I_e) + 1165 \qquad [1]$$

As indicated, at 0.48 A, the temperature was about 835 degrees C. At 1.25 A emission current, the tip temperature increased to about 1250 degrees C. These two extremes represent the typical maximum range of operating temperatures, yielding a rather modest throttling range of about 2.6 in emission current. Below about 825-835 degrees C, hollow cathodes tend to run unstably and extinguish. At about 1250 degrees C, the emitter lifetime drops dramatically due to changes in the emitter chemistry.

Low-power neutralizer

Figure 3 shows the keeper input power plotted against beam current for a prototype model SHCA used as a neutralizer for the NASA 8 cm engine. The geometry employed an open-keeper configuration, and was of smaller orifice diameter than the cathode depicted in the data of Figures 1 and 2. As noted, as the beam current increased, the keeper power consumed by the SHCA decreased, due to both the lower keeper voltage associated with the higher plasma density and the reduced keeper current necessary to sustain operation of the cathode. The keeper input power (in watts) for this SHCA can be approximated by:

$$P_{keeper} = 11.3 + 32.6 \cdot I_b \qquad [2]$$

where I_b is the beam current in amperes. At 0.08 A beam current, the input power was about 8.7 W, and it dropped to about 4.8 W at 0.20 A. The propellant flow rate at these conditions was about 0.06 mg/sec of xenon.

A design for a compact engineering model version of the SHCA has been developed, employing an enclosed-keeper configuration. The estimated mass is ≤ 100 grams, with dimensions ≤ 8.3 cm length x 2.0 cm diameter maximum. Technology challenges for SHCAs include demonstration of long-life and reliable heater-less ignition.

Moderate (2–40 Ampere) Current

Medium HCAs (MHCAs) are appropriate for electron emission currents in the range of about 2 amperes to approximately 40 amperes. They operate at xenon propellant flow rates in the range of 0.20-2.00 milligrams per second (approximately 2.0-20 sccm). Applications for MHCAs include discharge and neutralizer cathodes for kilowatt-class ion engines and Hall thrusters. These assemblies are normally constructed using hollow cathodes of 6.4-to-12.7 mm diameter.

Deep Space One Engine Cathodes

Recently developed designs include the discharge and neutralizer cathodes for the Deep Space One engine. Both cathode designs for the flight engine were developed at NASA GRC, and, with the exception of the heaters (manufactured at NASA GRC), were built by the flight contractor Boeing Electron Dynamics Devices. The cathodes have demonstrated more than 13,000 hours operation in space, and over 18,000 hours during testing of the flight spare engine at the Jet Propulsion Laboratory. The nominal range of emission currents for the discharge and neutralizer cathodes are approximately 4-15 amperes, and 2.5-3.3 amperes respectively.

Low-flow neutralizer

A higher-efficiency neutralizer was subsequently developed at NASA GRC for application to the Deep Space One engine. Prototype and engineering model assemblies of this new design were fabricated and performance characterized in both a stand-alone configuration, and integrated with an ion thruster. Figure 4 shows the keeper input power versus flow rate for the flight engine neutralizer (DS1), and for prototype (PM) and engineering model (EM) versions of the higher-efficiency neutralizer. The data were obtained from diode operation; i.e., no beam current. As indicated, the PM and EM neutralizers operate at input power levels 50-100% lower than those of the DS1 neutralizer when operated in a diode mode.

The EM neutralizer was subsequently integrated onto an EM version of the Deep Space One engine, and the flow rate for the neutralizer, in mg/sec, can be expressed by the equation:

$$\dot{m}_n = 0.070 + 0.102 \cdot I_b \qquad [3]$$

for beam currents in the range of 0.48 A to 1.76 A. Over the throttling range of the Deep Space One engine, the new EM neutralizer affords a considerable reduction in xenon propellant consumption as compared to the flight unit. The new neutralizer operates at approximately 70% of the flow rate required of the flight neutralizer at the engine full power condition of 2.3 kW, and at approximately 50% of the flow rate required of the flight neutralizer at the low power condition of 0.49 kW.

The keeper input power for the new neutralizer is about 17-18 W for thruster beam currents of about 0.71-1.76 A. Below about 0.71 A beam current, corresponding to thruster input power levels below about 0.9 kW, the input power is of the order of 28-30 W. The flight neutralizer operates at about 21-23 W keeper power for thruster beam currents of about 1.0-1.76 A, and about 31-35 W keeper power for beam currents below 1.0 ampere.

10 kW Ion Engine Cathodes

One focussed activity at NASA GRC is the development of the next-generation ion propulsion system as follow-on to the highly successful Deep Space One system. This advanced propulsion system is envisioned to incorporate a lightweight ion engine capable of operating over a 1-10 kW power throttling range with a 550 kg propellant throughput capacity. The engine concept under development has a 40 cm beam diameter, twice the effective area of the Deep Space One engine, while maintaining a relatively-small volume.[5,6]

The discharge cathode emission current requirement for the 40 cm engine can be approximated by:

$$I_e = 7.92 + 2.90 \cdot P_{in} \qquad [4]$$

where P_{in} is the engine input power in kilowatts. At 1 kW, the cathode emission current requirement is approximately 10.8 A, and at 10 kW it is approximately 37 A.

Several prototype cathode assemblies for the 40 cm engine have been fabricated and either integrated with the engine, or characterized separately in bell-jar testing. The discharge cathode assembly incorporates a 12.7 mm diameter cathode tube, has an enclosed-keeper electrode, and uses similar design and manufacturing processes as those employed in the Deep Space One cathode. A photo of one such assembly is shown in Figure 5.

Figure 6 shows the cathode tip and keeper face temperatures versus emission current for the discharge cathode assembly when operated in a diode mode with a planar anode. The cathode tip temperature, in degrees Celsius, can be estimated from:

$$T_{tip} = 171 \cdot \ln(I_e) + 506 \qquad [5]$$

The cathode tip temperature ranges from about 900 degrees C at 10 A, to about 1120 degrees C at 37 A. The corresponding keeper face temperature varies from about 445 degrees C to 570 degrees C. The internal pressure of the cathode varies with propellant flow rate and emission current. At about 15 A emission, the internal pressure varies from about 7.2 Torr at 0.30 mg/sec xenon flow rate, up to about 12.7 Torr at 0.88 mg/sec. At 35 A, the pressure varies from 11.8 Torr at 0.59 mg/sec to 15.0 Torr at 0.88 mg/sec.

