The New Horizons Spacecraft

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Abstract. The New Horizons spacecraft was launched on 19 January 2006. The spacecraft was designed to provide a platform for seven instruments designated by the science team to collect and return data from Pluto in 2015. The design meets the requirements established by the National Aeronautics and Space Administration (NASA) Announcement of Opportunity AO-OSS-01. The design drew on heritage from previous missions developed at The Johns Hopkins University Applied Physics Laboratory (APL) and other missions such as Ulysses. The trajectory design imposed constraints on mass and structural strength to meet the high launch acceleration consistent with meeting the AO requirement of returning data prior to the year 2020. The spacecraft subsystems were designed to meet tight resource allocations (mass and power) yet provide the necessary control and data handling finesse to support data collection and return when the one-way light time during the Pluto fly-by is 4.5 hours. Missions to the outer regions of the solar system (where the solar irradiance is 1/1000 of the level near the Earth) require a radioisotope thermoelectric generator (RTG) to supply electrical power. One RTG was available for use by New Horizons. To accommodate this constraint, the spacecraft electronics were designed to operate on approximately 200 W. The travel time to Pluto put additional demands on system reliability. Only after a flight time of approximately 10 years would the desired data be collected and returned to Earth. This represents the longest flight duration prior to the return of primary science data for any mission by NASA. The spacecraft system architecture provides sufficient redundancy to meet this requirement with a probability of mission success of greater than 0.85. The spacecraft is now on its way to Pluto, with an arrival date of 14 July 2015. Initial in-flight tests have verified that the spacecraft will meet the design requirements.

1. Introduction

A reconnaissance mission to Pluto and the Kuiper Belt immediately brings a number of issues to mind for those charged with the development of the spacecraft. Pluto's orbital distance from the Sun ranges from 28 to 40 astronomical units (AU) and has an orbital period of 248 years. Pluto's most recent closest approach to the Sun occurred in 1989, and it is now receding such that in 2015 the solar distance will be 32 AU. At those distances the travel time to Pluto is significant, and a spacecraft must be designed to be highly reliable, which implies both simplicity in design and sufficient redundancy to guard against single-point failures. The solar energy at Pluto is on the order of 1/1000 of the irradiance received in Earth orbit; therefore, the spacecraft must carry its own energy source. The only currently available technology is the radioisotope thermoelectric generator (RTG), which uses the thermal energy created by the decay of plutonium 238 to produce electrical energy. A single RTG was available to the development team, which limited the total electrical energy available to operate the spacecraft to approximately 200 W at the time of the Pluto encounter. The limitation in electrical energy also required a very efficient thermal design that did not require significant electrical energy to supply makeup heat. The communications system is affected by the power limitations as well as the necessity for managing variable data rates over these great distances (on the order of 500 bits/s to kilobits per second). The command and data handling system architecture, is dominated by the low data rates and the need to operate the spacecraft safely with large delays between the receipt of data and the ability to command the spacecraft, given the roundtrip light time at Pluto is 9 hours. Thus the spacecraft is required to operate autonomously throughout the close encounter with Pluto, and it will take months to return the entire data set to Earth.

With the limited power of a single RTG, both the power dissipation of each subsystem and the operation of the spacecraft were carefully considered. Design choices provided some relief, such as the design of the communication system described in section 6 of this paper. The remaining degrees of freedom in power management consist of the power cycling of instruments and other subsystems and the ability to minimize the power required for thermal management. The thermal management issues are described in section 8. Though it is possible to operate with various subsystem elements and instruments on at the same time, the plan for the Pluto encounter is to operate with only those units essential to a particular observation turned on at any one time.

The guidance and control system design is capable of pointing the spacecraft with sufficient accuracy for the science instruments to observe Pluto, its moons Charon, Hydra, and Nix, and other objects of interest during its journey. In addition, the attitude control of the spacecraft is capable of pointing the antenna toward Earth while requiring very little control activity during the journey to Pluto. For New Horizons, the guidance and control system operates in both spinning and 3-axis modes to meet these two requirements. Finally, the system provides the ability to make adjustments in the spacecraft trajectory to correct the residual launch trajectory errors, to correct for small disturbances encountered along the way to Pluto, and to modify the trajectory after Pluto to allow the reconnaissance of Kuiper Belt objects if NASA approves an extended mission.

The cost of flight operations for such a long-duration mission also influenced the system design. The first factor (already discussed) was to keep the spacecraft design simple. A second strategy invoked for New Horizons was to develop operational modes sufficiently robust to allow a very low level of ground monitoring and control during the long cruise periods from launch to Jupiter, and especially the nearly 8-year period from operations at Jupiter to Pluto. The operational approach (designated the "hibernation" mode) utilizes the robustness of operating the spacecraft in a passive spinning attitude control state. Special command and autonomy modes provide a logical path to transition into various safe states in case a spacecraft fault occurs during the periods of hibernation. And, the mode include a very-low-rate communication "beacon" to alert the ground should intervention be required.

During hibernation only the critical systems to assure health and safety are powered on. The operational scenario during hibernation is to monitor the beacon tone once a week. The beacon tone provides the signal for mission operations to determine if everything is OK (a green beacon) or if ground intervention is required (a red beacon). Housekeeping data are transmitted once a month to gain additional insight into spacecraft health. The spacecraft is returned to normal operational status for a short time each year for an annual checkout to confirm that all systems are operating to the required specification. The hibernation mode allows the spacecraft to be operated with a very small (less than 10) mission operation staff for the long cruise to Pluto, thus minimizing the operations cost of the mission. The impacts of the hibernation mode on the various subsystems are described in the sections below.

The primary purpose of the New Horizons spacecraft is to bring a selected set of science instruments sufficiently close to Pluto to gather information to meet the science requirements established by NASA in the Announcement of Opportunity AO-OSS-01, released on 19 January 2001. These requirements are described by Stern (2007b) and Weaver et al. (2007) in their papers in this volume. The New Horizons science team chose seven instruments for the New Horizons mission:

1. Alice, an ultraviolet spectrometer (Stein, 2007a)
2. Ralph, a multicolor imager/infrared imaging spectrometer (Reuter et al., 2007)
3. REX, an uplink radio science instrument with radiometer capabilities (Tyler et al., 2007)
4. LORRI, a long-focal-length panchromatic charge-coupled device (CCD) camera (Cheng et al., 2007)
5. SWAP, a solar wind monitor to address Pluto atmospheric escape rates (McComas et al., 2007)
6. PEPSSI, an energetic particle spectrometer to measure the composition of the ion species existing in region of Pluto (McNutt et al., 2007)
7. Venita Burney (VB) dust counter, which measures the density of fine dust particles in the solar system along New Horizons' trajectory from Earth to Pluto and beyond (Horanyi et al., 2007)

The placement of these instruments on the spacecraft is shown in Figure 1. The optical instruments (Alice, Ralph, and LORRI) are approximately co-aligned. The Alice instrument has a secondary optical aperture aligned with the high-gain antenna (HGA) used by REX, thereby allowing both instruments to make simultaneous measurements of Pluto's atmosphere during that portion of the mission trajectory when Pluto and Charon pass between the spacecraft and the Sun and the Earth, respectively. SWAP and PEPSSI are oriented to use both spinning and 3-axis control modes to collect data. The orientations are also set to minimize solar glint into the PEPSSI aperture and maximize the ability of SWAP to collect pickup ions during the approach and close encounter at Pluto. The VB dust counter is located on the side of the spacecraft pointing in the direction of the spacecraft velocity vector (the so-called "ram" direction) for the majority of the trip to Pluto.

The development team used these design constraints while transforming the concept of Figure 1 into the spacecraft shown in Figure 2. The following sections provide additional insight about design choices, principally focusing on how the spacecraft operates.

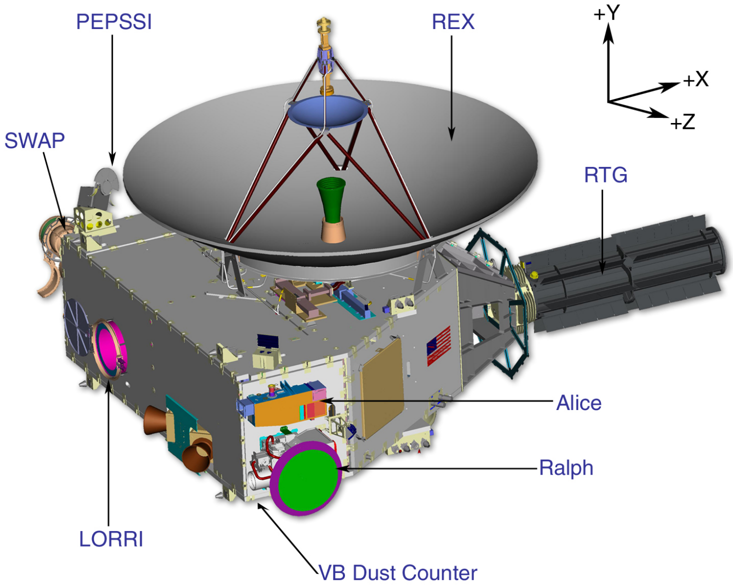

Figure 1. The New Horizons spacecraft supports seven science instruments and is powered by a single RTG.

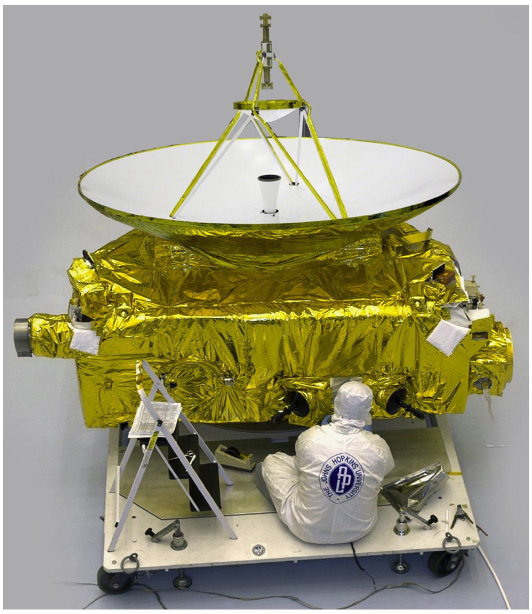

Figure 2. The New Horizons spacecraft operates either in spin mode or under 3-axis control. For the majority of its journey to Pluto, it spins about the +Y axis to which the antenna assembly is aligned (+Y low gain, medium gain, and high gain).

