Options for Returning Payloads from the ISS after the Termination of STS Flights

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During the upcoming post-STS (Space Shuttle) flight era, there will be a loss of capability of returning payloads from the International Space Station. A survey of means of returning such payloads is discussed, which includes the proposed large re-entry capsules currently being designed. However, these flights may be infrequent and as such, may not meet the required demand for returning smaller payloads at greater periodicity. Approaches applicable to small and mid-sized payloads will be presented, including the payload types of potential interest to the larger ISS (International Space Station) user community. Potential candidate options may include the use of propulsive or drag system designs to accomplish the de-orbit maneuver. One such system, termed Small Payload Quick Return (SPQR), will be discussed as a viable candidate system for providing near-term, on-demand sample return capability.

Nomenclature

A	=	area, m^2
C_D	=	drag coefficient, dimensionless
I_{sp}	=	specific impulse, s^{-1}
Kn	=	Knudsen number, dimensionless
M	=	Mach number, dimensionless
V_e	=	entry velocity, m/s
m	=	mass, kg
β	=	ballistic coefficient, $\frac{m}{C_D A}$, kg/m^2

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I. Introduction

THE end of the 30-year Space Transportation System (i.e., the Space Shuttle) flight program will mean a shift in the ability to use the International Space Station (ISS) as originally conceived. Gone will be the large crew rotation, the ability to send large integral cargo elements, and in particular, there will be a compromise in the ability to return payloads with the previous frequency. Service crews will ascend on more traditional rockets and Soyuz entry capsules designed originally for the Soviet lunar program. Cargo ascent capability will be augmented by resupply vehicles which will include the COTS (Commercial Orbit Transportation Service) Dragon (SpaceX) and Cygnus (Orbital Sciences Corp.). In addition, the traditional Russian Progress, as well the European ATV (Automated Transfer Vehicle) and the Japanese HTV (H-2 Transfer Vehicle) will also be available. The expected frequency with which this will occur is given in Figure 1 [1].

In terms of ISS utilization, the lack of payload return capability (or "down-mass") is of concern, and is shown in Figure 2 [1]. As can be seen, the only available down-mass capability is provided by the COTS SpaceX Dragon capsule. (There may be a European capsule at a later date.) As presently manifested, the flights are only once per year through 2011 and twice or three times per year in 2012-2015. Both biological and non-biological scientific experiments may be greatly enhanced if appropriate samples are routinely returned from the ISS. In addition, if any commercial interest is eventually to be realized, it will require not only ease of access to the ISS environment, but the equal ease of payload / sample return with an acceptable frequency.

The paper will examine some of the traditional and non-traditional means of solving the downmass problem by the timely or on-demand return of small and eventually mid-sized sample canisters.

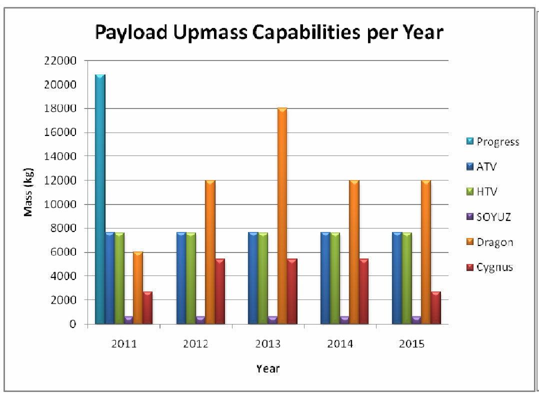

Fig. 1: Projected Future Upmass Capability

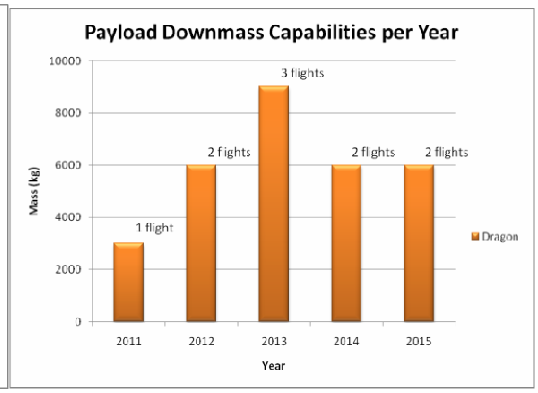

Fig. 2: Projected Future Downmass Capability

II. De-orbit Options

A variety of options will be discussed. These include "hot-gas" de-orbit systems (including the use of a supply vehicle to provide the de-orbit ΔV), "cold-gas" de-orbit systems, and non-propulsive concepts (i.e., drag concepts).

A. Hot-gas De-orbit System (e.g., Corona Program)

Since the incipience of the space age, there has been interest in de-orbiting objects from stable orbiting platforms. The first such system was the Discoverer 13, in 1960. This followed a period where systems were first intended to achieve circular (or stable) orbit, or they were part of the development of transcontinental sub-orbital systems such as ICBM (Inter-Continental Ballistic Missile) re-entry vehicles.