The neutralizer cathode under development for the 40 cm engine is similar in mechanical design to that used on the Deep Space One engine. It must however operate at higher emission currents (beam plus keeper currents); from about 2.2 A at 1.0 kW to about 6.8 A at 10 kW, assuming a 1.0 A keeper current. The propellant flow rate dependency on beam current is expected to be comparable to that of the low-flow neutralizer.

Hall Thruster Cathodes

A number of engineering model cathode assemblies have been developed and fabricated at NASA GRC to accommodate Hall thruster development program testing at NASA GRC and elsewhere. These include enclosed-keeper 10-Ampere and 25-Ampere versions provided to U.S. industry and universities for Hall thruster integration testing, and enclosed-keeper 25- and 40-Ampere versions for in-house Hall development testing. A 10-Ampere flight version was also manufactured at NASA GRC and flown on a U.S. military spacecraft and is briefly discussed in a later section. Figure 7 shows both the 25-A and 40-A cathodes.

The 25-Ampere design is capable of operating over a 5:1 range in emission current – from 5 A to 25 A, without keeper current. Below 5 A, keeper current is required to sustain operation of the cathode. The cathode was successfully integrated and used in a wear test of a T-220 10 kW Hall thruster where it was operated for approximately 1000 hours at 2.0 mg/sec xenon flow rate at 20 A.[7]

The 40-Ampere design also operates over a 5:1 current range; from 8 to 40 A. Below 8 A, keeper current is required to sustain cathode operation. This cathode was used in performance evaluations of the T-220 Hall thruster,[8] and a 50 kW Hall thruster,[9] both conducted at NASA GRC.

High (> 40 Ampere) Current

Large HCAs (LHCAs) are appropriate for electron emission currents of ≥ 40 amperes. They typically operate at xenon propellant flow rates of ≥ 1.0 milligrams per second. Applications for LHCAs include discharge and neutralizer cathodes for high-power (> 10 kW) ion engines and 50-kW class Hall thrusters. These assemblies are normally constructed using hollow cathodes of at least 12.7 mm diameter.

The development of a cathode with an operational range of 50 to 100 A, appropriate for use with a 50 kW-class Hall thruster, is on-going at NASA GRC.[10] A prototype design (see Figure 8) has been developed and performance characterized on xenon propellant, at about 19.6 mg/sec, over a range of approximately 14 to 100 A. Figure 9 shows the cathode tip temperature as a function of emission current for this design. The tip temperature (in degrees C) can be described by the equation:

$$T_{tip} = 167 \cdot \ln(I_e) + 619 \qquad [6]$$

Design modifications will be required to reduce the emitter temperature as they exceed appropriate levels above about 40 A to yield adequate life, but the cathode should operate satisfactorily for integration with Hall thrusters for short-duration (less than 1000 hour) testing.

Spot and plume mode operation (as defined by the AC component of the electrode voltage) were characterized to define the keeper current requirements for this cathode during Hall thruster integration testing. Figure 10 defines the minimum keeper current required to maintain operation of the cathode in spot mode, as a function of the secondary current to an external anode for 1.96 mg/sec xenon flow rate. The minimum total emission current for the cathode is approximately 8 Amperes. At 10 A anode current, the keeper current must be at least about 9.8 A to ensure operation in spot mode. The required keeper current drops to zero current, for anode currents in excess of 45 Amperes.

Cathode Lifetime

A number of semi-empirical and analytical models have been developed to aid in cathode design. Coupled with a large database established from operation of hollow cathodes ranging from 3.2 mm-12.7 mm in diameter on both inert gases and mercury propellant, a rapid convergence in design with intended emission current requirement is relatively straightforward. However, the requirements on hollow cathode life are growing more stringent with the increasing use of electric propulsion technology and demand for larger total impulse capability. This necessitates the development of tools for estimation of cathode life, as determination of life solely by testing becomes impractical.

The life-limiters of hollow cathode operation are fairly well identified, and can be broken down into three general areas: protocols, operational, and environmental. Protocols involve the application of appropriate procedures to preclude cathode emitter poisoning. These include manufacturing processes, propellant purity and feed system cleanliness, as well as cathode conditioning and environmental exposure limits. Operational life-limiters refer to damage to the cathode emitter or heater as a consequence of operation. These include the e.9ffect of temperature (Ba evaportation and depletion, and tungstate formation), emission current (erosion, and excessive temperature), and heater cycle-life. Environmental life-limiters refer to structural damage to the cathode associated with erosion from the discharge or beam plasma of the electric thruster.

To date, simple models have been implemented to estimate hollow cathode life (for 6.4 mm diameter cathode technology) based on the known operational life-limiters. For heater cycle-life, a Weibull reliability analysis was performed on results from a sample of three heater cyclic life tests. The heaters failed after 10,500-12,900 cycles in an accelerated test profile. The reliability analysis yielded a B_1 life (99% survival with 90% confidence) of 3921 cycles, and a B_{10} life (90% survival with 90% confidence) of 6679 cycles. Subsequently, additional life testing has been performed under conditions more-closely replicating heater operation on a hollow cathode, with a single unit achieving approximately 42,000 cycles prior to failure.

A simple model has also been implemented to address the other two mechanisms for operational end-of-life: barium evaporation, and barium reactions forming barium tungstate. From reference 11, the barium evaporation rate is estimated by:

$$\frac{Q}{Q_0} = 400 \cdot e^{-1.61\times10^4/T} \cdot t^{0.5} \qquad [7]$$

where Q is the barium lost in time t hours, Q_0 is the barium at beginning-of-life, and T is the emitter temperature in degrees Kelvin. For this analysis, a 50% loss of barium is assumed to define end-of-life. Using the life test of reference 12, Eq. 1 is normalized to 28,000 hours at an estimated emitter temperature of 1520 degrees Kelvin, yielding:

$$t = 1.84x10^{-5} \cdot e^{\frac{3.22x10^4}{T}} \qquad [8]$$

The emitter temperatures may be estimated from data obtained for a variety of cathodes of differing geometry. Figure 11 shows cathode orifice plate (tip) temperature versus normalized orifice diameter for various emission currents. The curves obtained fit the form of the equation:

$$T_{tip} = a - b \cdot \ln(D_o) \qquad [9]$$

where D_o is the normalized cathode orifice diameter. The emitter temperature is assumed to be approximately 50 degrees higher than that measured at the cathode tip using a thermocouple.