2. Spacecraft Configuration

2.1. MECHANICAL CONFIGURATION

The mechanical configuration of the spacecraft was driven by the need to spin the spacecraft about the axis defined by the antenna assembly; maintain spin axis alignment during launch; place instruments and propulsion system thrusters to meet observation requirements without interfering with one another; provide space for the spacecraft subsystems; and accommodate the launch loads. The Ulysses spacecraft (Wenzel et al., 1992) had similar design requirements. The New Horizons design team used the general configuration of Ulysses as a starting point for the spacecraft mechanical configuration, including the following:

1. A configuration that aligns the principal moment of inertia axis with the HGA, the +Y axis of Figure 1
2. Placement of the single RTG in the X-Z plane of the spacecraft to increase the angular momentum and maximize the distance of this source of radiation from the electronics and instruments

This configuration provides a highly stable platform to point the spinning spacecraft in a desired direction, usually toward the Earth.

The foundation of the spacecraft structure is a central cylinder located within the spacecraft main body. This single-piece, machined aluminum ring forging is the focal point for the load path and, as such, is the primary load-bearing member of the spacecraft. The central cylinder also incorporates the separating interface adaptor to the third stage. Directly attached to the central cylinder are the propulsion system propellant tank (centered within the cylinder) and the top, aft, and main bulkhead decks. The bulkhead decks provide flat mounting surfaces and interfaces for subsystems. Five additional side- and rear-facing decks attach to the aft, top, and main bulkhead decks. The decks are of an aluminum honeycomb–type sandwich structure in which areas subject to high stress include magnesium edge members.

The titanium RTG support structure is attached to the decks on the opposite side of the cylinder from the instruments. The support structure is made primarily of titanium to provide high stiffness, low mass, and low thermal conductivity. Since the RTG support structure also had to provide electrical isolation, an all-metallic electrical isolation system was developed that didn't compromise structural stiffness. The isolation system employs multiple layers of non-conductive surface coatings applied to metallic plates located at the structure joints. To meet RTG interface requirements, an aluminum interface flange is attached to the titanium structure on the RTG side. Additional requirements to ensure proper stiffness and prevent deterioration of material properties due to RTG induced heating necessitated the addition of cooling fins to the RTG adaptor collar. The cooling fins assured that the strength of the RTG interface flange which is made of aluminum had sufficient margin during the 3rd stage burn when the acceleration peaked at 10.8 *g*s and the interface temperature reached 260°C.

The propulsion system (hydrazine) propellant tank is centered on the axis defined by the principal axis and the center of mass of the spacecraft. At this location the variation in the fill fraction as the propellant is used has minimal impact on changes in the alignment of the spin axis. This ensures that when the spacecraft is spinning the HGA remains pointing in a fixed direction throughout mission life. The tank has a capacity of 90 kg of hydrazine. The total mass at launch was 77 kg of hydrazine and helium pressurant to meet the maximum allowable launch mass and to ensure stability of the combined spacecraft and third stage stack.

The placement of the propellant tank has two further advantages. It is surrounded by the cylinder that carries the launch loads directly into the third stage attach fitting. Since the tank is between the instruments and most of the electronics and RTG, the propellant further reduces the radiation effects of the RTG. The modest radiation output of the RTG and its placement guarantees that total dose received by the electronics is less than 5 krads (from all sources) throughout the primary phase of the mission. The majority of this dose is received at Jupiter when the spacecraft passes at a distance of 32 Jupiter radii (R_J). In addition, waste heat from the RTG and the electronics is used to keep the hydrazine sufficiently above the freezing point without significant use of makeup heater power.

2.2. SYSTEM CONFIGURATION

Figure 3 provides an overview of the major functional system elements and their connectivity (Kusnierkiewicz et al., 2005).

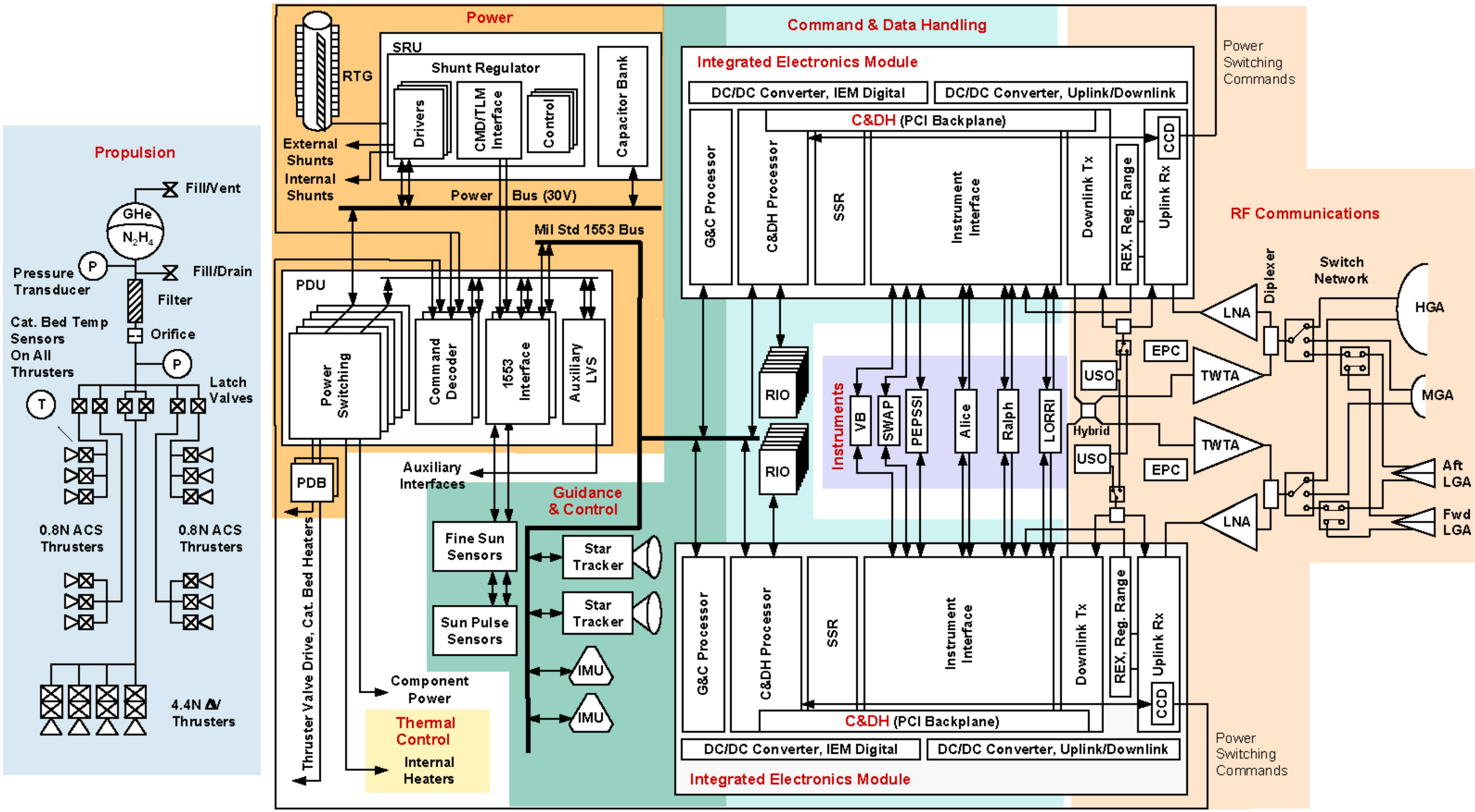

Figure 3. The New Horizons spacecraft is designed with significant redundancy to ensure that the mission requirements are met after a 9.5-year flight to Pluto.

To meet the overall reliability requirements there is significant block redundancy. There are two integrated electronic modules (IEMs). Each IEM contains a command and data handling (C&DH) processor; a guidance and control (G&C) processor; RF electronics, which are the heart of the communication system; an instrument interface card that provides connectivity to each instrument; and a 64-Gbit solid state recorder (SSR). In addition there is block redundancy in most of the remaining system elements, including the traveling wave tube amplifiers (TWTAs), star trackers, and inertial measurement units (IMUs). Other system elements (such as the power distribution unit, PDU) include redundant boards or have redundancy at the circuit level. For example, the shunt regulator unit has triply redundant control elements (Carlsson, 2005).

To improve system reliability, significant cross-strapping below the block level is included in the design. For example, a C&DH processor in IEM 1 can be used as the controlling unit and the G&C system can operate using the G&C processor in IEM 2. The instruments' interfaces and much of their electronics (up to the focal plane detectors) are redundant, and much of the redundant circuitry can be accessed from the instrument interface card in each IEM. This cross-strapping is continued with each C&DH processor being able to access each of the redundant 1553 buses that provide the data pathways to the IMUs, star trackers, and sun sensors.

Only a few elements of the system are not redundant. These include the RTG; the propulsion system tank, line filter, and orifice; the RF hybrid coupler; and the HGA. Each of the units has a very robust design and a long history of failure-free service.

3. Propulsion Subsystem

The propulsion system (Stratton, 2004) includes twelve 0.8-N thrusters, four 4.4-N thrusters, and the hydrazine propellant tank and associated control valves, as shown in Figure 3. The titanium propellant/pressurant tank feeds the thrusters through a system filter, a flow control orifice, and a set of latch valves that prevent flow of the propellant until commanded to the open position after launch. Helium was selected as the tank pressurant instead of nitrogen to allow the loading of an additional kilogram of hydrazine. Blowdown curves for the system are shown in Figure 4. Measurements of tank pressure and temperatures at various points in the system allow the mission operations team to monitor system performance and the amount of propellant remaining in the tank.

