68th International Astronautical Congress (IAC), Adelaide, Australia, 25-29 September 2017.

IAC-17-B3.4-B6.5.2

ISS CONTROLLED DEORBIT: CHALLENGES AND SOLUTIONS

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Abstract

In 2011, NASA and Roscosmos established a bilateral working group tasked to develop a comprehensive strategy for the safe deorbit of the International Space Station (*ISS*), the largest man-made space object in history with an assembled mass over 400 metric tons. Currently the *ISS* Partner community has agreed to extend *ISS* operations until at least 2024. Once the *ISS* operations are complete the *ISS* has to be safely deorbited into a safe unpopulated area, which now is assumed to be South Pacific Ocean Uninhabited Area. The history of Human Space Exploration has many examples of space station deorbits. The most recent is the deorbit of the 130 metric ton Russian *Mir* Space Station in 2001. Currently the primary source for performing the deorbit operations is considered to be a propulsion system of the Russian *Progress* Cargo vehicle, using Service Module *Zvezda* propulsion system for back up. Even though *Progress* vehicles are very reliable and currently used for conducting a majority of the *ISS* propulsive events, those vehicles do not have an optimal propulsive capability to support this complex task. This paper discusses nominal and contingency *ISS* deorbit scenarios and provides quantitative assessments of propellant requirements and debris footprint for each of the cases. The proposed approaches are also compared with the strategies previously used to deorbit large man-made space objects.
Keywords: *ISS*, deorbit, footprint, contingency

Acronyms/Abbreviations

Low Earth Orbit (LEO), National Aeronautics and Space Administration (NASA), Compton Gamma Ray Observatory (CGRO), South Pacific Ocean Uninhabited Area (SPOUA), International Space Station (ISS), Rocket Space Corporation Energia (RSC-E), Multipurpose Laboratory Module (MLM), United States Orbital Segment (USOS), Russian On-Orbit Segment (RSOS), Ballistic Number (BN), Control Moment Gyros (CMGs), Momentum management (MM), Guidance, Navigation&Control (GN&C), Service Module (SM), Motion Control Group (MCG), Thermal Control Systems (TCS), Command & Data Handling (C&DH), Electrical Power Systems (EPS), Communication and Tracking (C&T), Mission Control Center - Moscow (MCC-M), Rendezvous and Docking (R&D), Common Propulsion System (CPS), Functional Cargo Block (FGB), Docking Compartment 1 (DC1), Mini Research Module 2 (MRM2), Science Power Module (SPM), Mini Research Module 1 (MRM1), Orbital Debris Program Office (ODPO), Visiting Vehicle (VV), Local Vertical, Local Horizontal (LVLH), Torque Equilibrium Attitude (TEA), Micro Meteoroid Orbital Debris (MMOD).

1. Introduction

Since the launch of the first man made satellite into Low Earth Orbit (LEO), thousands of objects have re-entered Earth's atmosphere. Most of those objects burn up during deorbit but some have components that are capable of surviving re-entry and reaching the ground. National Aeronautics and Space Administration (NASA) requires the probability of ground causality from surviving debris for all objects launched in Space to be less than one in ten thousand [1].

Two recent large scale objects controlled re-entry examples include the *Compton Gamma Ray Observatory* (CGRO) and Russian Space Station *Mir*.

CGRO was safely deorbited on June 4, 2000. The CGRO deorbit targeted 41 km final perigee and a 5,000 km in-track footprint. CGRO had a mass of ~17,000 kg, and was observed to begin break-up at approximately 72 km.

The Russian *Mir* Space Station, which was a total of ~130,000 kg when deorbited on March 23, 2001, planned for a maximum target final perigee of 80 km and in-track footprint of no greater than 6,000 km. A real-time decision to extend the final impulse from a *Progress* vehicle resulted in an estimated ΔV of 28 m/s, an estimated target final perigee of 63 km, and actual debris footprint of ~1,500 km in-track. *Mir* was observed to begin aero thermal break-up at approximately 70-77 km (individual module separation estimated at 77 km, massive aero thermal breakup at ~70 km) [2].

Russian *Mir* Space Station so far is the largest man-made object that successfully completed the controlled deorbit into the South Pacific Ocean Uninhabited Area (SPOUA). Detailed analysis was performed prior to the deorbit to assess *Mir* Space Station controllability at low altitudes, fuel and thrust requirements as well as system health prior to initiating deorbit operations. Propellant limitations made it very challenging for the team to find scenarios that would result in safe deorbit in nominal and contingency cases [3].