The emitter life limit due to the formation of barium tungstate is estimated empirically from the Ba reduction data of Lipeles and Kan,[13] normalized to the conditions of 28,000 hours and 1520 degrees Kelvin, yielding:

$$t = 1.37x10^{18} \cdot 10^{-1.095^{-2}T} \qquad [10]$$

where T is the emitter temperature in degrees Celsius.

One can apply the above formulations to estimate the operational life-limiter for the discharge cathode of the Deep Space One ion engine. The number of discharge cathode ignitions/heater cycles is anticipated to be < 1000 to extend the engine throughput capability to at least 130 kg, so heater cycle life is not a factor. For 130 kg propellant throughput, the engine would be required to operate for approximately 33,700 hours at its lowest throttle condition (TH0, 0.48 kW) and approximately 11,900 hours at its highest throttle condition (TH15, 2.29 kW).

The Deep Space One discharge cathode emitter temperature (in degrees Celsius) is estimated to be:

$$T_{emitter} = 278 \cdot \ln(I_e) + 546 \qquad [11]$$

where I_e is the cathode emission current in amperes. Figure 12 plots the estimated discharge cathode life for both Ba evaporation and barium tungstate formation as a function of engine input power. Also plotted is the required operating time to process 130 kg. As noted, substantial life margin is expected for most of the engine throttling range. However at full power the model predicts no margin for barium tungstate formation, and only about 50% margin for Ba depletion at 130 kg throughput. Even though there are rather large uncertainties in the cathode life model, the results suggest that a redesign of the discharge cathode to reduce operating temperatures at full power may be warranted to extend engine life substantially beyond 130 kg.

A high-fidelity, and more comprehensive, cathode life model is under development at NASA GRC.[14] To date, the model examines the evolution and loss of Ba from the emitter impregnate. The coupled thermochemistry-diffusion model for xenon hollow cathode operation shows good qualitative agreement with observable phenomena in hollow cathodes.

The environmental limit to cathode life, structural damage due to ion erosion, is the topic of two efforts at NASA GRC: development of in-situ diagnostic capabilities to rapidly assess cathode wear, and an experimental investigation of the formation of high energy ions in high-current hollow cathodes.

The production of energetic ions in a hollow cathode discharge is well documented.[15-18] These energetic ions have been detected at energies well in excess of the discharge voltage. Energetic ions are capable of eroding not only ion thruster cathode and keeper electrodes, but also discharge chamber surfaces such as the screen grid. Such erosion processes can lead to thruster performance degradation and ultimately engine failure. In this regard, cathode production of energetic ions could be a potential thruster life-limiter particularly for long duration missions. The mechanism behind the production of these energetic ions is still unresolved.[18] Proposed mechanisms include multiply-charged ion processes, potential hills, charge-transfer effects, and z-pinch acceleration.[18,19] Though the theories explain the generation of energetic ions given the presence of certain mechanisms, there has not been compelling experimental evidence to support any of them.

In order to address and better understand this issue, an experimental investigation of the formation of high - energy ions in high-current hollow cathodes has been initiated at NASA GRC. The goal of this investigation is to characterize the energy distribution and estimate the magnitude of the energetic component of the emitted ions generated by a high current hollow cathode. These measurements will be made using a high fidelity low energy analyzer and Langmuir probe diagnostics. The data obtained will be used to develop a model that describes the production of the energetic ions as a function of operating condition.

International Space Station

NASA GRC manufactured and delivered the engineering model, qualification model, and flight cathodes for the International Space Station (ISS) plasma contactor system. During the total development effort, 31 cathodes (15 development model, 2 engineering model, 1 qualification model, 1 quality-control model, and 12 flight model) were manufactured. The qualification model and first 8 flight units are shown in Figure 13.

The ISS cathode is an enclosed-keeper configuration constructed from a 6.4 mm diameter refractory alloy tube. The cathode during development testing has demonstrated approximately 19,000 hours lifetime, and over 42,000 ignitions. For the ISS application, the cathode must emit up to 10 A of electron current to local space plasma and do so at a potential difference of less than 20 volts. It does so for xenon propellant flow rates ≥ 0.57 mg/sec. It also must be capable of at least 6000 ignitions with ≥ 99% reliability. The nominal keeper current is 3.0 A, and during operation without emission to the space plasma, consumes approximately 36-45 W. The complete requirements and verification matrix for the cathode may be found in Reference 2.

Low-Flow ISS Cathode

To increase the on-orbit lifetime of the plasma contactor, and hence reduce ISS logistics requirements for PCU re-flight, an activity was initiated to develop a reduced-flow rate cathode. A reduction in the required flow rate of xenon through the cathode would extend the on-orbit propellant load of xenon gas in the PCU beyond the nominal 2-year life. The operational requirements for the cathode (emission current, dynamic range, clamp voltage) would remain as specified, but modifications to the cathode design that could accommodate a reduction in flow rate (below the minimum of 0.57 mg/sec xenon) would be investigated.

To this end, a number of prototype cathodes were designed and fabricated which would accommodate rapid changes in the mechanical design to the critical features that control the required flow rate. To satisfy the electron current demand and potential required of the cathode, a minimum ion production is necessary. Hence design features that control the cathode ionization efficiency, and ion transparency were investigated. These features included the shape, size, and aspect ratio of the orifice of the enclosed keeper anode, as well as the cathode-anode electrode spacing, among others.