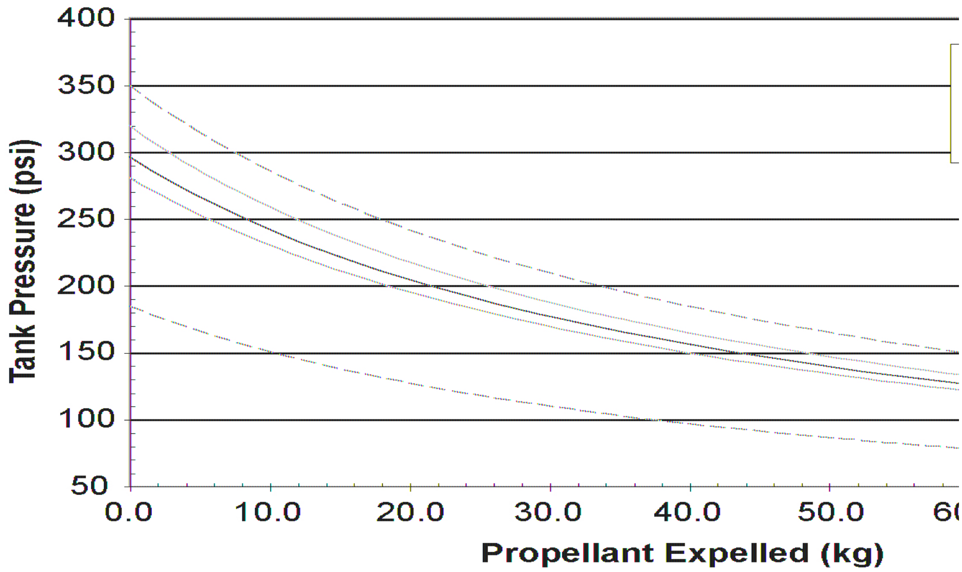

Figure 4. New Horizons propulsion system blowdown profile.

The 16 rocket engine assemblies (REAs) are organized into 8 sets and placed on the spacecraft as shown in Figure 5. Pairs of the 0.8-N thrusters (each thruster from a different set) are usually fired to produce torques and control rotation about one of the three spacecraft axes. The one exception to the use of coupled thruster firings to control spacecraft rates is that of controlling rates about the spacecraft ±X axis during science observations, where uncoupled thruster firings are required to meet the maximum spacecraft drift rates allowed during this operational mode. Control rates for each of the spacecraft axes are shown below in Table II. One pair of the 4.4-N thrusters is aligned along the $-$Y spacecraft axis to provide ΔV for large propulsive events such as trajectory correction maneuvers (TCMs). The second pair of 4.4-N thrusters is aligned to produce thrust along the +Y axis. These thrusters are rotated 45° in the Y–Z plane to minimize the plume impingement on the HGA dish. The net propulsive effect of these thrusters is therefore reduced. They still provide the required redundancy and the ability to generate thrust in both directions without a 180° rotation of the spacecraft.

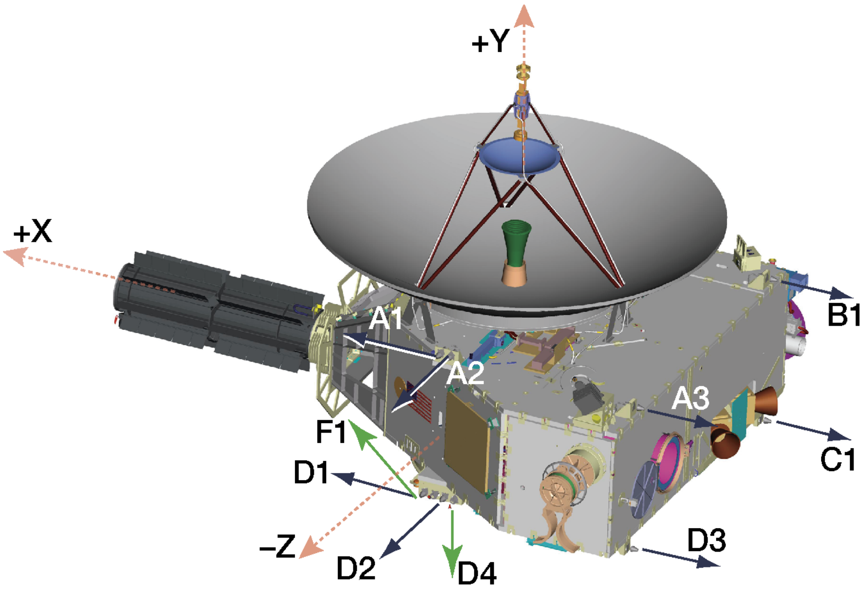

Figure 5. The propulsion system thrusters are organized in eight sets (six of which are shown in the figure). The sets are distributed such that firing of pairs of 0.8-N thrusters located on opposite sides of the spacecraft produces couples for attitude control. The 4.4-N thrusters are paired in two sets on the −Y/+Z spacecraft edge (F1 & D4 shown) and the −Y/−Z spacecraft edge (not shown) and are used when larger ΔV maneuvers are required.

Each thruster requires a heater to warm its catalyst bed to a minimum temperature prior to use. Each thruster catalyst bed has both a primary and a secondary heater element, with each element drawing approximately 2.2 W of power. Control of the catalyst bed heater circuits is grouped functionally by pairs (to minimize the number of switches required), so that a total of 16 switches control the heater elements, allowing great flexibility to operate the spacecraft safely while drawing the minimum required power. Operational requirements for catalyst bed heater use are enveloped by the spacecraft power usage, discussed in section 7.3 and shown below in Figure 10).

The pulse duration and total on-time of each thruster are commanded very precisely, providing accurate control of the total impulse generated during a maneuver. The 0.8-N thrusters can be turned on for periods as short as 5 ms, as further described in section 4.2. The initial propellant load was allocated between primary mission TCMs, attitude control (including science and communication operations), and primary mission margin. At the end of the primary mission, sufficient margin may allow for an extended mission to one or more objects in the

Kuiper Belt. This allocation is given in Table I. The original margin was augmented during the final mission preparations when the unused dry mass margin was converted to additional propellant. Due to the excellent performance of the launch vehicle, the total margin has nearly doubled over that budgeted in Table I.

Table I. New Horizons propellant budget allocations at launch.

	ΔV (m/s)	**Propellant Usage (kg)**
Primary mission TCM	110	22.3
Altitude control		29.3
Primary mission margin	132	25.2
Original margin allocation	*91*	*17.5*
Additional margin obtained from unused spacecraft dry mass allocation	*41*	*7.7*
Total navigation ΔV	**242**	
Total propellant load		**76.8**

Nutation is controlled (damped) both passively and actively on New Horizons, depending upon the spacecraft state and the current event executing (e.g. TCM, precession maneuver, etc). If the spacecraft is performing a TCM or if the spacecraft is in Sun Acquisition state, then nutation damping is exclusively passive. If the spacecraft is precessing, then active nutation control is used only if the nutation angles grows exceedingly large. The size of this threshold is determined by onboard tunable parameters. Near the end of a precession maneuver (last few tenths of a degree), special onboard logic will fire thrusters so that (1) precessing the spacecraft reduces nutation angle and (2) reducing nutation angle will precess the spacecraft in the proper direction. Because the rotational frequency and nutation frequency are different values, it could take several additional minutes to complete a precession maneuver. Since the threshold to actively control nutation angle is a tunable parameter, the threshold could be decreased as propellant is depleted to ensure the nutation angle drops below a pre-determined value in a set amount of time. There should always be some small residual passive nutation damping through the life of the mission. Thus the propellant required for nutation control is very small and subsumed in the value for attitude control.

4. Guidance and Control

The guidance and control system uses a suite of sensors to determine the attitude of the spacecraft, the propulsion system thrusters as actuators, and one of the redundant guidance and control processors to compute the required control actions necessary to meet the commanded attitude state (Rogers et al., 2006). The attitude sensors (see Figure 3) include the IMUs, the star trackers, and the sun sensors. The guidance and control system is capable of providing spin axis attitude knowledge of the spacecraft to better than ±0.027° (3σ) and spin phase angle knowledge within ±0.30° (3σ). The same ±0.027° knowledge (3σ) is provided for all axes when the spacecraft is in 3-axis mode. The control algorithms must maintain the spacecraft attitude to within ±0.0014° (3σ) and the spacecraft rotational rate to within ±0.0019°/s (3σ).

4.1. ATTITUDE CONTROL MODES

Figure 6 illustrates the various attitude control modes (3-axis, active spin, and passive spin) and the four nominal state classes (TCM, operational, Earth acquisition, and Sun acquisition). TCMs can be conducted in any of the three attitude modes. The choice of mode depends on factors such as the size of the ΔV to be imparted to the spacecraft, the desire to maintain the telemetry link while executing the burn, etc. The various operational states depend on spacecraft activity and other constraints such as the duration of the mode, the need for conservation of propellant, the level of ground monitoring, etc.

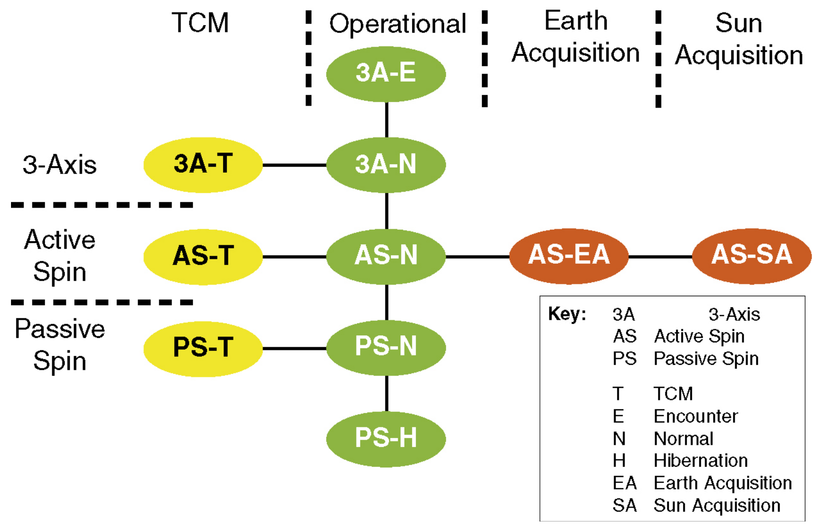

Figure 6. The guidance and control system provides three operating modes to support TCMs, nominal operations, and two safe states (Earth and Sun acquisition).