A lot of that experience was used to develop a strategy for the International Space Station (*ISS*) nominal and contingency deorbit operations. This paper presents scenarios for *ISS* deorbit operations and proposes *ISS* deorbit strategies. This paper also defines the assumptions and forecast developed by NASA and Rocket Space Corporation Energia (RSC-E) for preparing and executing nominal and contingency *ISS* deorbit operations.

The primary deorbit strategy implemented three burn approach similar to the one used during *Mir* Space Station deorbit, when three burns were executed over the period of four orbits. First two burns shaped a target orbit with required perigee and the final burn provided *Mir* Space Station deorbit into the SPOUA [3].

Currently, low thrust capability remains to be the main constraint for the *ISS* deorbit. Similar to *Mir* Space Station deorbit, *Progress* vehicle docked to SM aft is considered to be the main source providing final deorbit burn. However, comparatively to *Mir* Space Station, *ISS* is almost three times heavier. This leads to extension of portion of the orbit at which final burn can be executed and significantly reduces burn efficiency due to gravitational losses, which increase in ΔV requirement. If for *Mir* Space Station deorbit case, low propellant availability was considered to be a limiting factor, low thrust capability of all available thruster systems is considered to be a main constraint for the *ISS* deorbit.

2. *ISS* deorbit configuration

At the time of deorbit the total mass of the *ISS* including just not launched Multipurpose Laboratory Module (MLM) is assumed to range between approximately 420,000 kg to 450,000 kg and consists from United States Orbital Segment (USOS) and Russian On-Orbit Segment (RSOS). The current mass of the *ISS* without MLM is approximately 400,000 kg, depending on visiting vehicle configuration. Contingency deorbit scenario mass estimates for the *ISS* will be dependent on the delivery schedule of the MLM as well as visiting vehicle configuration.

Due to the large size of the USOS solar arrays, solar array configuration has a very large effect on the possible frontal area of the *ISS*, producing high drag in some instance and aiding the aerodynamic trimming of the *ISS* in others. In the nominal *ISS* deorbit scenario, the *ISS* solar arrays are planned to be biased to increasing the drag of the *ISS* where possible to aid natural orbital decay. This high drag solar array biasing is planned for the time period between the start of *ISS* deorbit operations at approximately 400 km and the start of final *ISS* deorbit operations, tentatively at 279 km altitude. Orienting the solar arrays into a 'high drag' mode comes at the expense of power generation at some beta regimes, but will aid the orbital decay of the *ISS*, saving propellant for the final set of deorbit burns.

Once the *ISS* has decayed to an altitude that final deorbit operations can begin, the *ISS* Program intends to keep the *ISS* aerodynamically trimmed to achieve maximum Ballistic Number (BN) when possible. The benefits of aerodynamic trimming include lower torques which could affect attitude control, lower loads from natural atmospheric drag as the *ISS* enters a comparatively thicker atmosphere (and thus a lower point of rupture), and fewer ballistic uncertainties in the critical final operations.

The *ISS* Program has identified a candidate trim configuration for the *ISS*, which will be used for lower altitude operations to ensure attitude control authority is available at the lowest possible altitudes. For the purposes of this study, existing *ISS* flight configurations were used to determine a weighted average BN across all beta angles for the minimum and maximum *ISS* mass values.

These values are:

- *ISS* Maximum Mass; Nominal Array Biasing: 113.4 kg/m^2
- *ISS* Minimum Mass; Nominal Array Biasing: 95.9 kg/m^2
- *ISS* Maximum Mass; Increased Drag Array Biasing: 105.1 kg/m^2
- *ISS* Minimum Mass; Increased Drag Array Biasing: 88.8 kg/m^2

All controlled *ISS* deorbit plans require the *ISS* be able to control attitude to accomplish successful impulse burns, either via Control Moment Gyros (CMGs) or on-board thrusters. The altitude at which the US Momentum management (MM) function will saturate under only aerodynamic loads in a circular orbit is not precisely known, but the NASA Guidance, Navigation,

& Control (GN&C) team has performed Monte Carlo / Sensitivity analysis for non-propulsive attitude control. This analysis indicates that for 3-sigma max atmospheric/density conditions the ±*XVV – ZLV* MM controllers could accommodate non-propulsive attitude control (keep momentum below 3-CMG capacity) for circular altitudes as low as 250 km. For nominal atmospheric/density conditions, the ±*XVV* MM controllers could accommodate non-propulsive attitude control (keep momentum below 3-CMG capacity) for circular altitudes as low as 232 km [4].

Once the *ISS* perigee drops below the lower altitude limits of the CMG MM controllers, propulsive attitude control at 100 kg/day has been assumed. This estimate presumes that a radially docked *Progress* vehicle or the MLM module will be available to provide efficient roll control. Without a nadir *Progress* or MLM, the propellant consumption is estimated to be 400 kg/day once propulsive attitude control begins, so propulsive control should be deferred as long as possible. Under the assumption of natural atmospheric drag-dominated decay from 270 km (maintaining a circular orbit), the momentum manager could be used for a longer period because the density variations over the orbit would be more manageable than an eccentric orbit whose perigee drops below MM altitude limitations.