Five different cathode configurations were investigated, including the baseline ISS cathode design. From these tests, the most promising configuration was selected for more detailed characterizations. From these characterizations in diode and triode (emission currents up to 10 Amperes) mode operation, a flow rate reduction of approximately a factor of two (to about 0.30 mg/sec xenon) appeared feasible. Two engineering model versions of this low-flow cathode configuration were subsequently manufactured at NASA GRC. These units were to undergo detailed performance characterizations, with one unit to subsequently undergo mission-profile life testing to qualify the design for flight. Unfortunately resources under this ISS sustaining engineering task were only sufficient to complete manufacturing of the engineering model units.

Space Flight Applications

A number of cathodes designed and/or fabricated at NASA GRC have recently been successfully flown on space missions. Space flight cathodes constructed of 6.4 mm diameter cathode tubes have been qualified and launched on the Delta II, Space Shuttle, and Taurus launch vehicles. The Delta II launch involved the Deep Space One engine. The discharge and neutralizer cathodes were of a NASA design but manufactured by Boeing. To date the cathodes have accumulated over 13,000 hours of operation in space.

The plasma contactor systems were launched on the Space Shuttle October 11, 2000 on STS-92, integrated to the Z1 truss. The two cathodes on-orbit manufactured by NASA have accumulated approximately 10 ignitions and about 3000 hours of operation each while controlling the vehicle potential. A description of the on-orbit operation of these cathodes can be found in reference 20.

Three cathodes were also manufactured for the Russian Hall Effect Thruster Technology (RHETT) program.[21] One cathode was subsequently launched on the Hall-Effect electric propulsion system aboard the Space Technology Experiment (STEX) spacecraft of the National Reconnaissance Office by a Taurus booster. The cathode on-orbit performance was as expected.

Space Plasma Simulation

Hollow cathodes, and hollow cathode plasma sources, have been successfully implemented at NASA for the generation of large-scale plasmas to simulate the low-Earth orbit plasma environment for a variety of flight hardware test activities. Hollow cathodes in various designs are readily capable of generating low-temperature (< 0.2 eV) plasmas at densities in the range of 10^5-10^7 #/cc with good uniformity over large test volumes using inert gases. The hollow cathode source design requirements follow directly from the required source ion production rate. One may determine the required ion production rate of the source by noting that the production rate is equivalent to the ion loss rate. The ion loss rate is estimated from the required plasma density, the Bohm velocity, and the vacuum facility chamber surface area. Once the ion production rate is known, the hollow cathode can be designed accordingly.

One such space-simulation application involved the creation of a plasma of 1.0×10^6 #/cc density with better than a factor of 2 uniformity, at 0.2 eV temperature, at the NASA Plumbrook Space Power Facility (SPF) conducted for the Defense Nuclear Agency. The SPF chamber is the world's largest vacuum facility with dimensions 32 m diameter by 39 m height. Two hollow cathode plasma sources were used to generate the simulated space plasma environment, each positioned approximately 19 m above the chamber floor at opposing sides. The sources incorporated 6.4 mm diameter hollow cathodes with keepers, and an anode housing using an arrangement of permanent magnets to increase plasma production efficiency.

Commercialization

A Market Opportunity Analysis (MOA) of NASA's cathode technology was executed for NASA's commercial technology office by the Great Lakes Industrial Technology Center. The MOA concluded that NASA's cathode technology dominates the space applications market and has value in the growing terrestrial applications market. It also concluded that existing commercial suppliers of hollow cathodes would benefit from incorporating NASA's technology.

Based on the MOA, NASA is actively pursuing commercialization of cathode technology via licensing agreements with U.S. private industry for both space and terrestrial applications. The licensing agreements include rights to patented technology and transfer of know-how and trade secrets. NASA patents cover design aspects, manufacturing processes, and operating procedures for the cathodes. Part of each license agreement describes a plan for NASA to train the licensee and transfer this additional information. One such activity is described below.

Materials Processing
NASA has entered into a joint effort with a Midwestern firm for the development of a hollow cathode for use as an electron source for the production of diamond-like coatings, as replacement for thermionic filament cathodes. The cathode requirements for this commercial application are considerably different than those typically imposed for space flight applications. In space applications, reliability and long-life are paramount considerations, with cost of manufacturing relatively unimportant. Additionally, the interfaces and environment (e.g., ground-processing) are typically adjusted to accommodate the cathode requirements.

For this commercial application, cost of manufacturing is extremely important, with a manufacturing cost target of $1-2 per hour of operation. Although the reliability and lifetime requirements are modest, the cathode must be designed to accommodate the industrial conditions, which include a rapid-turn-around from atmosphere. Other requirements include operation on Ar gas at about 0.30 mg/sec, and an emission current capability from about 1-3 Amperes.

Multiple units have been manufactured and tested at NASA GRC and are under integration at the industrial partners' manufacturing facilities. Figure 14 shows the cathode tip temperature for one of these assemblies as a function of emission current. This temperature, in degrees Celsius, can be described by the equation:

$$T_{tip} = 226 \cdot \ln(I_e) + 864 \qquad [12]$$

The data indicate that the cathode operates well within its operational limits for the required emission current range.

Concluding Remarks

A number of hollow cathodes have been recently developed at NASA Glenn Research Center for a variety of electric propulsion applications, for emission current requirements ranging from 100 mA to 100 A. Activities include improving cathode efficiency, understanding life-limiting mechanisms and extending life, developing compact cathode assemblies for low-power electric propulsion, and developing high-current cathode designs for high-power engines.

Cathode assemblies have also been developed for spacecraft charge control for the International Space Station, for low-electron temperature space plasma simulation, and for ground-based materials processing applications. NASA is actively pursuing commercialization of cathode technology via licensing agreements with U.S. private industry for both space and terrestrial applications.