An example of a system de-orbited from a stable orbit, Corona, was flown in the early 1960s (Figure 3). The system was an orbiting photo-reconnaissance facility which returned film through the reentry vehicle (prior to the ability to transmit CCD telemetry data). The combined system, which included the reentry vehicle (RV) and service module, would go through a standard operation sequence (abbreviated): a) attainment of proper attitude via RCS (Reaction Control System), b) spin up (to account for thrust misalignments during the de-orbit burn), c) initiation of the de-orbit propellant burn (using a solid propellant rocket), d) separation of the RV from the Service Module, e) re-entry of the RV, f) separation of the heatshield from the payload canister via parachute deployment, g) recovery of the payload canister through an air recovery system (air-snatch via various aircraft platforms). At first, the success rate was rather dismal (initially, 8 failures). However, with time, the overall system improved such that it proved sufficiently reliable. Eventually, 102 successful flights occurred and the project was terminated in 1972 due primarily to technological advances [2].

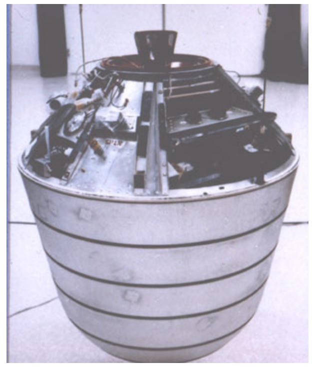

Fig. 3: Corona Style Reentry Vehicle

In application of such a system to the ISS payload return problem, the Corona Project had many attributes in terms of systems and operations that could be replicated (e.g., the Service/De-orbit Module and RV combination; the de-orbit sequence). The Corona re-entry system was large (54.4 kg with a 18.1 kg payload canister with the ballistic coefficient, β, equal to 220 kg/m^2), but the basic system could be appropriately scaled larger or smaller depending on the application. The NASA Ames BioSatellite Program and subsequent LifeSat design study were based on larger scaled versions of the Corona capsules [3]. However, there are various factors that would – in the present environment – make this system less attractive. First, the ‘hot-gas’ system would require extensive testing and safety analysis due to the inclusion of a solid rocket motor. Second, the de-orbit propulsive capability would be in the 100-200 m/s, thus making re-contact with the ISS a safety problem should the firing sequence be mis-aligned (and in addition, it is difficult to cause the solid motor to cease function without a flight termination system). Finally, the cost of developing and qualifying the system, due to the previous points, may not make the system attractive.

B. Cold-Gas ISS Payload Return System

Cognizant of the difficulties of a ‘hot-gas’ system, there was a brief study initiated at NASA Ames in 2004 to develop a ‘cold-gas’ system [4]. The system, shown in Figure 4, was intended to address the perceived need of independent return of samples from the ISS, thus anticipating the end of the STS era well in advance. The primary science requirement came from the Human Research Initiative (HRI), whereby the majority of samples were to remain frozen during the re-entry transportation sequence. This capability was thought to be critical, particularly in the event of an on-board freezer failure and resulting loss of research samples.

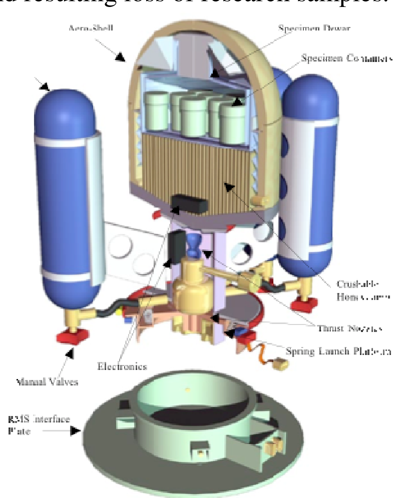

Fig. 4: Earlier ISS Cold-gas Sample Return Capsule (SRC study)

The concept was to use cold gas (compressed air) for the de-orbit system, thus circumventing an important ISS safety concern (no asphyxiation hazard or over-pressurization of the ISS should one of the cylinders malfunction). The guidance and spin-up maneuver were to be accomplished using the JEM robotic arm. At the predetermined time, the properly pointed arm would spin and release the system at the correct attitude. After approval of safety systems, the properly aligned system would then receive the command signal to undergo the de-orbit 'burn.' After this was accomplished, the re-entry capsule would separate from the de-orbit system and re-enter at the desired location. The capsule had attributes such as being self-orienting and also not requiring a parachute. A crushable material (honeycomb) would attenuate the ground or water impact. Finally, a beacon would be used to locate and extract the samples from the interior.