RSC-E analysis indicates RSOS propulsive attitude control systems are capable of maintaining controllability of the *ISS* down to approximately 100 km altitude with *ISS* solar arrays trimmed to provide low torques [5]. Assuming the solar arrays cannot be trimmed in such a way, RSC-E analysis indicates RSOS propulsive attitude control systems will be able to maintain *ISS* controllability down to approximately 120 km with solar arrays perpendicular to the velocity vector [5]. This is supported by the 2008 Aerospace Corporation Controllability Assessment indicating aero torques will reach 50% control authority at 96.6 km while in an ideally trimmed *ISS* configuration, and 100% control authority while in a trim attitude at 92.7 km.

Service Module (SM) Motion Control Group (MCG) and GN&C are two systems that are critical for ensuring successful deorbit. The Russian attitude control system will be necessary to maintain attitude control during deorbit operations in the high density atmosphere near penultimate and final orbit. Combined US and Russian Attitude Determination System are necessary to ensure attitude control and accurate pointing for *ISS* deorbit burn operations. Other systems necessary to support *ISS* deorbit operations (primarily in support of GN&C include Thermal Control Systems (TCS), Command & Data Handling (C&DH), Electrical Power Systems (EPS) and Communication and Tracking (C&T).

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3. Burn configurations and engine thrust

It is expected that the final deorbit burns of the *ISS* will be executed by Mission Control Center - Moscow (MCC-M). Available thrusters for the final *ISS* deorbit burn are summarized in Table 1 and the following section.

Table 1: *ISS* engine data

	SM		*Progress-MC*
Main Engines			
Quantity	2		1
Thrust (kgf) each	315 ± 12		300 ± 30
Specific Impulse (sec)	293.7 ± 3		302 +5/-7
Engine on times (sec)	10 to 400		0.5 to 400
Attitude Control Thrusters	All but Roll	Roll	All
Quantity	24	8	28
Thrust (kgf) each	13.3 ± 0.6	13.3 ± 0.6	13.3 +3/-3.5
Specific Impulse (sec)	285 +5/-2	250 ± 5	285 +5/-2
Engine on times (sec)	0.03 to 600	0.3 to 600	0.03 to 600*

** Some thrusters are qualified to fire from 0.03 to 2,000 seconds up to 6 times.*

At the time of deorbit possible engine configurations may be different depending on *Progress* vehicle locations and canting.

In the majority of the analyzed *ISS* deorbit scenarios, at least one *Progress* vehicle is assumed to be docked to *ISS*, either at SM Aft, allowing the main *Progress* engine to provide the necessary impulse for final deorbit burns, or at a radial port, allowing *Progress* Rendezvous and Docking (R&D) engines to provide additional impulse on top of the main engine burn. The *Progress* main engine can provide 300 kgf for up to 880 seconds, using 1 kg/sec of propellant. This engine is fed from the *Progress* Common Propulsion System (CPS). Since the Progress CPS system cannot be refueled following docking to the *ISS*, *Progress* CPS propellant is planned to be used as sparingly as practical prior to the final deorbit burn in order to maximize the impulse from this main engine during the final burn.

In addition to the main engine, an aft docked *Progress* vehicle has 8 R&D engines pointed aft which can provide a total of ~100 kgf for up to 2,000 seconds, using 0.358 kg/sec of propellant. These engines are planned to be fed from the Functional Cargo Block (FGB) propellant tanks. Due to their larger propellant reserve (fed from FGB tanks) and longer engine on times limitations, these engines are used for set-up burns, attitude control, and as an additional source of impulse during the final deorbit burns.

For *ISS* deorbit cases with a radially docked *Progress*, either at Docking Compartment 1 (DC1)/MLM/RSOS Node or Mini Research Module 2 (MRM2), several *Progress* aft pointed mid-ring engines can be used to provide additional deorbit impulse. There are slight differences in expected engine thrust in the aft direction due to clocking differences between the DC1/MLM/RSOS Node & MRM2 ports.

A *Progress* vehicle docked on the zenith MRM2 port or the nadir DC1/RSOS Node ports will have 4 mid-ring engines pointed directly aft, providing ~53 kgf for up to 2,000 seconds, using 0.1675 kg/sec of propellant.