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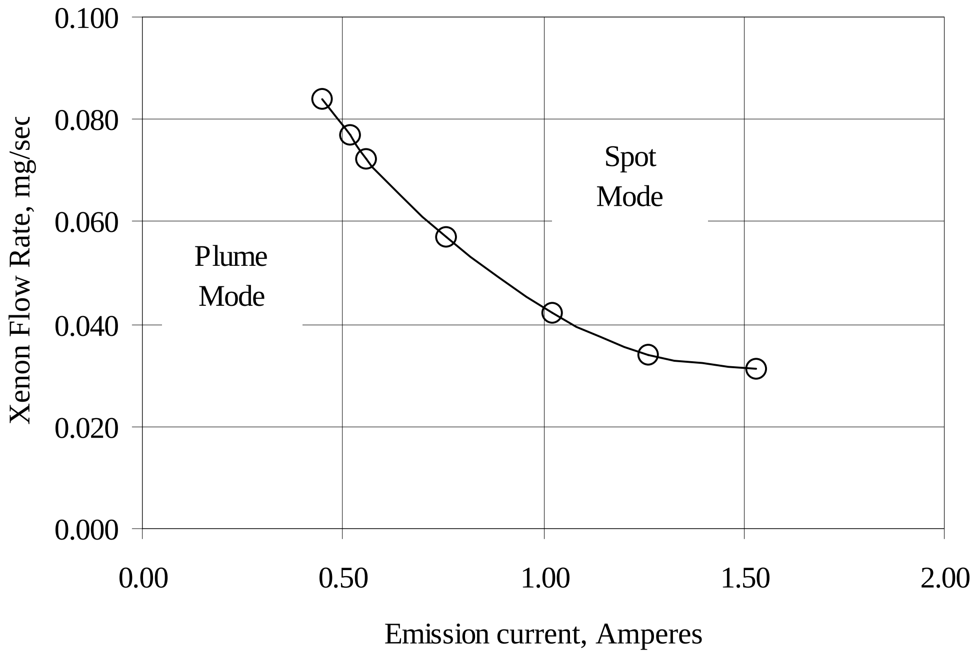

Figure 1 - Minimum flow rate for spot mode operation versus emission current; prototype small hollow cathode assembly.

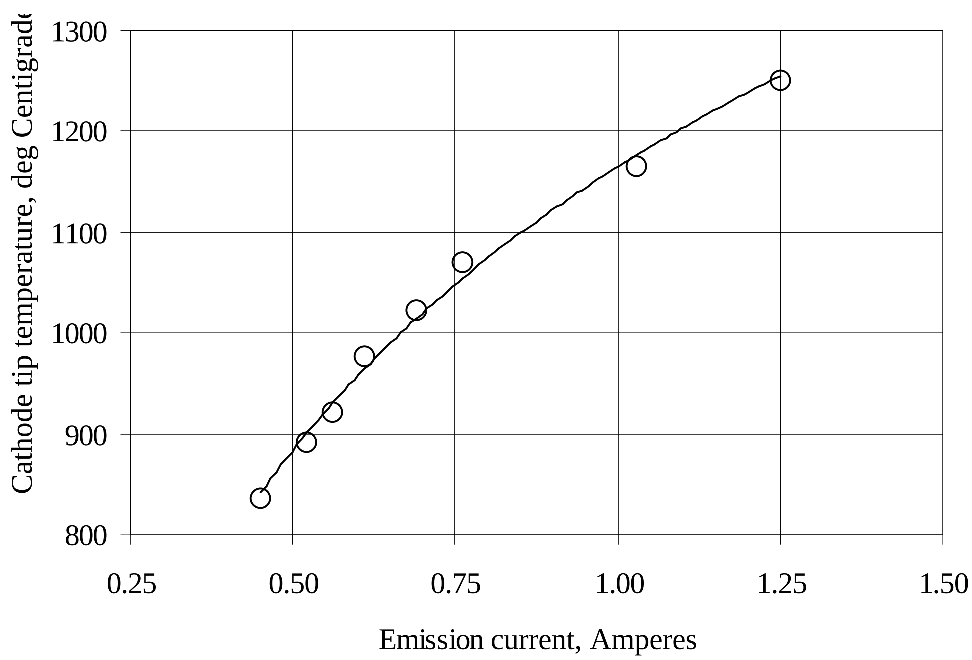

Figure 2 - Cathode tip temperature versus emission current; prototype small hollow cathode assembly.

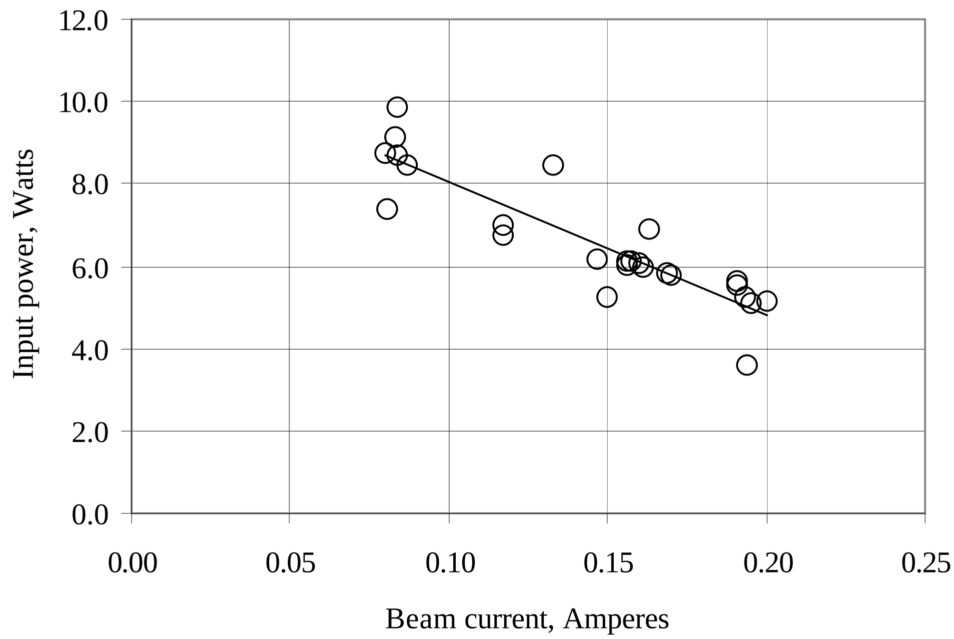

Figure 3 - Keeper input power versus beam current; neutralizer assembly.