As can be seen, there were several advantages of this system over the 'hot-gas' example. First, the small size permitted an amount of cold-gas that was of such quantity that ISS safety due to accidental over-pressurization was a lessened concern (though it should be noted that by using helium as the propellant, the specific impulse, $I_{sp,}$ is significantly higher thus requiring markedly smaller tanks). Second, the guidance/attitude control system was eliminated by using features of the ISS. Perceived disadvantages included a) an untested, self-orienting re-entry capsule (based on the AESOP shape proposed by the Aerotherm Corp.); b) a projected 5 years and 30+ million dollars to test and implement the design. In addition, the ISS operations, including the use of the robotic arm and any related pointing errors, were never studied. It should be noted however, that if the project had been funded when originated in 2004, it is likely that some kind of workable system may have been in place by the time of the STS flight series termination. Due to the potential attraction of a cold-gas de-orbit system, this is considered again in a separate section below.

C. Drag-assisted Techniques (Rapid Orbital Decay)

As an alternative to the rocket (thrust) based re-entry concepts, the authors investigated drag-based de-orbiting systems. Prior studies have examined satellite disposal by drag methods [5-10]. Instead of using thrust ('hot' or 'cold' as in the preceding sections) to perform the de-orbit maneuver, a set of drag-devices are used to perform a rapid, carefully timed orbit decay. To illustrate, if an erectable drag device is initiated from a circular 350 km orbit (at times the ISS altitude), and the ballistic coefficient, β is 1 kg/m^2, the orbit decay to 100 km occurs within 36 hours (Figure 5). If the ballistic coefficient is lowered by a factor of 10 to 0.1 kg/m^2, the comparable orbit decay occurs in as little as 4 hours. In this free molecular flight regime, the drag coefficient, C_D, is assumed to be in the vicinity of 1-2 depending on local surface inclination. A 1 kg/m^2 drag device will require approximately 0.5 m^2 per kg of payload if the drag device is a flat plate. Thus, a 20 kg system would require a 10 m^2 drag device, with a relatively modest diameter of approximately 3.6 m.

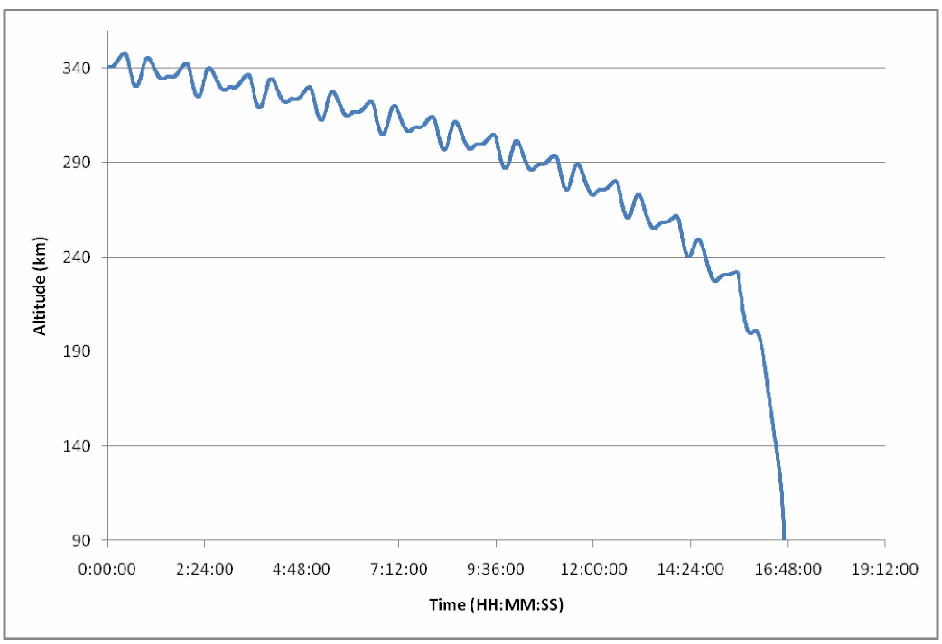

Fig. 5: Sample De-orbit with β=1 (Jacchia-Roberts Atmospheric Model)

III. SPQR (Small Payload Quick Return)

To use drag for orbit decay re-entry, distinct flight phases, or stages, were found to be required for control and targeting accuracy. The overall entry sequence is briefly outlined in Figure 6. The sequence commences with a gentle expulsion of the system from the ISS via a mechanical or manual release. For ISS safety concerns, the β of the initial ejected system should be less than that of the ISS such that orbit decay will occur rapidly enough to remove the re-contact concern. Once the SPQR system has gone through Proximity Operations and is positioned / oriented properly, a timing command is provided for the orbit decay sequence initiation:

- Stage 1: Erection of the Exo-Brake and descent from 350 to approximately 100 km.
- Stage 2: Deployment of the TDRV (Tube Deployed Re-entry System) from 100km to 10 km.
- Stage 3: Deployment of the parachute / parafoil from 10 km to ground / retrieval.