A *Progress* vehicle docked on the nadir MLM port will have 8 mid-ring engines pointed approximately 45 degrees between aft and starboard / port, providing ~75 kgf for up to 2,000 seconds, using 0.335 kg/sec of propellant. Docking a *Progress* vehicle on this port is only valid for a short timeframe post MLM arrival and prior to RSOS Science Power Module (SPM) docking. Clocking for a *Progress* vehicle docked on the nadir Mini Research Module 1 (MRM1), port is similar (40 deg vs. 45 deg). These engines are planned to be fed from the *Progress* propellant system as well as FGB propellant tanks. During deorbit operations these Progress vehicles will be used to shape the target orbit ensuring the required perigee is achieved at the time of the final burn.

The SM Aft main engines are currently operational and capable of providing a significant amount of ΔV to the *ISS*. While a *Progress* vehicle docked to the aft SM port would prevent firing of the SM main engines in a nominal scenario, *ISS* deorbit requires the maximum impulse from all available sources. Capability to execute an SM main engine burn while a *Progress* is docked on the SM aft port is being explored, and is expected to require software updates.

The SM main engines are capable of providing ~603 kgf along the longitudinal axis of the ISS for up to 340 seconds, using ~2.1 kg/sec of propellant. Because of the clocking of these SM main engines, there is not expected to be any loss of efficiency from this thrust due to a *Progress* docked to SM aft. This engine is fed from the SM propellant system, which can be refilled from FGB propellant tanks in a process that takes ~24 hours.

Fig. 1 provides one of the possible engine configurations that may be available for nominal and contingency *ISS* deorbit scenarios.

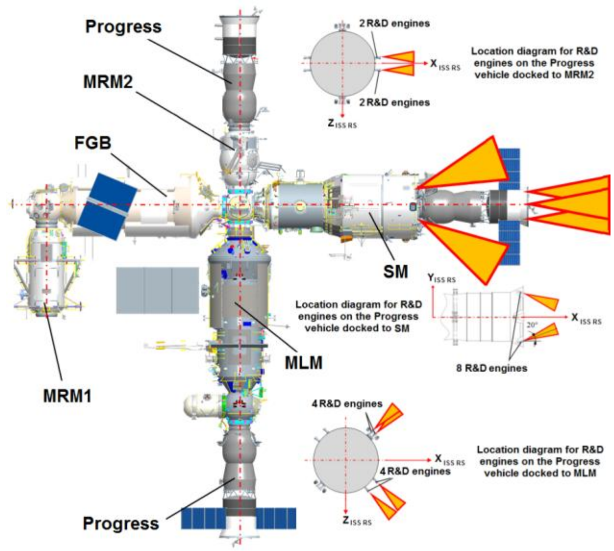

Fig.1: *ISS* engine configuration with 3 *Progress* vehicles docked to SM, MRM2 and RSOS Node

4. Footprint

The size of the debris footprint of a deorbiting structure is a function of its mass, ballistics properties, and final perigee. The lower the target perigee of the final orbit, the smaller the debris footprint. The estimated debris footprint size of the *ISS* has been calculated in a 2008 Aerospace Corporation study as a function of expected break-up altitude and capture perigee. Fig. 2 illustrates these results.

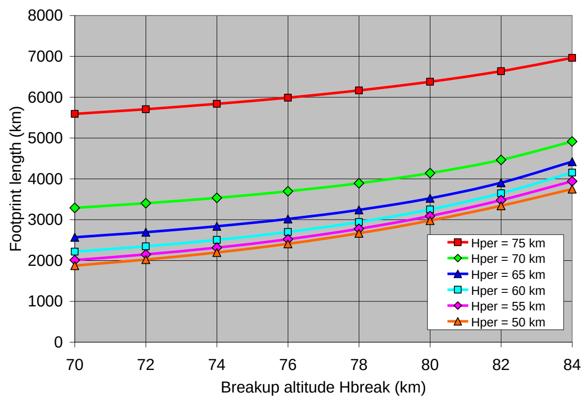

Fig.2: *ISS* debris footprint as function of breakup altitude & capture perigee

Nominal planning aims for full deorbit of all *ISS* debris into a 6,000 km or less footprint within the SPOUA. The US deorbit planning team assumes a breakup altitude of 84 km for US components, thus according to Fig. 2, achieving a 6,000 km or smaller footprint requires target perigee $H\pi$ no higher than ~72 km.