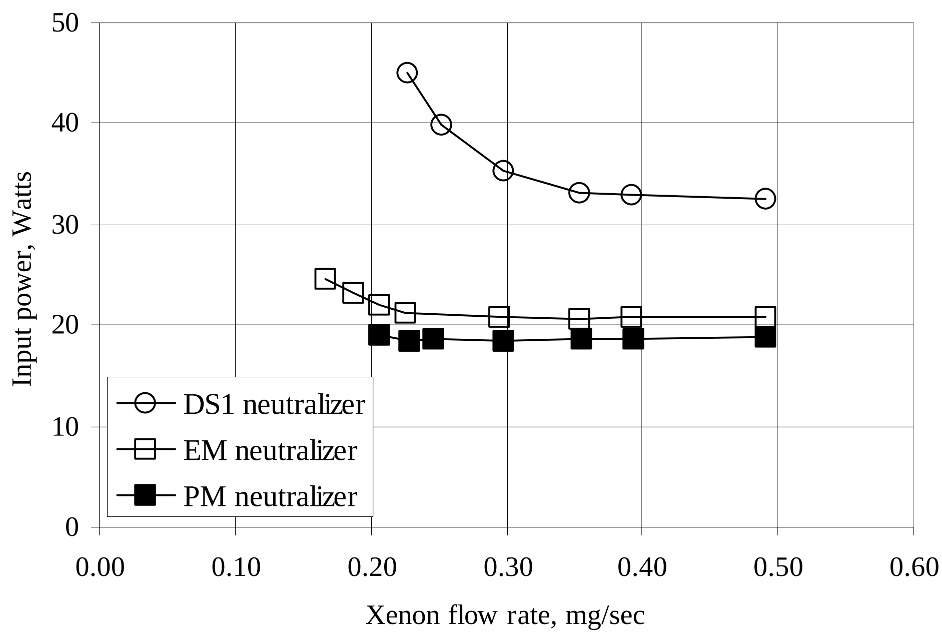

Figure 4 - Keeper input power versus flow rate.

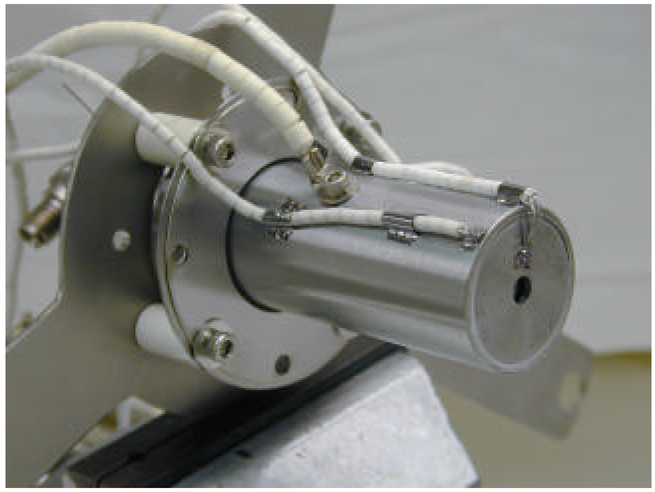

Figure 5 – Prototype discharge cathode assembly for 10 kW engine.

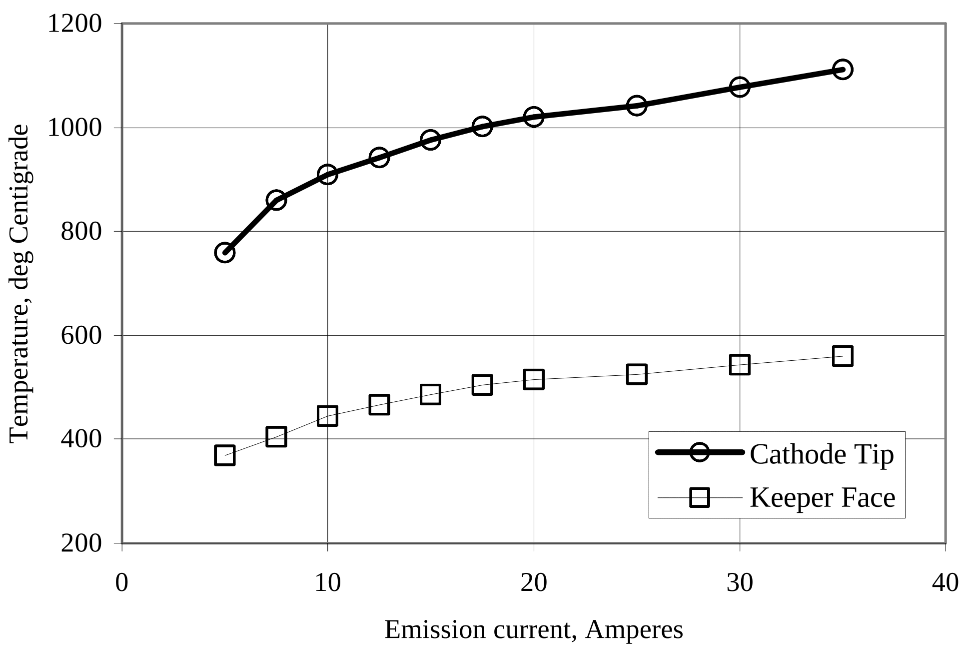

Figure 6 - Cathode temperatures versus emission current; 40 cm thruster discharge cathode operated with a planar anode.

Figure 7- Engineering model 25 A and 40 A cathodes.

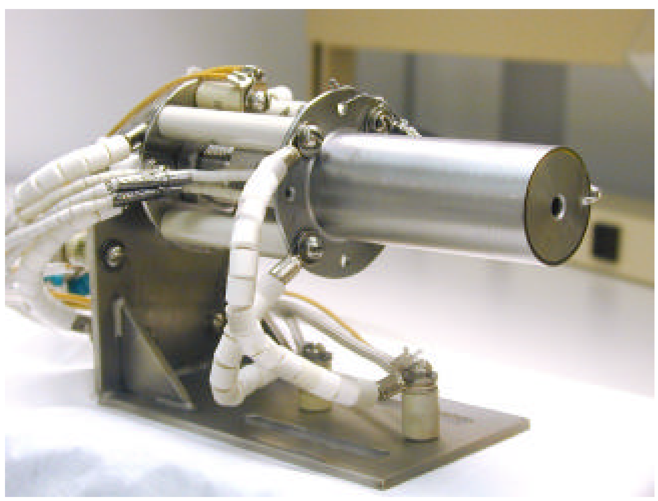

Figure 8 – Prototype large hollow cathode assembly.