At launch, the spacecraft was put in the Passive Spin Hibernation (PS-H) state. This state precludes any onboard attitude control and minimizes the electrical power demand. As the name denotes, it is the primary mode used during the long cruise from Jupiter to Pluto, when there will be a low level of monitoring from the ground and the spacecraft is put into "hibernation". The Passive Spin Normal (PS-N) state also precludes any onboard attitude control activity, but does not limit other power demands. This mode is used during significant portions of the cruise from Earth to Jupiter and at other times in the mission. The Active Spin Normal (AS-N) state is used when an attitude maneuver is required and the spacecraft is spinning. This state is used to maintain the spin rate at its nominal 5 rpm value or to precess the spacecraft to a new orientation. The 3-axis mode states allow the spacecraft to be rotated about any spacecraft axis or desired axis to point in a particular orientation. These modes are used to point a particular instrument to a desired target or to scan a particular instrument field of view across a particular target (see Figure 7).

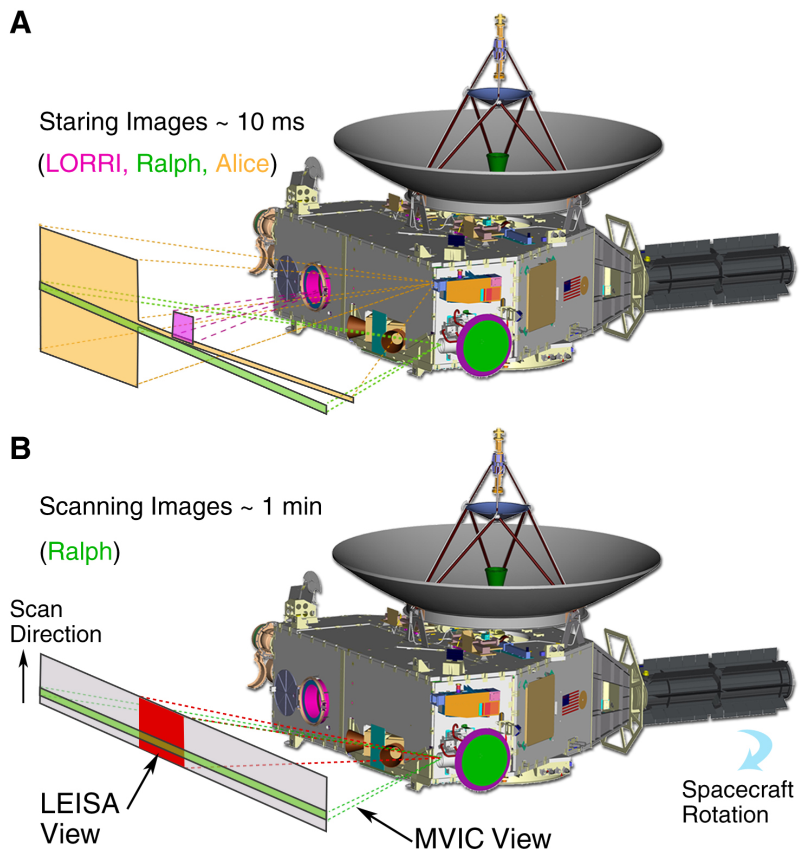

Figure 7. (A) The optical instruments (Ralph, Alice, and LORRI) are aligned such that their fields of view nearly overlap. Only small motions of the spacecraft are needed to image the same area.

(B) Precise control of the spacecraft attitude is needed both for staring and to provide smooth scanning motion about the +Z axis for Ralph.

The 3-Axis Normal (3A-N) state is used during most instrument activities, including instrument commissioning, engineering tests, and the Jupiter science campaign. The 3-Axis Encounter (3A-E) state is used during the period of closest approach at Pluto where acquiring data has priority. The difference between 3A-N and 3A-E is how the onboard autonomy system reacts to critical faults (e.g., a reboot of the C&DH processor). In the 3A-E state, the autonomy system attempts to correct the fault while keeping the spacecraft in 3A-E state to maximize science data acquisition. Conversely, critical faults in 3A-N state (and other states) result in the autonomy system exercising one of its two “go-safe” chains, Earth Acquisition and Sun Acquisition.

The Earth Acquisition Go-Safe chain transitions the spacecraft from its current mode and state into Active Spin Earth Acquisition (AS-EA) following the state transitions shown in Figure 6. Once in Earth Acquisition mode, the autonomy system configures the spacecraft with the HGA pointed to Earth using emergency uplink (7.8125 bits/s) and downlink (10 bits/s) rates.

The second “go-safe” chain, Sun Acquisition Go-Safe, is exercised by the autonomy system as a last resort in the event that communication from the ground cannot be established with the spacecraft while in Earth Acquisition mode. This can be the result of a corrupted onboard ephemeris (which is needed by the spacecraft to determine Earth position), failed star trackers (which are needed for inertial reference), prolonged period of command inactivity (command loss timeout), etc. In Sun Acquisition (AS-EA) mode, the autonomy system configures the spacecraft to orient the HGA towards the Sun and to send a radio signal that a critical fault has occurred on the spacecraft (a so-called “red” beacon tone). Since Sun Acquisition Mode assumes that any Earth knowledge has been lost, the Sun provides the only “guidepost” for the spacecraft pointing. Additional information on the autonomy system is provided in section 9.

4.2. ATTITUDE CONTROL REQUIREMENTS AND PERFORMANCE

The thrusters generate the control torques that change the attitude motion required to execute the slews to targets and to point or scan the desired instrument toward or across the target. There are two major strategies of observation by the optical instruments:

1. To first point the appropriate instrument field of view (see Figure 7) and "stare" with the spacecraft rotation limited to values of less than one image pixel during the maximum expected exposure time. This rate limit was determined by the size of the LORRI image pixel and the expected maximum exposure time of 10 ms.
2. To scan about the spacecraft Z axis with a rate control sufficiently precise to allow the Ralph instrument to perform time delay integration of faint images.

Both of these requirements are met if the body rates are controlled with an accuracy of 34 μrad/s.

The thrusters are used in a pulse width modulation mode to achieve the desired control precision, with the IMUs used as the fine rate control sensors. The minimum pulse width available in the system of 5 ms is a value set by design based on the actuation time constants of the thruster control valves. The actual performance of the guidance and control system, as measured during the commissioning tests in the spring of 2006, are given in Table II. These numbers improve (decrease) as the feed pressure of the spacecraft propulsion system decreases.

Table II. The spacecraft G&C system controls the body rates using small torques provided by thruster pairs on the spacecraft. The resulting minimum control body rates, as measured after launch, are provided by the thruster pulse duration of 5 ms.

Thruster Combination	Average Rate Change from 5-ms pulse (μrad/s)
+X axis (A2)	23
–X axis (B2)	16
+Y axis (C1/D1)	25
Y axis (C3/D3)	23
+Z axis (A1/D3)	15
–Z axis (A3/D1)	16

5. Command and Data Handling

The Command and Data Handling (C&DH) functions—command management, science and engineering data management, timekeeping, and the capability to provide autonomous system recovery and safing—are implemented with the resources within the two redundant IEMs (see Figure 3):

- C&DH processor card
- SSR card
- Instrument interface card
- Critical command decoder on the uplink card
- Downlink formatter on the downlink card

Communication between cards within the IEM is over a peripheral component interconnect (PCI) backplane. Remote input/output (RIO) units provide temperature and voltage measurements for monitoring the health and safety of the spacecraft. These RIO units communicate with the C&DH using an inter-integrated circuit (I^2C) bus. A MIL-STD-1553 serial bus interface is used between the IEMs to allow for redundancy and cross-strapping as well as the transfer of commands and data between G&C components and the G&C processors.

The C&DH system consists of these hardware resources and the software running on the C&DH processor. The processor in IEM 1 is designated C&DH 1 and the processor in IEM 2 is designated C&DH 2. During normal operations, the C&DH section of one IEM is fully powered and designated as prime, the so-called "bus controller", while the second unit is powered in a standby mode. During the long cruise between Jupiter and Pluto, this second unit will be powered off. The fault protection system monitors, from multiple locations, a pulse-per-second "heartbeat" from the bus controller. If the heartbeat is interrupted for a period of 180 seconds, the redundant C&DH will be powered on and take command as the bus controller. In addition, other safing actions will be taken as described in the section on fault protection and autonomy.

5.1. COMMAND MANAGEMENT

Ground commands are received via the RF uplink card. A critical command decoder on the uplink card provides a simple, robust mechanism to receive and execute commands without the operation of the C&DH processor. These include C&DH processor resets, switching of bus controllers, and all other power switching commands. The C&DH software receives CCSDS (Consultative Committee for Space Data Systems) telecommand transfer frames from the ground via the RF system uplink card. The C&DH software extracts telecommand packets from the transfer frames. These telecommand packets are delivered to other subsystems on the spacecraft or to the C&DH processor itself. The telecommand packets that the C&DH processor receives can contain commands to be acted upon in real time or commands to be stored for later use. Commands intended for later use are designated as command macros. Command macros can consist of a single simple action (such as commanding power on or off to a subsystem) or a string of switching commands and information (data) for use by the various subsystems. Each C&DH subsystem has 0.75 Mbytes of storage for command macros.

The C&DH subsystem supports time-tagged rules. These trigger and execute one or more of the command macros when the value of the onboard mission elapsed time (MET) is greater than or equal to the value of the MET specified in the rule. Mission Operations uses time-tagged rules to implement almost all onboard activities, including science activities at Jupiter and Pluto. There is storage for 512 time-tagged rules.