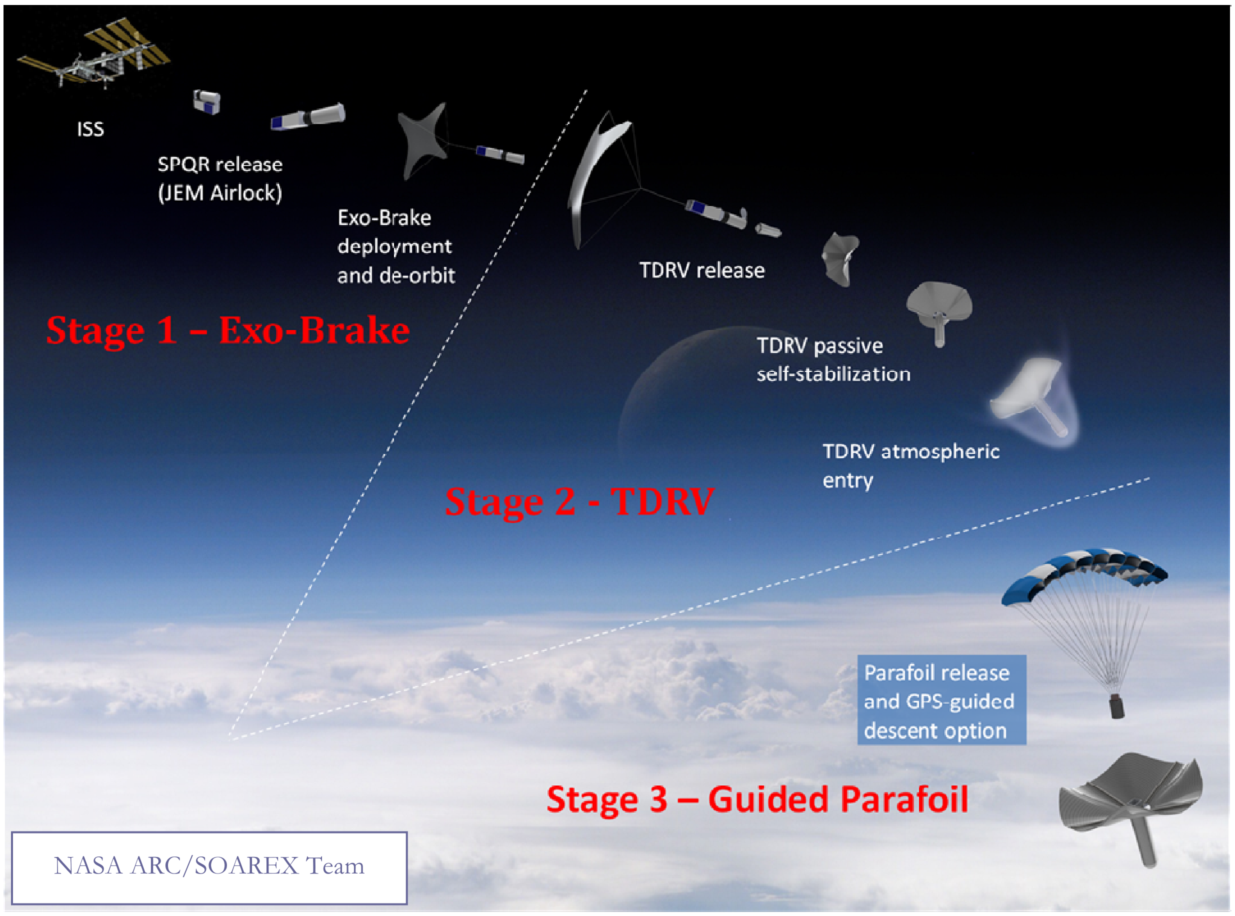

Fig. 6: SPQR Concept of Operations

After the Stage 1 segment is complete, Stage 2 commences the 'hot' atmospheric entry portion of the overall sequence. An essential feature of the TDRV re-entry system is its ability to self-stabilize, so there is no requirement for spin-stabilization imposed on the Stage 1 vehicle. Due to the low β of the TDRV (10 kg/m^2), the sonic altitude is very high, permitting early deployment of the Stage 3 parachute release if required. Stage 3 commences with the separation of the Phase Change Thermal Control Unit (PCTCU), including the parachute canister, from the TDRV. The entire sequence may be seen in Figure 7, which depicts what is essentially a plot of altitude vs. time. The three different flight regimes (based on the Knudsen number) are provided and illustrate the very different flight phases and related fluid mechanics. The TDRV release point is also essential for downrange targeting and is determined with on-board GPS on Stage 1. During Stage 3 flight, a guided parafoil may be used which could make up for additional cross-range errors.