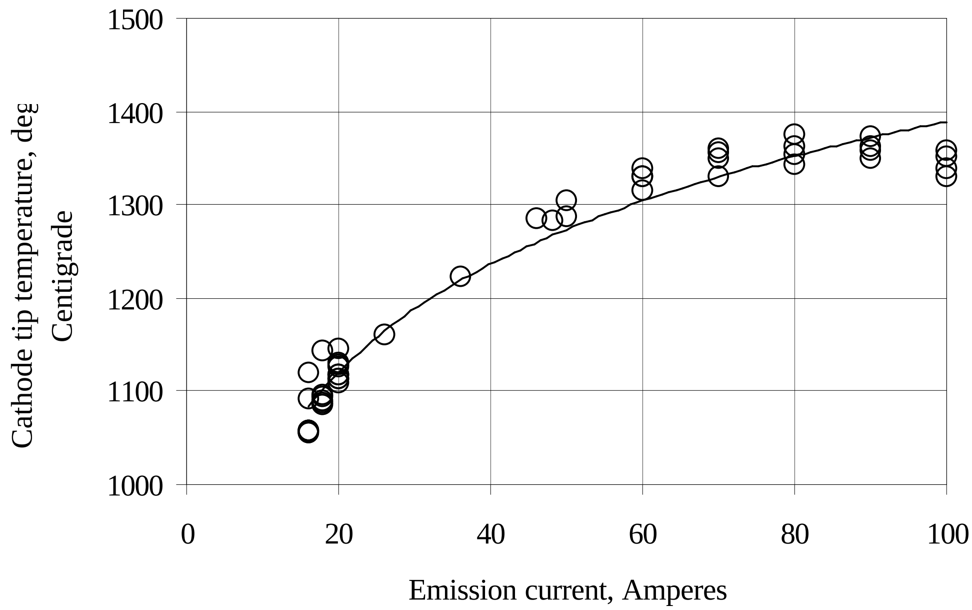

Figure 9 - Cathode tip temperature versus emission current; prototype large hollow cathode assembly.

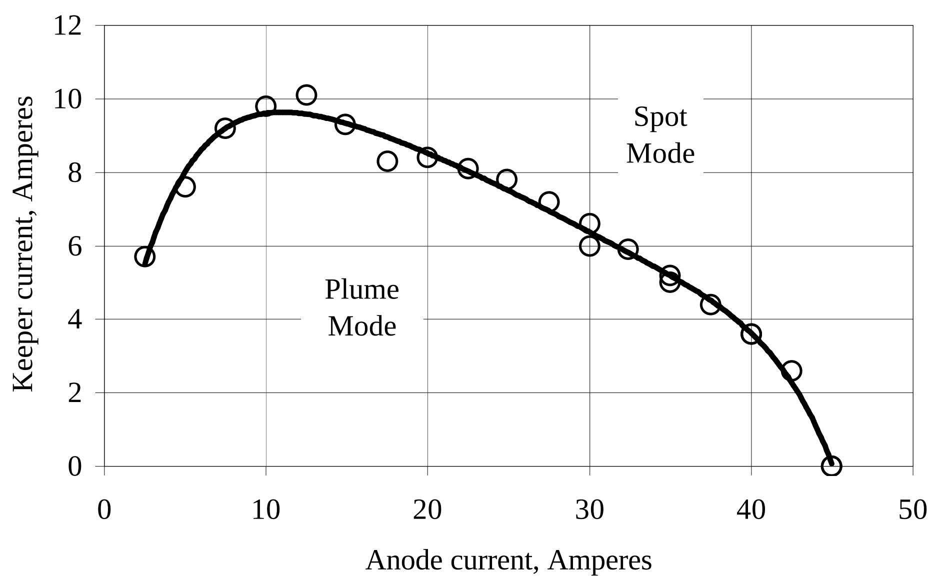

Figure 10 - Keeper current versus anode current for spot-plume transition; prototype large hollow cathode assembly; 1.96 mg/sec xenon flow rate.

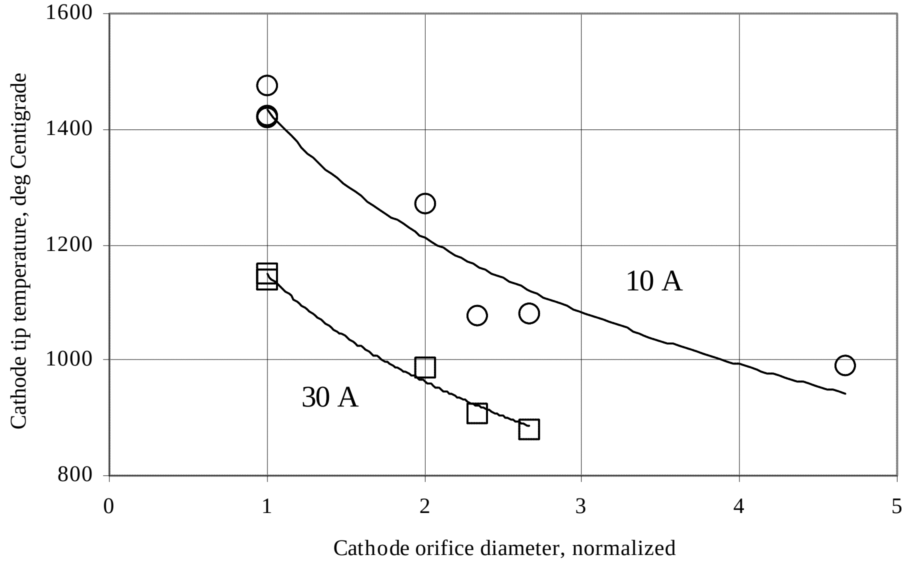

Figure 11 - Cathode tip temperature versus normalized cathode orifice diameter; 10 and 30 Amperes.