Based on the behavior observed during *Mir* re-entry, breakup of *ISS* is expected to be a multi-stage sequence. Three debris generating events were identified during *Mir* atmospheric re-entry [2]:

1. Solar array and radiator separation (Solar Array Debris)
2. Intact module rupture and separation (Rupture Debris)
3. Individual module fragmentation (Primary Debris)

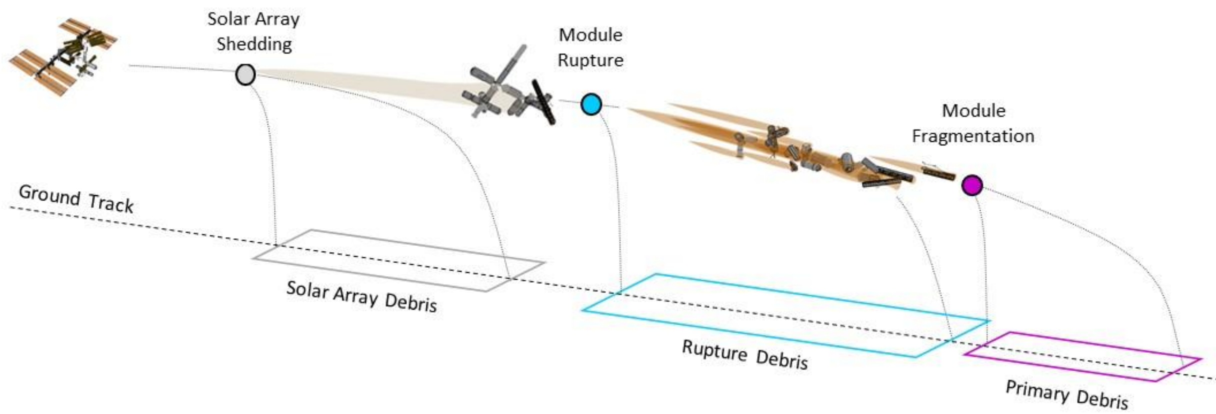

Fig.3: Expected debris generating events & impact on total footprint, based on *Mir* experience

The size estimates for each of the debris footprints generated uses representative objects (heel, toe, left and right) to appropriately ‘bound’ the debris area. The mass / area ratio assumed by the NASA Orbital Debris Program Office (ODPO) for representative objects of each footprint section, as well as expected nominal breakup altitude for each representative object is in Table 2.

Table 2: NASA OPDO footprint modeling assumptions

	Primary Footprint	Rupture Debris Footprint	Solar Array Footprint
Toe Piece Mass to Area Ratio	2000 kg/m^2	2000 kg/m^2	150 kg/m^2
Left/Right Piece Mass to Area Ratio	150 kg/m^2	150 kg/m^2	50 kg/m^2
Heel Piece Mass to Area Ratio	2.5 kg/m^2	2.5 kg/m^2	2.5 kg/m^2
Nominal Breakup Altitude	70 km	105 km	110 km

The operations team must maintain very precise control over the final trajectory’s phasing and timing in order to match the desired ground track. Based on the approach presented in [5], rough calculations for the impact of the final burn (ΔV) and required propellant for several deorbit scenarios and available thrust capabilities to ensure a 6,000 km debris footprint (0.15 orbit) are presented in Table 3.

Table 3 [5]: Correlation of ΔV and propellant

Opt	Engines	Thrust (kgf)	Fuel (kg)	ΔV (m/s)	Time	H perigee (km)	H apogee (km)	Backup orbits	Backup on next day
1	SM	603	800	4.5	340 s	119	145	1	-
2	1 *Progress* + SM	400-603	1,856	12.0	850-340 s	131	157	3	-
3	2 *Progress*	453–153	1,945	12.3	850-1,150 s	131	157	3	-
4	2 *Progress* + SM	453-657	2,198	13.3	850-340 s	131	157	3	-
5	3 *Progress*	528–228	2,678	15.6	850-1,150 s	153	186	5	3
6	3 *Progress* + SM	228–528-603	3,378	20.1	1,150-850-340 s	156	192	8	3

The table 3 shows that between 800 and 3,400 kg of propellant will be required based on the selected scenario. If the scenario with SM only is selected the low perigee allows only one attempt for the final burn. It is clear that this option can be addressed only hypothetically. Maximum number of attempts can be achieved if the option 6 is selected using three *Progress* vehicles and SM. This option allows for 11 back up orbits and the second day allows attempts on three orbits. The same approach can be used in option 5, but using only three *Progress* vehicles without SM and expending 600 kg of propellant less. Backup deorbit on next day was a requirement for *Mir* Space Station deorbit [3]. Options 2 and 3 using a single *Progress* vehicle and SM or two *Progress* vehicles became most beneficial requiring only 1,900 kg of propellant for final burn if ability to deorbit on the next day is not a requirement.