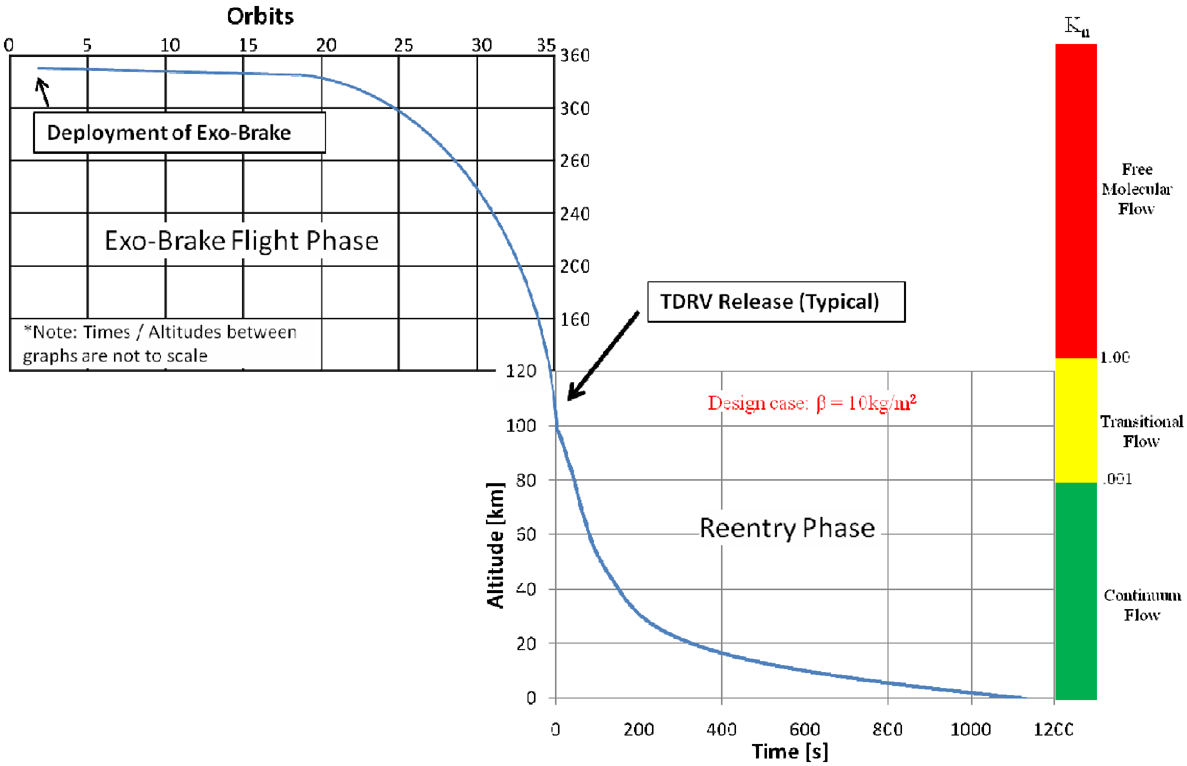

Fig. 7: Sample SPQR Altitude vs. Orbit/Time Plot

A. The Exo-Brake

The Exo-Brake is the drag device that produces the necessary drag during the Stage 1 free-molecular flight regime phase [11]. The designs considered included the exo-atmospheric parachutes [12, 13], aero-assist concepts and solar-sails [14-17]. The parachute concepts were originally used in the NASA sounding rocket program through the 70s and achieved diameters of almost 20m, with air-retrieved payloads of 120kg. Unfortunately, not only were the techniques to build and package these unique devices lost, the design is not as applicable due to uncertainty in time to deployment, low thermal limits, and complications due to shroud lines. The aeroassist concepts tend to have an inflation device used to maintain the essential shape, which introduces a single-point-failure. In addition, the requirement to maintain the erected/deployed state could be compromised by the long duty cycle (leaks), as well as any shock-impingement from the forebody. Finally, solar-sails were also examined but have the essential feature of depending on thin deployable beams to maintain the lift (or drag) surface. As the trajectory altitude decreases and dynamic pressure increases, the beams would eventually buckle prior to the deployment of the Stage 2 re-entry device.

The design concepts that emerged tend to have a combination of the more desirable attributes from the above concepts. First, a drag-device that would survive increased dynamic pressure/heating prior to the Stage 2 release would be primarily a tension structure, with a minimum number of attachment lines to ease the complication in rapid deployment. Secondly, the material and attach lines had to include selective areas to survive the higher heating, as well as possible shock-impingement. Various designs were analyzed using free-molecular flow approximation codes, which have validated the current design approach. At present, a flight experiment is being designed for use on sounding rocket platforms to validate the basic packaging, ejection/erection design and overall performance.