The C&DH software implements the onboard autonomy system. An autonomy rule is a postfix expression whose inputs can include any spacecraft telemetry point. An autonomy rule is triggered when it has evaluated to "true" for the required number of times in a given interval. A triggered autonomy rule also executes one or more of the command macros. There is storage for 512 autonomy rules. The autonomy system is further described in section 9.

5.2. TIME MANAGEMENT

The accuracy of the correlation of MET to Universal Time (UT) is of major importance to support navigation, G&C activities, and the collection of science data. The ultra-stable oscillator (USO) used as the onboard source of a 1 pulse per second (PPS) signal maintains the spacecraft time base. Careful design of the ground to MET clock register in the C&DH system ensures onboard correlation of better than ±4 seconds when the spacecraft is at Pluto (a 9-hour round trip light time delay). Science instrument time correlation (post-facto) requirements are ±10 ms for REX, LORRI, Ralph, and Alice. Post-launch measurements in July 2006 verified that the post-facto timing correlation is well within this requirement.

5.3. DATA MANAGEMENT

The instrument interface card communicates to each of the instruments. Communication consists of:

1. Providing commands and MET from the IEM to each instrument
2. Collecting housekeeping data and low-rate science data from the instruments to the IEM
3. Receiving and formatting instrument high-rate science data and IEM-generated header data
4. Distributing spacecraft time markers to each instrument
5. Processing and transferring IEM analog telemetry voltages (data input from the RIO devices using the I^2C bus)

Each instrument interface card (in IEM 1 and IEM 2) provides connections to both sides of each instrument's redundant electronics, thereby providing significant cross-coupling and increased reliability.

The SSR consists of 64 Gbits of non-volatile storage organized in 16 independently addressable segments corresponding to the 16 physical memory stacks on the SSR card. Raw science data can be streamed onto the recorder at rates up to 13 Mbits/s. Once a segment is filled, the recorder switches to the next segment. Raw science data is read off the SSR, compressed, and written back to the recorder for later transmission. Erasing of the SSR is done on a segment basis, after all data in that segment are either transmitted or compressed and stored to another segment. Once a segment is erased it is available for storage of new data. The C&DH software provides several mechanisms to control the writing and playback of data onto the recorder. The recorder uses the concept of data types to allow differentiation of, and thereby control, the data. There are a total of 51 data types, including raw (non-packetized) high-speed science data; compressed versions of the science data, with each type of compression of a particular type of science data resulting in a different data type; data from instruments that produce low-speed, packetized science data; and instrument and spacecraft housekeeping data.

The C&DH software also implements SSR bookmarks. Bookmarks allow access of data on the SSR associated with a specific activity. Bookmarks are opened at the start of the activity and closed at the end of the activity. Data types are included in the bookmark. Bookmarks are a substitution for using MET to define the data to be compressed or played back. All commands to compress or play back data can be specified using either bookmarks or MET. The priority of the data types to be played back from the SSR can be specified. This applies both to a playback by bookmark or a playback by MET.

The New Horizons software implements both lossless and lossy compression. Non-packetized science data are read off of the SSR, compressed and formed into CCSDS packets, and written back to the SSR. There is also the option to read the non-packetized science data off of the SSR, form the data into CCSDS packets without doing any sort of compression, and write the data back to the SSR.

Lossless compression can be combined with subframing, or windowing. Rather than performing the lossless compression on the entire image, it is possible to specify up to eight subframes of the image and then perform the lossless compression on the data within these subframes.

A special method of subframing is applied to images collected by the Linear Etalon Imager Spectral Array (LEISA) infrared sensor within the Ralph instrument. This is known as dark-sky editing. A LEISA frame consists of one spatial dimension (X) and one wavelength dimension (Y). Observations are made with the spacecraft oriented such that a rotation causes the observed object to pan across the focal plane at a rate of approximately one spatial pixel per frame. For each frame, a subframe can be defined such that the number of pixels that most likely contain the object is determined and stored on the SSR. A single subframe “walks” across a sequence of LEISA images. The original Y offset of this sub-frame, the direction it moves from image to image, and the pixel rate it moves from image to image can be specified, as well as the size of the subframe.

6. Communication System

The New Horizons RF telecommunication system (DeBoy et al., 2005) provides the command uplink, telemetry downlink, and essential elements of both the REX instrument and radio navigation capabilities. The system is designed to provide communications with the Earth using the Deep Space Network (DSN) in both spinning and 3-axis stabilized attitude control modes. The system (see Figure 3) consists of an antenna assembly, RF switch network, hybrid coupler, redundant TWTAs, redundant USOs, and the uplink and downlink cards located in the redundant IEMs.

The antenna system includes both a forward antenna system (aligned with the +Y axis of Figure 1) and a hemispherical low-gain antenna (LGA) mounted below the spacecraft (not shown in Figure 1) and aligned along the −Y axis. The forward antenna assembly (Schulze and Hill, 2004) consists of the +Y hemispherical-coverage LGA, and two co-aligned parabolic antennas. The 2.1-m HGA was designed to meet a requirement for a minimum of 600 bits/s downlink telemetry rate at 36 AU to return the Pluto data set. The HGA provides better than 42 dBic gain for angles within 0.3° of the +Y axis. The secondary reflector assembly consists of a medium-gain antenna (MGA) and the HGA subreflector. The MGA allows communication at larger angles between the +Y axis and Earth (up to 4°), and specifically allows commands to be received by the spacecraft at ranges up to 50 AU. The two LGAs provide communication with Earth at any attitude orientation early in the mission. The LGAs are capable of maintaining two-way communications up to distances of approximately 1 AU from Earth. Beyond that distance (reached in the spring of 2006), only the MGA and HGA can be used.

6.1. COMMAND RECEPTION AND TRACKING

The uplink card provides the command reception capability as well as a fixed down-conversion mode for the uplink radio science experiment (REX). The uplink command receiver utilizes a low-power digital design that significantly reduces the power consumption of this critical system. Previous deep space command receivers consume ~12 W of primary power; the digital receiver developed for New Horizons consumes approximately 4 W. Since both receivers are typically powered on, the total power savings is 16 W, a mission-enabling achievement for New Horizons, with its very limited total power budget of ~200 W. The uplink card also provides critical command decoding, ranging tone demodulation, X-band carrier tracking, and a regenerative ranging subsystem. New Horizons also flies non-coherent Doppler tracking and ranging developed by APL (Jensen and Bokulic, 2000), implemented largely on the uplink card and capable of providing Doppler velocity measurements of better than 0.1 mm/s throughout the mission.

Range tracking of interplanetary spacecraft is normally accomplished by sending tones phase-modulated on an RF carrier from the DSN to the spacecraft, receiving the signal with a wide-bandwidth spacecraft receiver, and retransmitting that signal back to the DSN stations. There the retransmitted signal is processed to determine two-way round trip light time and thus the distance to the spacecraft. The primary error source in this system is the received uplink noise accompanying the ranging modulation signal on the spacecraft, which is also amplified and retransmitted back to the DSN. In addition to supporting standard tone ranging, the New Horizons communication system has incorporated a regenerative ranging circuit (RRC) to limit this turnaround uplink noise. This implementation uses a delay-locked loop (DLL) that generates an onboard replica of the received ranging signal and adjusts the timing of the onboard signal to align it with what the spacecraft receives. The signal transmitted to the ground is free of wideband uplink noise, significantly reducing the primary error source contribution. This will enable New Horizons to determine spacecraft range to a precision of less than 10 m (1σ) from near-Earth ranges to beyond 50 AU, or achieve mission range measurement accuracy for integration times orders of magnitude shorter than those for sequential ranging.

6.2. ULTRA-STABLE OSCILLATOR PERFORMANCE

The New Horizons USO is a critical component of the RF telecommunication system, providing the precision reference frequency (30 MHz) for the uplink, downlink, and the radio science experiment (REX). The USO is built at APL, and is based on heritage of systems developed over the last 30 years for missions such as Mars Observer, Cassini, GRACE, and Gravity Probe B. New Horizons carries two USOs. Each USO is a pristine version of an ovenized crystal oscillator. Short-term frequency stability (Allan deviation) at 1-second and 10-second intervals is better than 3×10^{-13} and 2×10^{-13}, respectively. This stability, and the USO low-output phase noise (< −125 dBc/Hz at 100 Hz), is crucial to REX.

6.3. DOWNLINK SYSTEM PERFORMANCE

The downlink card in each IEM is the exciter for the TWTAs and encodes frame data from the spacecraft C&DH system into rate 1/6, CCSDS turbo-coded blocks. It also calculates and inserts navigation counts into the frame data to support the noncoherent Doppler tracking capability. In addition, it is used to transmit beacon tones during the hibernation cruise period. The redundant TWTAs are the high-power amplifiers for the downlink signal. The hybrid coupler connects the downlink exciter outputs and the TWTA RF inputs. This allows either TWTA to be connected to either downlink card. The two TWTA outputs are connected via the RF switch assembly to the antennas. The network allows both TWTAs to simultaneously transmit through the HGA (if there is sufficient spacecraft power), with one transmitting a right hand circular (RHC) polarized signal and the other a left hand circular (LHC) polarized signal. The DSN has the capability to receive both signals and combine them on the ground to enhance the received signal-to-noise ratio and thereby increase the data rate by approximately 1.9 times above that using a single TWTA. The resulting downlink data rate (which includes 2 db of margin in the link calculation) is shown in Figure 8.