5. Nominal *ISS* deorbit plan

This section outlines the altitude strategy and propellant consumption assumption for the nominal *ISS* deorbit plan. Under this scenario the *ISS* has reached its End of Life, thus this scenario assumes all major *ISS* and visiting vehicle capabilities are intact at the commencement of deorbit operations. Nominal *ISS* deorbit is currently expected to occur in 2024, and requires a Joint International Agreement between all *ISS* partner countries on the end date and final disposal plan of the *ISS*. Subsequent planning of deorbit operations and events will take place over the course of several months, and will require numerous dynamic operations. Critical altitudes to which the *ISS* deorbit planning team is working are reflected below:

279 km "Point of no return" altitude for *ISS*. At this altitude, propellant delivery will not be able to counteract natural orbital decay.

270 km Phase repeat orbit (4 day ground track repeat). Natural orbital decay for *ISS* from this altitude expected to be ~1 month (TBC).

198 km Phase repeat orbit (1 day ground track repeat).

141 km Perigee at which apogee decays at 2 km/orbit: we do not plan more than two orbits with perigees below this value.

130 km Estimated minimum operational altitude of USOS external avionics and systems.

110-120 km Altitude at which US solar arrays and radiators are expected to separate, based on observed *Mir* re-entry.

100 km Maximum expected module rupture altitude (TBC). Minimum expected attitude control altitude via operational RSOS [1].

84 km Minimum expected module rupture altitude.

75 km Aerospace Corporation's 2008 assessment of the maximum allowable vacuum perigee that will cause *ISS* fragments to lie within a 6,000 km footprint (TBC with further analysis: footprint length is dependent upon the altitude at which the *ISS* ruptures).

50 km Guaranteed capture. 50 km vacuum perigee is the NASA Std 8719.14 requirement for re-entry targeting in all new NASA programs [1]. The *ISS* was authorized before this standard was implemented, and can therefore be exempted, as long as the deorbit is shown to be safe.

6. Final orbit set-up

Due to the propellant storage limitations of the *ISS*, all *ISS* deorbit scenarios require the assistance of natural orbital decay from atmospheric drag for varying periods of time during the deorbit sequence. The *ISS* currently operates at an average altitude of approximately 400 km. In the interest of providing the most efficient use of propellant for deorbit operations and maximize propellant available for the final set of burns, the nominal deorbit scenario plans for natural orbit decay from the *ISS* operational altitude down to approximately 279 km, with any necessary vehicle phasing burns designed to lower the altitude of the *ISS*. Due to the current altitude limitations of crewed visiting vehicles, final crew departure is expected to take place during this timeframe, prior to *ISS* reaching 333 km.

The *ISS* Program considers 279 km the point of no return, where resumption of propellant supplies and reboosts would not be able to balance out the natural altitude decay due to atmospheric drag. Depending upon upcoming visiting vehicle schedule and capabilities, it would be appropriate at or near this threshold altitude to commit remaining propellant on board the *ISS* for the final set of deorbit burns.

Without considering additional acceleration from any deorbit burns, the time *ISS* takes to decay from its operational altitude to the 279 km can vary widely depending on the assumed *ISS* ballistic configuration and the atmospheric density, which is a function of solar cycle. Taking a range of both factors into account, and assuming start of *ISS* deorbit operations in Jan 2024, Fig. 4 illustrates possible orbital lifetimes of the *ISS* as a function of BN and solar activity.