B. The TDRV (Tube Deployed Re-entry System)

The basic TDRV deployment sequence is depicted in Figure 8. The general topology is based on the SCRAMP (Slotted Compression RAMP) vehicle [18] that has been under development for a number of years through the SOAREX (Sub-Orbital Aerodynamic Re-entry Experiment) flight series at NASA Ames. The basic vehicle has the payload section forward of the drag device, thus producing a large static margin by having the center-of-pressure aft of the vehicle. In the TDRV variation [19], the aft-mounted drag device is comprised of a high temperature fabric matrix held in place by a series of struts. The drag device is sufficiently flexible to be stowed around the forward payload section in a cylindrical arrangement (hence, ‘tube deployed’). The large deployed drag surface permits a low ballistic coefficient (β=10 kg/m^2), which results in very high altitude for peak heating. This, in combination with the natural coning motion of the probe, diminishes the effect of the shock-interaction region on the flexible fabric matrix. Typical stagnation heating rates are of the order of 20-30 W/cm^2, depending on the entry state vector dictated by the Stage 1 release. During the SOAREX-7 flight conducted on May 28, 2009, the first TDRV was successfully deployed at an altitude of 134 km (Figure 9). Though the entry velocity was low (Ve < 2 km/s; M < 5), the probe recovered from a deliberate tumbling motion and demonstrated the self-orienting capability, stability and high drag. A video frame from the side-looking camera shows the probe stabilized at high altitude (Figure 10). Additional flights are currently being designed for the higher re-entry velocity tests to validate the aero-physics calculations.

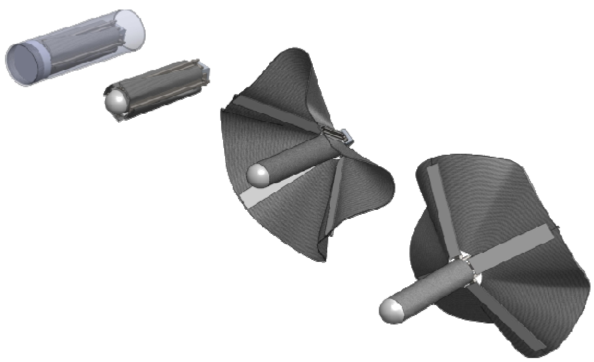

Fig. 8: TDRV Deployment

Fig. 9: TDRV Deployment 134km May 2009

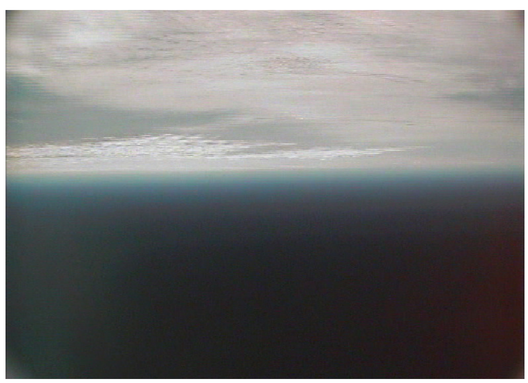

Fig. 10: TDRV Onboard Camera Showing High Altitude Stabilization

C. The SPQR Vehicle Design Concept

The basic SPQR spacecraft design consists of two 75 cm tubes which are hinged in a jack-knife arrangement to permit compatibility with a small ISS air-lock (e.g., the JEM airlock). The SPQR has the functionality of a small satellite, which includes basic power, thermal, telemetry, command/control functional elements. The satellite function is housed in one of the tubes essentially at the middle of deployed linear tube. Most of the volume consists of two empty bays which house the Exo-Brake and TDRV at opposing ends. Each of these is deployed by an electrically actuated latch which permits a spring-loaded piston to eject the particular device. Small body-mounted solar arrays provide adequate power during the de-orbit phase.

D. The SPQR/PCTCU Canister and Payload Description

The PCTCU (Figure 11), [20] canister is an integral unit which combines the ISS biological specimen containment volume with the Stage 3 parachute/recovery assembly. This assembly is what is recovered on the ground. The canister payload volume is 10.5 cm in diameter, by 31 cm in length. It is designed to accommodate a variety of payload elements including 3 CubeSats (there is a NanoRack capability that will exist on the ISS to permit simple packaging around this common standard). Thermal control during the de-orbit and re-entry phase is accomplished using a phase change material contained on one end-plate of the PCTCU. Current analyses indicates that thermal control can be maintained to <4°C throughout the Stage 1-3 operational phases [21]. It is possible that cryogenic samples may be returned by super-cooling the phase change material, or adding a LN2 feature to the basic experiment payload volume.

Fig. 11: Phase Change Thermal Control Unit integrated module (model)

E. Control and Targeting Techniques: 2-Point Control

An essential feature of the 3 stage concept is final targeting accuracy for reasons of safety and ease of recovery. To commence, a 2-point control scenario is currently being investigated to establish the initial targeting capability. This consists of the timing of two key events: first, the timing of the initiation of the Exo-brake deployment, and the rapidity to which the full drag profile is attained along the flight velocity vector; and second, the timing of the Stage 2 TDRV release.