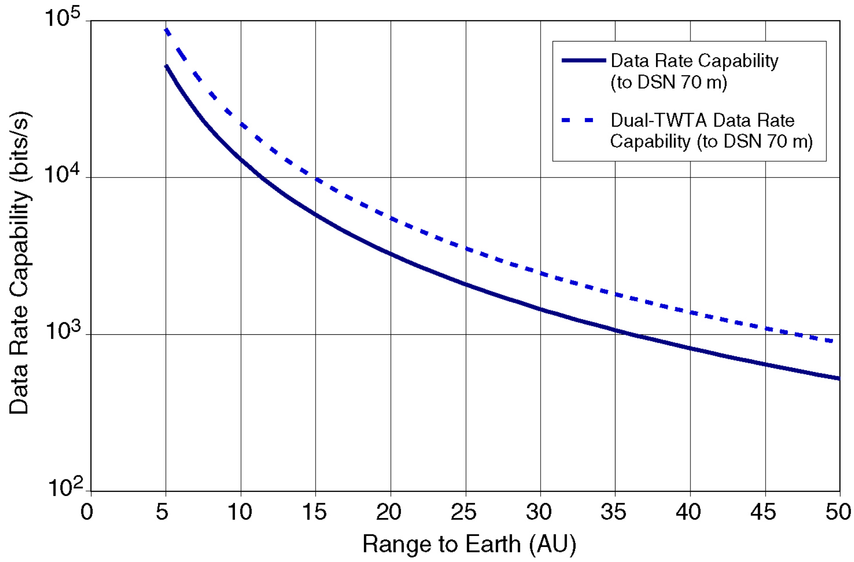

Figure 8. The RF system provides downlink telemetry above 1 kbit/s at Pluto and above 700 bits/s at 50 AU. A mode using both the redundant TWTAs at once and combining the signals on the ground increases the data rate such that over 900 bits/s can be transmitted at 50 AU.

The downlink system will guarantee that the entire Pluto data set (estimated to be 5 Gbits after compression) in 172 days with one 8-hour pass per day using the DSN 70-m antennas. If there is sufficient power, such that both TWTAs can be used, the time to downlink the data set can be reduced to less than 88 days.

7. Power System

The power system supplies and distributes power to the spacecraft subsystems and the instruments. It also provides hardware redundancy and fault protection. The key components of the power system (see Figure 3) are the general purpose heat source radioisotope thermoelectric generator (GPHS-RTG), the shunt regulator unit (SRU), the external power dissipation shunts, the PDU, and the propulsion diode box (PDB). Table III gives a functional summary of the power system.

Table III. Spacecraft power system summary. Numbers in parentheses are the spare services within the power system at launch.

RTG beginning of mission power	245.7 We
RTG minimum power at Pluto	180.0 We
Power shunt rating	312 W
Regulated bus voltage	+30V ± 1V
Switched loads	86 (12)
Thruster/latch valve drivers	56 (4)
Pulsed loads	148 (16)
Opto-isolated contact closures	8 (8)
Spacecraft configuration relays	24 (1)
Digital telemetry	1.22 kBytes
Analog telemetry	0.58 kBytes

7.1. RADIOISOTOPE THERMOELECTRIC GENERATOR PERFORMANCE

The "F8" GPHS-RTG supplied by the U. S. Department of Energy (Harmon and Bohne, 2007) is the latest in a series of RTGs of the same design supplied for NASA missions beginning in the late 1980s. The unit converts heat generated by radioactive decay of the plutonium heat source into electricity using silicon-germanium (SiGe) thermocouples (designated "unicouples" because of the close physical proximity of the “hot” and “cold” junctions of the thermocouple). The plutonium heat source is made up of 72 individual pellets of plutonium dioxide. Each pellet is encased in an iridium cladding and assembled (four to a module) into the 18 heat source modules that make up the total RTG inventory. The RTG assembly is illustrated in Figure 9. The fully fueled RTG had 132,465 curies of plutonium at launch with a half-life of 87.7 years (NASA, 2005). The thermal output of the heat modules was 3,948 W at launch (Cockfield, 2006), which is radiated to space during flight. Results of spacecraft RTG integration and early mission performance tests have been flawless, with the F8 normalized power output tracking above its contemporaries (Ottman and Hersman, 2006).

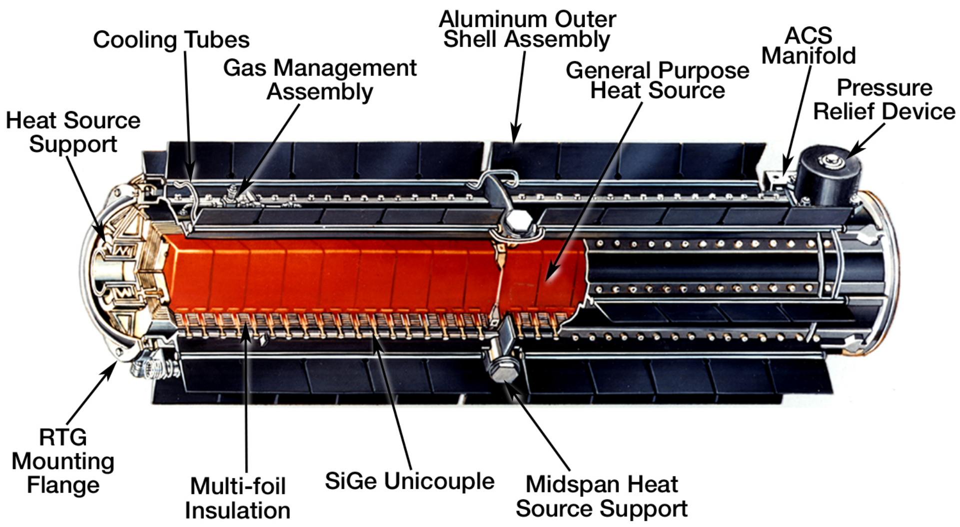

Figure 9. The GPHS-RTG has 9.75 kg of plutonium dioxide distributed in 18 GPHS modules. The modules are designed to contain the plutonium in case of a launch accident. The aluminum outer assembly radiates the heat from the modules into space. The unicouples generate electricity based on temperature differential between the modules and the outer assembly. Prior to launch, the assembly is filled with an inert gas to keep the unicouples from oxidizing. Once in space, the gas pressure relief device allows the gas to escape and the RTG then operates at maximum efficiency (Image courtesy of the Department of Energy).

7.2. SHUNT REGULATOR UNIT

The SRU is the power interface with the RTG. It is designed to maintain the bus voltage to 30 V by dissipating the excess RTG power in either external resistive shunts (waste heat is radiated to space) or internal spacecraft heaters, depending on the thermal control needs. The three SRU loop controllers use a majority-voted design to control the 16 sequential analog shunts that maintain a constant load on the RTG during spacecraft operational mode changes. Each full shunt is capable of dissipating 19.5 W, with the first two shunts steerable to internal heaters in half-power increments. The shunts were designed to be $n + 1$ redundant with a fully fueled RTG and can be individually disabled if a fault is detected. Relays within the SRU maintain the configuration of the shunts, internal heaters, or external dissipaters. The spacecraft bus voltage is set at 30.25 V, which is slightly above the maximum power transfer level of the RTG, to account for voltage drops as the power is distributed to the individual loads. The SRU also contains a 33.6-mF capacitor bank that provides for short duration current surges at load turn-on or fault condition.

7.3. POWER DISTRIBUTION UNIT AND PROPULSION DIODE BOX

The PDU contains fully redundant solid-state power switching, pulsing, and monitoring functions and hardware-based fault protection features for the spacecraft. The communication interface to the IEM is through the redundant 1553 interfaces with redundant universal asynchronous receiver/transmitter (UART) serial links for the passage of critical commands and telemetry. Loads such as the IEM, command receiver, USO, and the PDU 1553 board are configured as critical loads with both the primary and the redundant units powered on by default. Software and hardware enabling is required to turn off any of the spare units. Dual-level low-voltage sensing sheds non-critical loads if the bus voltage drops below the set points (nominally 28.5 V and 27.0 V, respectively). Additional bus protection features within the PDU also address the power-limited nature of the RTG power source. The circuit breakers have selectable levels for tight load monitoring and allow multiple attempts at power cycling the loads. Loads are fused as a final protection measure. The PDU also contains spacecraft configuration relays, sun sensor interface, sensing of load telemetry (both voltage and current) and temperature telemetry from the power system. The configuration relays provide a non-volatile means of storing spacecraft state information. They are used by the autonomy system in certain fault conditions (e.g., temporary loss of power or a switch to the redundant C&DH system) to assure that critical spacecraft state information is maintain onboard. The PDB is the power system interface between the PDU and the propulsion thrusters, latch valves, and the catalyst bed heaters.

Figure 10 shows the power for each of the nominal operating modes. The spacecraft power requirements vary as different subsystems are powered to perform various functions. The requirement is smallest during the cruise mode, when most instruments and some subsystems such as the transmitter and IMUs are powered off. Data return after the Pluto encounter is performed in a spinning mode. The downlink power assumes that only one transmitter is on. If the second TWTA is powered to increase the data rate downlink, as discussed in section 6, the spacecraft must be in a passive spin state so that other components, such as the catalyst bed heaters, can be powered off to accommodate the additional 31 W. TCM operations can be performed in several ways (see section 4.1); the minimal power mode shown here is the active spin mode. Science operations at Pluto are performed in the 3-axis control mode, with only those instruments powered that support a particular observation (the maximum is shown here). The two safe modes (Earth Safe and Sun Safe) require the propulsion system's catalyst bed heaters to be powered, thus requiring the same power as a TCM. Figure 11 illustrates the expected power provided by the RTG over the lifetime of the mission and the required power for each of these modes relative to that power. Thus, New Horizons can expect to operate with significant power margin for the Pluto encounter and adequate margin for a Kuiper Belt object encounter at up to 50 AU.