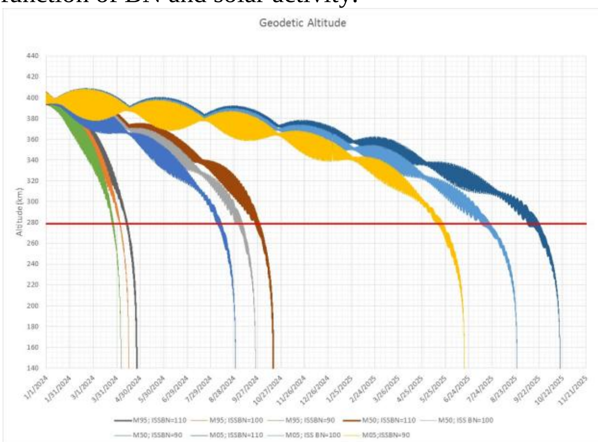

Fig. 4: *ISS* orbital lifetime estimates

In order to accurately predict orbital lifetime across the entire altitude regime from *ISS* operational altitudes to deorbit starting altitudes, this analysis does not assume any deorbit burns are performed. Note that the actual deorbit profile of the *ISS* is expected to include multiple deboost burns during this time period to accelerate decay into the atmosphere as soon as possible. As can be seen in the previous figure, the time it takes *ISS* to naturally decay from its operational altitude to 270 km can range from 3 months assuming low *ISS* BN and high solar activity up to 17 months assuming high *ISS* BN and low solar activity. Also noteworthy is that if *ISS* deorbit is planned to occur during a period of low solar activity, deorbit operations must start 1-2 years prior to the targeted re-entry date. The *ISS* deorbit planning team recommends that 'high drag mode' solar array operations be implemented when possible during this period of natural orbital decay. This is achieved by biasing the solar arrays to be as normal to the velocity vector of the *ISS* as possible while still providing sufficient power generation, increasing drag on the vehicle. Deorbit vehicle delivery planning should begin well in advance of the expected nominal deorbit due to the time required to naturally decay (depending on solar activity) to the deorbit altitude and *Progress* delivery schedule. The starting date of this analysis is also another determining factor in orbital lifetime, as solar cycle activity is directly dependent on date. The preceding analysis assumes start of *ISS* deorbit operations in 2024, near the peak of solar cycle activity. The orbital lifetime at different starting dates for *ISS* deorbit operations will vary in accordance with the predicted trends in solar cycle activity.

The final *ISS* orbit ground track is planned to be set up from a 4-day phase-repeating circular holding altitude of 270 km, which the *ISS* will cross soon after the point of no return altitude of 279 km once the final decision to deorbit the *ISS* has been made by the *ISS* Program. This phase repeat altitude gives back-up opportunities to begin the deorbit sequence if necessary, in increments of 4 days from the originally-intended plan; however, due to the high propellant cost of maintaining this phase repeat altitude, back-up opportunities beyond 1-2 are not expected.

The descent from the 270 km holding attitude to the penultimate and final orbits must be planned in advance and managed very accurately to achieve the desired target ground track. Such descent can be very short and elliptical at the expense of propellant, or natural decay can assist over a longer period lasting between 2-7 weeks [4] in which case the resulting penultimate perigee will be nearly circular. If the plan emphasizes or necessitates propulsive descent from 270 km, the *ISS* Program presumes that all attitude control will be under propulsive control, at a propellant consumption rate that is steel TBD.

The *ISS* Program refers to the next-to-final orbit as the "penultimate" orbit. The *ISS* Program is currently optimizing deorbit of the *ISS* assuming a penultimate orbit that is 145 x 200 km, from which a large propulsive burn pushes the *ISS* to capture perigee on the following orbit. This orbit presumes propellant is used to lower *ISS* perigee, and provides opportunity for back-up final burn attempts in the event of an off nominal scenario. Alternatively, if natural atmospheric decay is used over the course of several weeks rather than propulsive deboost, this orbit is assumed to be circular at approximately135 km.

The final revolution of the *ISS* will be dictated by the necessity to achieve atmospheric capture and contain the debris footprint within the SPOUA. Assuming a 3.32 km perigee drop for each m/s ΔV, the final *ISS* re-entry burn requires a de-boost impulse of at least 21 m/s from the aforementioned 145 km penultimate perigee to achieve a capture perigee of 75 km. A burn of 24 m/s drastically improves this deorbit scenario, achieving a capture perigee of 65 km and shortening the expected debris footprint. The actual required ΔV must be larger due to the longer, less efficient nature of the final burn.

Based on existing *ISS* Visiting Vehicle (VV) thrust capabilities, the *ISS* deorbit planning team assumes that the final burn is executed in the Local Vertical, Local Horizontal (LVLH) mode near Torque Equilibrium Attitude (TEA) with minimized yaw. The *ISS* deorbit planning team is investigating alternate burn profiles and attitudes utilizing nadir / zenith docked *Progress* vehicles.

The amount of propellant needed to reach 279 km from *ISS* operational altitudes is dependent on the timeframe available for natural atmospheric decay. In the nominal *ISS* deorbit scenario, the *ISS* is assumed to have sufficient time to allow for totally natural decay from operational altitudes to 279 km, with sparse propellant expenditures only for visiting vehicle events, phasing, and attitude control. In this scenario, only the aforementioned attitude control and small phasing propellant costs are accounted for during this timeframe of natural atmospheric decay. It is preferable for phasing and debris avoidance maneuvers to be performed retrograde during this timeframe to aid natural orbital decay.

Under the nominal Russian *ISS* deorbit plan, the penultimate orbit is set up from ~200 km using the aft *Progress* R&D engines. Three 2,000 second engine firings on successive revolutions will be used to set up the penultimate orbit of approximately 200 x 145 km. These three burns utilize 8 R&D engines, and are estimated to cost a total of 2,150 kg of propellant.

Alternatively, allowing natural atmospheric drag to provide the necessary decay results in a penultimate orbit that is circular at approximately 135 km altitude. This option saves significant amount of propellant that would be used to lower perigee from 200 km, but does not provide any additional backup opportunities in the event of a failure of the final burn and will require attitude control propellant for a longer period of time.