Prior to the initiation of the SPQR flight, de-orbit calculations are made which include de-orbit initiation position and atmospheric uncertainty effects (e.g., F10.7 forcing function effects which drive the upper atmosphere density profile [22]). The intention would be to place the Stage 2 TDRV release at such a position at 120 km altitude that the ground point could be appropriately targeted. During the actual flight, GPS data from the SPQR spacecraft, as well as data from the U.S. Space Command tracking system, would permit the actual Stage 2 release point to be calculated. In essence, the first control point is a 'predictor' and the second control point is a 'corrector.' For further refinement, the Stage 3 parafoil option would make up for additional targeting errors. The other option would be to use an aircraft to perform a routine air-retrieval.

In the current development plan, a remote target location such as Kwajalein is used for the initial testing and development of capability. This has the added advantage of allaying safety concerns of accidental entry over a populated area. It should also be noted that a Stage 1-only device was studied and appeared to be feasible from an aero-physics perspective. However, the targeting errors without the staging control strategy would result in a very expensive long range air-retrieval, and thus deemed not practical.

F. Propulsive and Other De-orbit Options

In addition to the drag induced rapid orbital decay, two other SPQR variations and a growth option were examined. In both cases, the Exo-Brake development is replaced by a de-orbit maneuver either with an integral cold-gas de-orbit system or as a de-orbit piggy-back on an ISS cargo vehicle.

1. SPQR Cold-Gas Concept

In this option, the SPQR Exo-Brake is replaced by a cold-gas de-orbit module that performs the communication, attitude, control, and propulsive function (Figure 12). An existing ACS (Attitude Control System) module with significant flight heritage was examined for possible modification. This included two additional nitrogen tanks and an axial nozzle/valve system to achieve the desired 100 m/s ΔV de-orbit maneuver. For ISS application, additional safety features would be included to make it compatible with the internal volume and operations. Larger and heavier than the baseline SPQR, the system would be approximately 1.8 m in length with a mass of 60 kg. Transported in one of the cargo vehicles, it would be stowed in a modified double rack until ready for use. The system would be transported through the appropriate hatches, loaded with the PCTCU experiment canister, and processed through a standard air-lock. After ejection from the ISS, it would undergo a similar sequence of de-orbit events as described in Section A and B. The TDRV would then be ejected prior to encountering the atmosphere interface, and the remaining entry sequence would appear as the standard SPQR sequence.

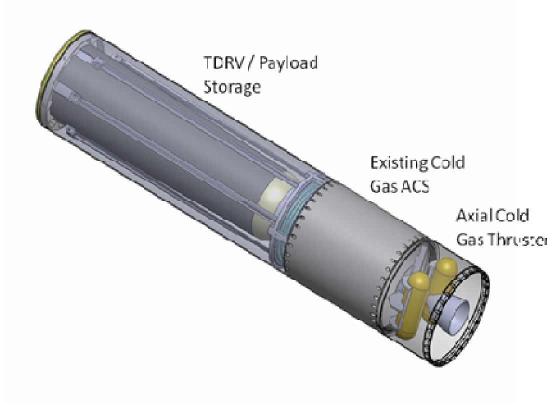

Fig. 12: Cold Gas Concept

2. SPQR Piggy-back De-orbit Concept

This option, shown in Figure 13, replaces the entire de-orbit system development (Exo-Brake or Cold-Gas) by mounting the TDRV/SPQR section to the exterior of a cargo vehicle (Cygnus, HTV, ATV). The TDRV would be loaded in the ISS interior, and in a special module transported/loaded in the simplified SPQR system. This process would require an EVA and potentially the use of a robotic arm. In this case, the simplified system would not require the telemetry or power subsystems as it would be provided by the cargo vehicle. After the cargo carrier is un-berthed from the ISS and performs the pre-determined de-orbit maneuver, the simplified SPQR element ejects the TDRV. From this point, the standard SPQR/TDRV entry and recovery sequence occurs.

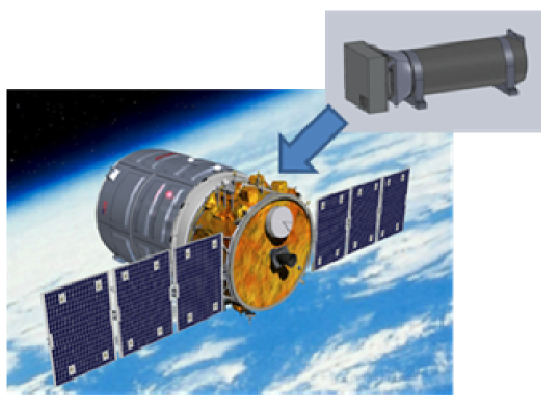

Fig. 13: Cygnus Piggy Back (courtesy Orbital Sciences Corp.)