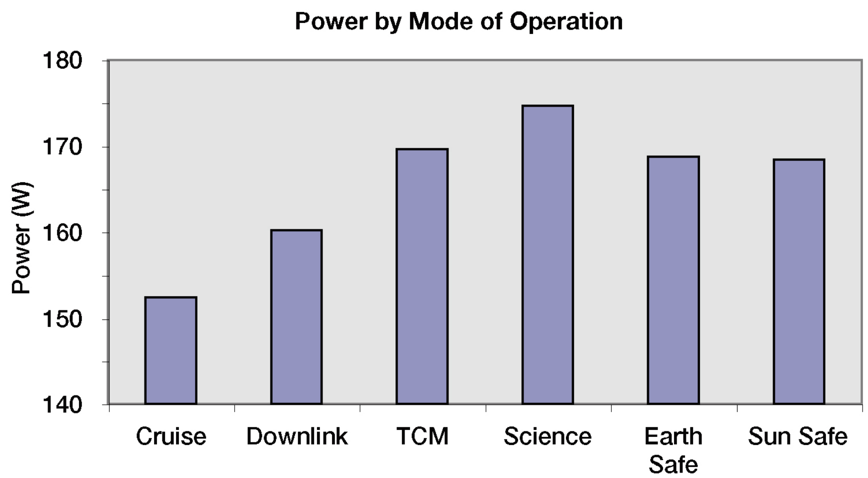

Figure 10. Spacecraft and instrument power for each of the operating states were tailored to ensure safe power margins through the primary mission.

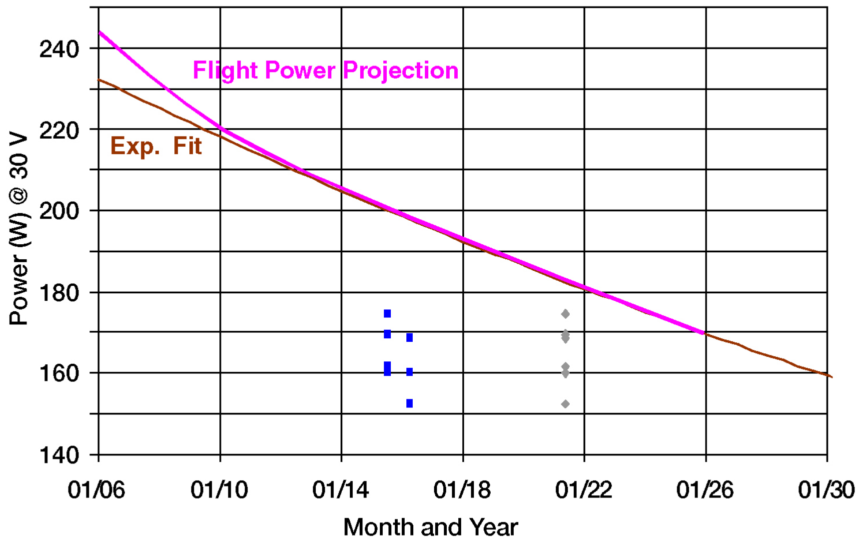

Figure 11. The RTG power output decreases with time, primarily as a result of the plutonium decay half-life of 87 years. The generator supplies sufficient power even at 50 AU (~ 2021) to support the power requirements for all necessary spacecraft states for an encounter and data playback (blue squares indicate Pluto and gray diamonds indicate a Kuiper Belt object at 50 AU).

8. Thermal Management

The New Horizons thermal design balances the power and waste heat provided by the RTG and the natural heat loss through the thermal blankets, instrument apertures, and control mechanisms to ensure each of the system elements remains within safe operating temperatures through the mission. The spacecraft subsystems are contained within the thermos bottle–like core of the spacecraft. The average internal spacecraft temperature varies from slightly under 40°C (during early operations with the lower deck facing the Sun at 1 AU) to around 20°C over the mission. Key features of the New Horizons thermal design are shown in Figure 12.

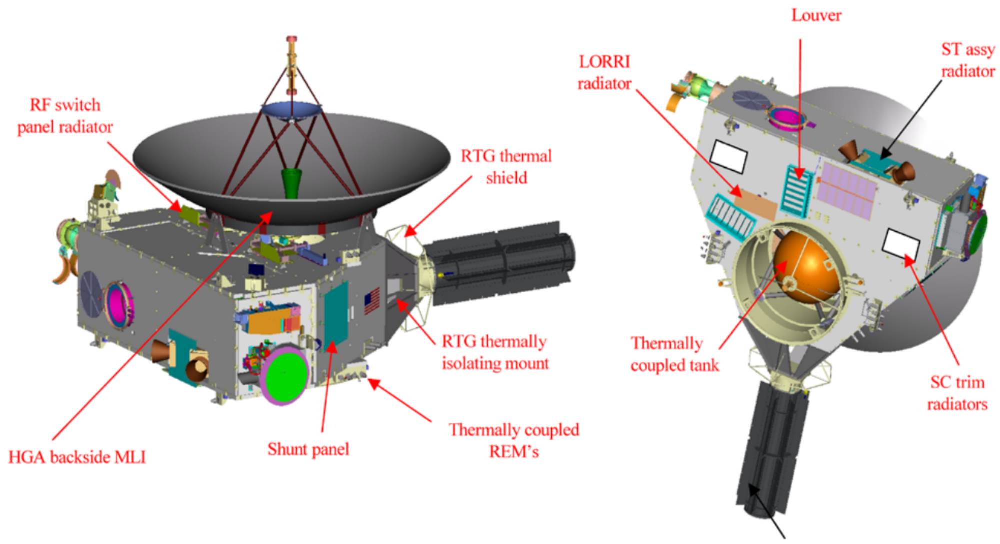

Figure 12. Key thermal design features of the New Horizons spacecraft.

The "thermos-bottle" design ties all the subsystems and components to the bulk spacecraft temperature. All the propulsion system components, including the rocket engine assemblies, are thermally tied to the spacecraft bus and are kept warm through thermal contact with the structure. The instruments are all thermally controlled based on conduction to the bus. Only the HGA and the star trackers are thermally isolated from the spacecraft bus.

The multilayer insulation blankets surrounding the spacecraft provide the dominant heat leak. The design uses the internal electronics dissipation and approximately 15 W of waste heat from the RTG to support the internal temperatures. The minimum allowable spacecraft heat leak was measured during spacecraft thermal-vacuum testing to be about 105 W with a 10°C bus temperature. Early in the mission, louvers actuated when the internal box temperatures exceeded 25°C to keep the internal temperature from getting too high. These events occurred when periods of the high internal dissipation combined with spacecraft attitudes that produced large solar loading. Since then, the bus has been running at nearly room temperature based on the onboard thermal management system.

The thermos-bottle design approach allows the spacecraft heater system to be shifted away from temperature control to control based on managing the total power level in the spacecraft. Because of the 9-hour round trip light time to Pluto, and because available electrical power is limited, it was necessary to combine the thermal and power management functions to simplify the overall design. The onboard heater control software monitors the power dissipated in the spacecraft and turns on various heaters to keep the total power close to a commanded, target value. There are 2.5-W, 5-W, 10-W, and 20-W heaters spread around the central spacecraft core. The locations of the specific heaters are less important than the total power input to the system.

Because some components, such as the TWTAs and catalyst bed heaters, do not dissipate all their power inside the spacecraft, an allowance was made in the software to apply only a fraction of those loads to the measured internal dissipation. The correct functioning of this heater control software was verified in thermal vacuum testing. During simulated mission operations, the total power dissipation goes through some profile, as the various boxes are turned on and off. The onboard heater software finds the total internal power dissipation by adding the different loads, including the reductions for those loads where the power is dissipated outside the thermal envelope. Then, heaters are turned on to make up the difference between the measured and targeted value. The total power dissipated within the spacecraft is the combination of the box dissipation and the heater power.

9. Autonomy and Fault Protection

The New Horizons mission has the longest primary mission duration to date. The primary science goal can only be achieved after a 9.5-year journey culminating in a complex set of observations requiring significant time to transmit the data to Earth. During spacecraft development, much thought and energy were devoted to fault protection. This effort continues as the operations team evaluates in-flight mission performance. The fault protection architecture uses the redundancy of the spacecraft system (shown in Figure 3) if off-nominal operation is detected. Basic elements of fault protection are resident in redundant elements of the PDU. The PDU monitors C&DH bus traffic and will automatically switch to the alternate C&DH system if it detects that nominal C&DH processor activity of the controlling system has stopped. The major elements of fault protection are implemented by software running on the controlling C&DH processor. This software is the principal component of the autonomy subsystem. The software evaluates telemetry data in real time and, based on the evaluation, takes one or more of the following actions:

1. Execute a set of commands to correct a detected fault.
2. Generate a “beacon tone” to alert operators that an event on the spacecraft requiring attention has occurred.
3. Execute one of two “go safe chain” command sets, which puts the spacecraft into either an Earth Safe or a Sun Safe state, as described in section 4 (in the event of a critical fault).

The evaluation of onboard data is performed by a set of rules that check for data that exceed defined limits for a period of time. The time period (or persistence) of the exceedance varies from rule to rule. The persistence length minimizes the chance of a rule “firing” on noisy data, or on transient data that occur during a commanded change in spacecraft pointing. Processors (other than the C&DH processor, whose activity is monitored by the PDU circuitry) are monitored via a set of “heartbeat” rules that use a telemetry point to determine if the processor is stuck at either a “one” or a “zero” state. The persistence of each of these heartbeat rules is adjusted as appropriate to match the nominal operation of the specific processor. The autonomy software can also compute dynamic limits. For example, the autonomy system monitors the propulsion system for potential propellant leaks. The system monitors the propellant as a function of both the pressure and the temperature of the propellant tank using the ideal gas law to compute a current volume and compares it to an initial value set at a previous time appropriate to the phase or mode of spacecraft operation. At the time of launch, the autonomy system used 126 rules to determine the state of health of the spacecraft.

The command sets are organized as user-defined macros and stored in memory space defined by the C&DH system. The macros can include any allowable C&DH command and can be used to power units on or off, change spacecraft modes, enable or disable autonomy rules, or execute other macros. These macros can be executed by either real-time commands or by the autonomy subsystem. The macros can also be executed by time-tagged commands, allowing the commands in the macros to execute at a specific time in the future. The autonomy subsystem used 132 macros at launch. This set has been modified as the spacecraft position along its trajectory has changed and will continue to be modified as different phases of the mission occur, system performance changes, and operational experience dictates.