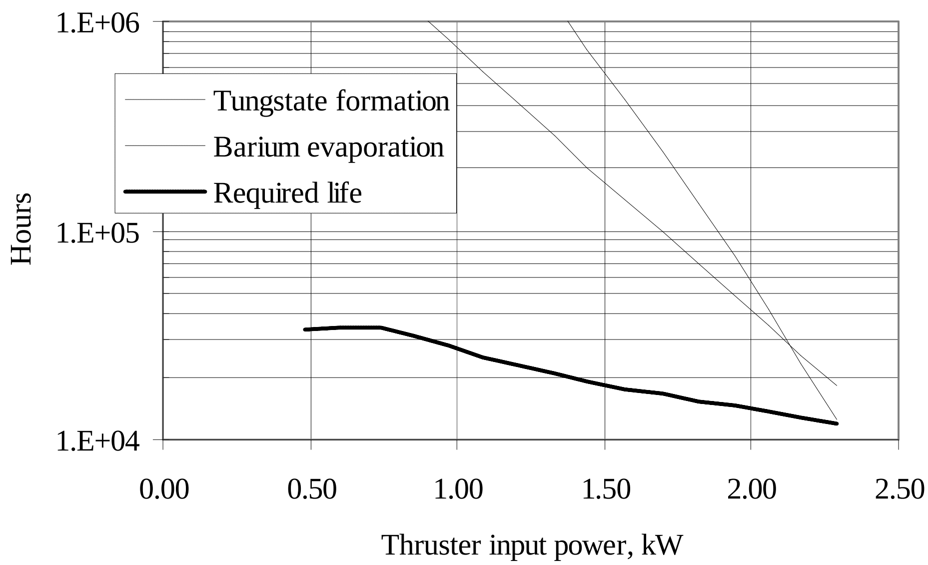

Figure 12 - Estimated discharge cathode life-limiters for the Deep Space One engine; 130 kg propellant throughput.

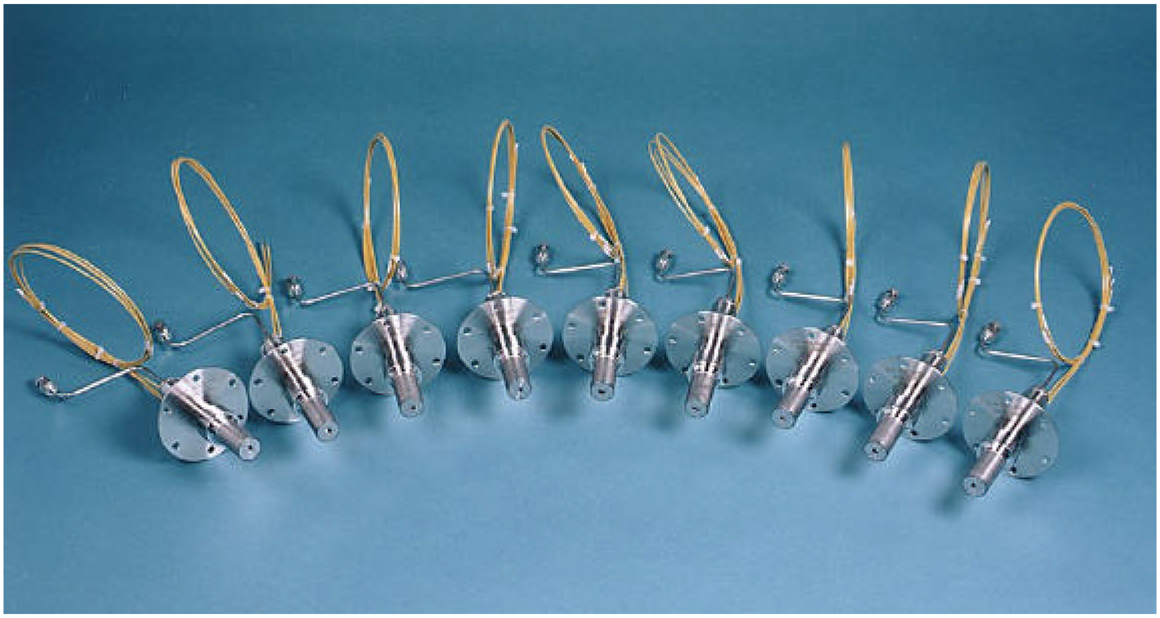

Figure 13 – Qualification and flight cathodes for the International Space Station.

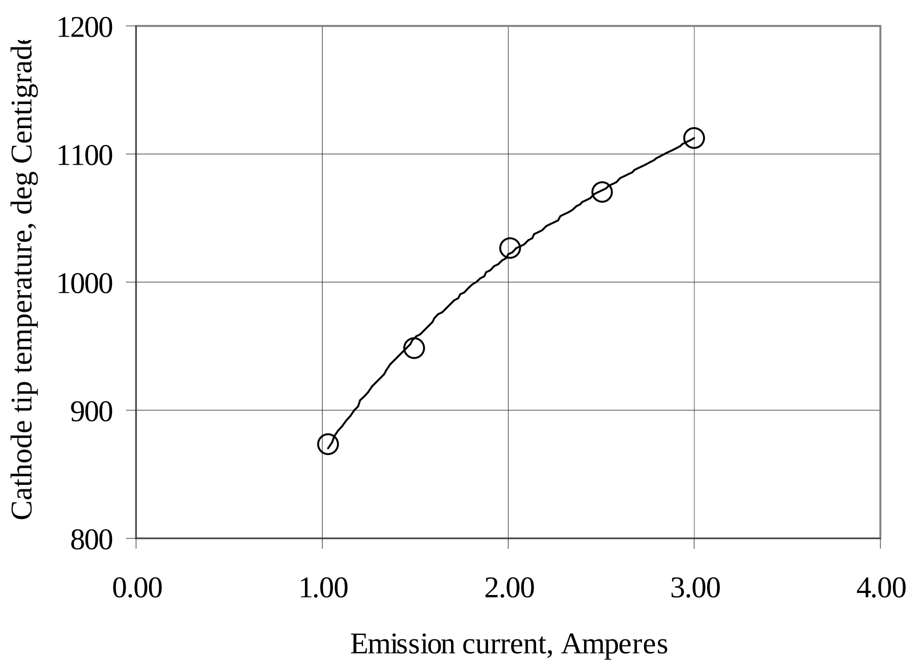

Figure 14 - Cathode tip temperature versus emission current; materials processing cathode assembly.