3. Development Path and Evolution to a 75kg System

Once the 20kg SPQR is functional, a growth path to a larger system could be contemplated. In this case, a larger 2.5m TDRV-B (evolved TDRV) would be stored externally on the ISS. The previously described SPQR satellite function would reside inside a compartment on the large TDRV-B. A larger payload canister would be loaded inside the ISS, and mounted through the aft-end of the TDRV-B and locked in place. Similar to the smaller SPQR, an Exo-Brake would be deployed from the aft-end and the same 2-point target control strategy would be utilized. In this case, the entire TDRV-B would land at an appropriate landing site (land-based or ocean).

G. Discussion of Design Options

Table 1: Design Trade Space

De-orbit Method	Mass	Complexity	DDTE/ Cost	Development Time	Least Problematic	Safety Ranking
A. Hot-Gas	4	4	4	4	4	4
B. Cold-Gas	3	3	3	1	2	3
C. Piggy-back	2	2	2	2	3	2
D. Exo-Brake	1	1	1	3	1	1

A top level comparison may be made of the the various SPQR system options (Table 1). Rankings are based on a scale of 1 to 4, with 1 being most preferable. In all key parameters, the more traditional hot-gas system would be the most challenging and expensive. The cold-gas de-orbit system is heavier, and is relatively complex in terms of the number of parts. However, the development time may be the shortest due to the mature ACS module which forms the core of the device (assuming that the ISS safety issues regarding the compressed N_2 can be readily overcome). The cargo vehicle piggy-back option also has some attraction in that the de-orbit module development is averted by using the cargo vehicle itself as the de-orbit module. This path would also circumvent the development of the Exo-Brake de-orbit device and, at first, may lead to more accurate re-entry targeting. However, loading the piggy-back SPQR would require a lengthier EVA and the 'on-demand' sample return functionality would be diminished. Finally, the SPQR 'baseline' using the Exo-Brake is ranked. It appears to be the lightest, least complex and safest of the options. In this case, there are no high pressure cylinders, pyrotechnic devices, or complex EVA. However, it does require the rapid development and demonstration of the Exo-Brake. In addition, until the 2-point control is properly developed and shown to adequately overcome the density variations in the upper-atmosphere, there may an initial reduction of targeting accuracy. A practical development path may be to develop two of the approaches in tandem – with the second option being a back-up. From the above rankings, the Exo-Brake option with the cargo vehicle appears to be, at least initially, a preferred combination.

IV. Summary

- There is going to be a greatly diminished capability to return scientific or commercial payloads from the ISS during the upcoming post-STS operation phase. The development of a small (3-5 kg) to mid-sized (50kg) payload capability returned on a routine basis could greatly enhance the utility of the ISS.
- The development of both 'hot' and 'cold' gas de-orbit systems have disadvantages that include safety, development time, and cost. Any propulsive system would require extensive safety features to avoid ISS re-contact.
- An all-drag re-entry system referred to as the Small Payload Quick Return (SPQR) system is discussed which completely avoids the problem of the development of the 'De-orbit Module' feature, and avoids the issues regarding ISS re-contact. Two critical technologies which enable this capability include the Exo-Brake, and the TDRV (Tube Deployed Re-entry Vehicle). Both are under development at the NASA Ames Research Center.
- The Exo-Brake is a free-molecular flow, erectable drag device which permits rapid orbit decay from the ISS altitude to the Stage-2 TDRV deployment. The design is unique, though inspired by previous experience in the sounding rocket program, as well as aero-pass re-entry studies.
- The TDRV is a low-ballistic coefficient, self-orienting re-entry system used for the 'hot' portion of the atmospheric entry. A previous test flight (SOAREX-7; May 2009) indicated high stability and rapid recovery from a tumble. The low ballistic coefficient permits the point of maximum convective heating to be at very high altitude, thus lowering the maximum convective heating rate.
- Guidance and precision are initially attained by two-point control. This includes a) rapid erection of the Exo-Brake at a predetermined orbit location, and b) the point at which the Stage-2 TDRV system is released. Both of these are assisted by position and altitude information provided by an on-board GPS. The trial target location is the Kwajalein atoll to enhance initial safety during development. Techniques for greater control authority (particularly for overcoming the F10.7 atmospheric forcing variable) are also discussed for further evolution of the concept.

- Due to the overall simplicity of the drag system, and the relative ease and safety with which critical system elements may be tested, the SPQR system may be functional within 2 years. Initial studies indicate that the internal canister temperature can be maintained to <4° C during the course of the overall re-entry period. Particular advantages over drag devices are developed, and also contribute to a lower cost system.
- Two other options also are shown to have some attraction. These include a) a cold-gas de-orbit system based on an existing ACS assembly, and b) using one of the COTS or other transport vehicles as the 'de-orbit module.'

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