The capabilities of the autonomy system are used to support a number of mission operation tasks as well as providing fault protection. For example, the “command load” sequences generated by the mission operations team are loaded into one of two memory segments. Upon the completion of one sequence, an autonomy rule is used to switch to the next sequence. The autonomy rules also check to see that an appropriate sequence has been loaded into the second memory segment, and if it has not, a rule fires causing the system to enter the “go safe” chain and point to Earth. And, as discussed in section 4.1 Attitude Control Modes, in the Encounter mode, the autonomy system will attempt to correct any onboard fault and continue the observation timeline.

10. Performance and Lifetime

For mission success, the New Horizons spacecraft must continue to function through post-Pluto data playback. As discussed above, the system design included numerous element redundancies to increase the probability of proper operation beyond that of a single-string design. A probabilistic risk assessment (PRA) was carried out to guide overall system architecture design and to support trade studies, with initial results given at the concept study review and final results several months prior to launch. While the result is not intended to be a true statistical measure of the probability of success for the New Horizons mission, setting a design goal of 85% and tracking the PRA result through the design cycle gave assurance that risk was appropriately managed. The results of the final PRA (Fretz, 2005) are shown in Figure 13 for the best- and worst-case launch dates and mission lifetimes for minimum science and full science. The bands in Figure 13 give the 90% confidence limits for each calculation, with the mean shown as the vertical line in each distribution. The mean probability of success remains above the design goal of 85% for the actual mission duration, and is acceptable for all possible mission durations based on changes in launch date.

Prior to launch, the sensitivity of the mean probability of success to changes in failure rates for individual system elements was examined, in part to understand the effect on the probability of success of unresolved anomalies from integration, test, and pre-launch activities. In addition, this sensitivity analysis studied how the probability of success estimates depend on the accuracy of the failure rates for individual system elements. The results from altering the failure rates fall well within the uncertainty range for the PRA results in Figure 13, indicating that the impact of unresolved issues or inaccurate element failure rate estimates on the overall mission reliability was negligible.

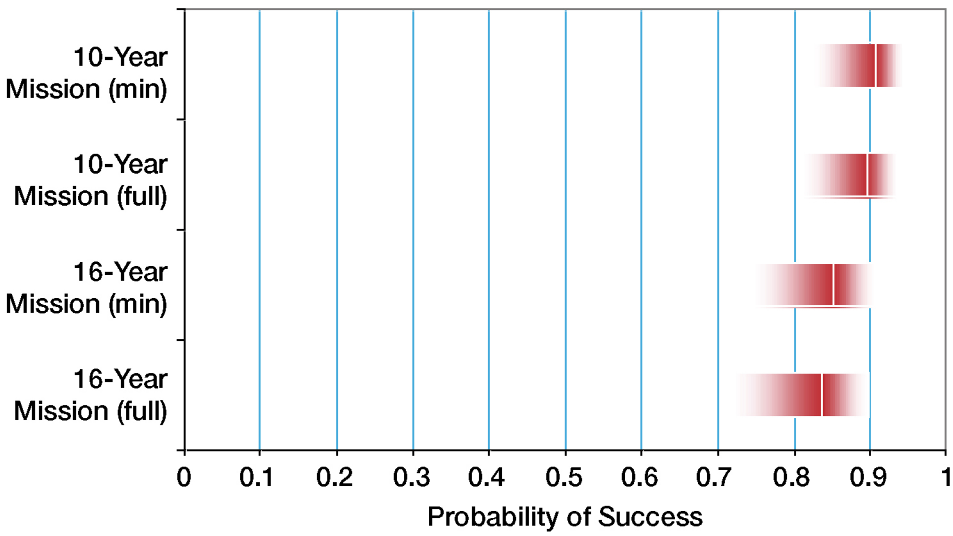

Figure 13. An analysis was performed to determine the probability of the New Horizons system design to successfully perform the mission and meet the science objectives. With the successful launch early in the 2006 launch window, the probability of meeting the full mission requirements are well above the mission requirements and the probability of mission success in an extended mission (through the Kuiper Belt to at least 50 AU) is also very high.

Management of both limited-life system elements and use of consumable resources onboard the spacecraft is essential to achieving these lifetime probabilities. Limited-life items are listed in Table IV. The limited-life items are discussed in three categories:

1. Items exercised very few times under normal operations (e.g., latch valves, RF switches, PDU configuration relays, instrument door, relay, and non-volatile memory programming cycles)
2. Items that are purely time dependent (e.g., IMU leak rate)
3. Items used regularly under normal operations that must be carefully managed (thruster valve cycles, TWTA and IMU on-time/laser intensity, and IEM and SSR EEPROM/flash memory cycles)

Table IV. New Horizons limited-life items.

Latch valve cycles
Thruster valve cycles
Telecom system RF switch cycles
PDU configuration relay cycles
Alice instrument aperture door cycles
Ralph instrument side select relay cycles
Instrument processor non-volatile memory programming cycles
TWTA filament on-time
IMU leak rate
IMU on-time/laser intensity
IEM Processor flash memory programming cycles
SSR EEPROM/flash memory erase cycles

The latch valves that supply attitude control thrusters were commanded to the open state shortly after launch and there are no plans to operate them again. The latch valves that supply ΔV thrusters are opened only for TCMs, resulting in a very small number of cycles. RF switches are operated to check out the redundant elements of the RF system. Some operations remain as the communication link uses the MGA or HGA as the mission continues, but these operations are few in comparison to the lifetime capabilities of the switches. The same is true for the operation of the PDU configuration relays, instrument door, instrument relays, and non-volatile memory programming cycles, where the operational limit exceeds the expected usage by orders of magnitude.

The IMU leak rate was measured before launch and was determined to be substantially small, such that it provided margin greater than 10 times the requirement. Information on gross leak rates is available via telemetry No change in the rate has been observed during the first year of flight.

The TWTA filaments are rated at 260,000 hours. The planned operating time through the Pluto downlink is less than 45,000 hours. Thus there is sufficient margin in the use of a single TWTA to meet mission requirements.

The IMU operating hours are limited by the operating life of the gyro lasers, due to degradation in the laser intensity. These units were carefully selected during manufacturing to ensure lifetimes for all 3-axis measurements exceeding 42,000 hours. Pre-launch estimates of the projected lifetime actually exceeded 52,000 hours. These units operate a significant amount of time during the early cruse to Jupiter and during the Jupiter encounter. After Jupiter, the spacecraft spends most of the next 7 years in the hibernation mode, during which the IMUs are off. By this means, the planned operating time on each IMU is less than 23,000 hours.

The thruster valve cycle limit of 273,000 was adopted from the limit used on the Cassini program. This limit was derived from qualification tests, which demonstrated performance beyond 409,000 cycles. New Horizons uses the same thruster as the Cassini and Voyager missions. The Voyager spacecraft, which experienced its only failure at over 500,000 cycles, provides valuable in-flight data to increase confidence in the thruster valve cycle performance. New Horizons plans to cycle each thruster valve approximately 200,000 times, well under the 273,000 cycle limit.

The New Horizons spacecraft is fully functional with significant margins for both consumables (see Figure 14) and limited-life resources to support the full mission requirements at Pluto. The initial cruise phase and Jupiter encounter provide many opportunities to fully exercise the system elements and to allow the mission operations team to demonstrate the kind of operational activities planed for the Pluto reconnaissance. With the excellent launch and the power supplied by the RTG, the spacecraft is expected to operate productively well into the Kuiper Belt, providing NASA and the Science Community the opportunity to explore this region after the 2015 Pluto rendezvous is completed.

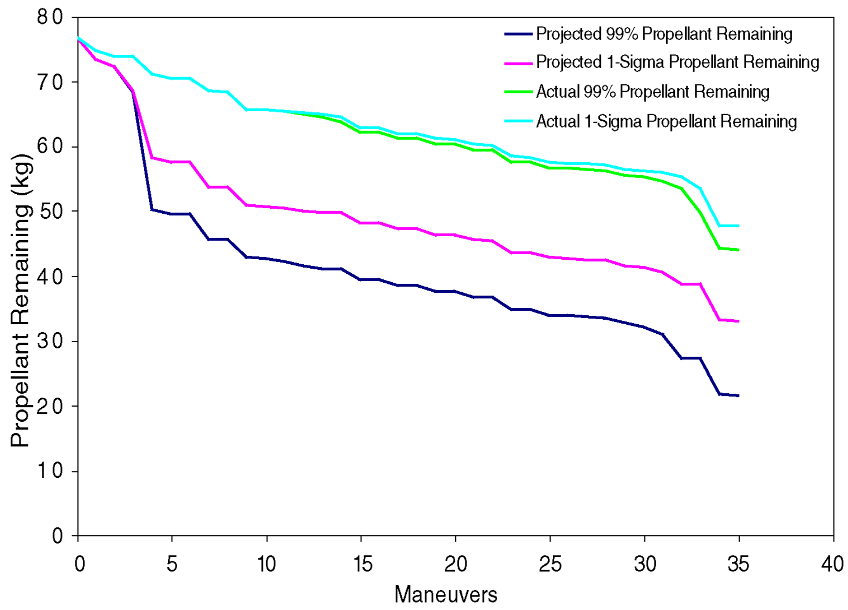

Figure 14. The hydrazine propellant is the only onboard consumable for the mission. Consumption analysis included 35 events through Pluto data return that required expenditure of hydrazine. The pre-launch prediction indicated that a good margin would remain available (> 20 kg with a 99% probability). Because of the excellent trajectory provided by the launch vehicle that number has increased to over 40 kg (Figure courtesy of Stewart Bushman).

Acknowledgments

The New Horizons spacecraft is an accomplishment that represents the work of many hundreds of individuals and many organizations. The New Horizons spacecraft development was performed at The Johns Hopkins University Applied Physics Laboratory under NASA Contract NAS5-97271, under the leadership of Dr. S. Alan Stern, the New Horizons Principal Investigator. In addition, Dr. Stamatios M. Krimigis provided essential leadership during the early phases of the program, when the mission was vulnerable to cancellation, until its value was re-enforced by the National Academy study of 2002.

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