The *ISS* Program estimates that *ISS* momentum management is capable of maintaining attitude control using CMGs during nominal deorbit operations until they are no longer effective below ~250 km. After that point, 100 kg of propellant per day is assumed to be necessary to maintain attitude and perform adjust burns throughout the deorbit period of the *ISS*, with a nadir *Progress* to provide efficient roll control.

Total propellant necessary for attitude control during this time period is dependent on expected lifetime. Under the eccentric 200 x 145 km set up plan, only a few days of attitude control is expected to be required, totaling a few hundred kilograms. With a circular set up utilizing natural orbital decay, attitude control is expected to be required for 2-4 weeks as *ISS* decays to 135 km for a total of approximately 1,400-2,800 kg of propellant for attitude control assuming an efficient roll control method is available.

Several potential burn scenarios have been identified by the *ISS* Program for the final burn. The final burn must be the highest ΔV that can be delivered while the vehicle is still capable of attitude control in the increasingly dense atmosphere at lower altitudes. Fig. 5 illustrates a proposed final burn sequence that correlates to option 6 from table 3, which maximizes the impulse available to deorbit the *ISS* during the final series of burns.

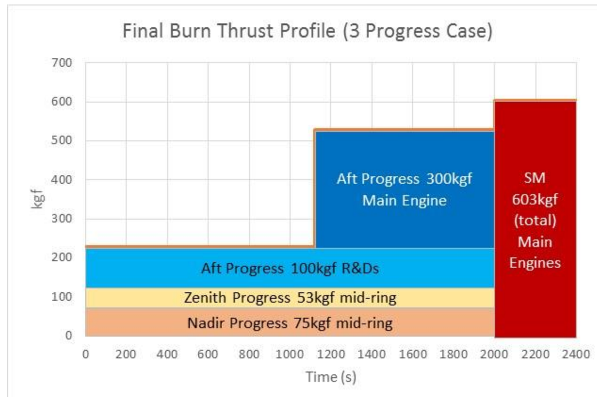

Fig. 5: Final deorbit burn nominal sequence

In the final *ISS* deorbit burn sequence, the following engines are activated (where available) to deorbit the *ISS*:

1. 8 R&D engines on the aft docked *Progress* vehicle (100 kgf of overall thrust), fired for 2,000 seconds. Total propellant cost ~670 kg from FGB tanks.
2. The main engine of the aft docked *Progress* vehicle (300 kg of thrust), fired for 800 seconds. Total propellant cost ~800 kg from *Progress* CPS tanks.
3. 4 mid-ring engines on the vehicle docked to MRM2 (53 kgf of overall thrust), fired for 2,000 seconds. Total propellant cost ~335 kg from FGB tanks.
4. 8 mid-ring engines on the vehicle docked to the MLM (75 kgf of overall thrust), fired for 2,000 seconds. Total propellant cost ~670 kg from FGB tanks. After arrival of RSOS Node, clocking on this port will be the same as MRM2. *Progress* mid-ring thrusters will provide 53 kgf and use ~335 kg of propellant from FGB tanks for the 2,000 second burn.
5. SM reboost engines (603 kg of overall thrust), fired for 340 seconds. Total propellant cost ~800 kg from SM main tanks.

Total estimated propellant consumption and the expected delta velocity change during the final burn is estimated in Table 4.

Table 4: Expected ΔV and fuel of *ISS* deorbit burn sequence per VV configuration

	Vehicle:	SM	Aft Progress		Zenith	Nadir Progress	Total	Total
	Engine:	Main	R&D	Main	Progress (MRM2)	(MLM) mid-ring	ΔV, m/s	fuel, kg
VV configuratio	0 *Progress*	4.5					4.5	800
	1 *Progress*	4.5	4.36	5.24			14.1	2,270
	2 *Progress*	4.5	4.36	5.24	2.31		16.4	2,605
	3 *Progress*	4.5	4.36	5.24	2.31	3.27	19.7	3,275

*Assumes *ISS* mass of 450,000 kg, does not take into account inefficiency of long burns

An additional ‘backup’ burn sequence can provide margin and ΔV in the event of an under-burn in the nominal burn sequence or in the event that additional propellant is still available following the final burn sequence.

Under nominal *ISS* deorbit, the *ISS* Program estimates between 5,500-7,500 kg of propellant will be necessary for execution of all deorbit operations. Much of this propellant is expected to be available in the FGB tanks, and can be used by SM main engines and docked *Progress* mid-ring or R&D engines. Any additional propellant will need to be delivered by *Progress* vehicle(s). The final deorbit impulse utilizes the aft *Progress* main engine, which will require a *Progress* vehicle with at least 800 kg remaining in the CPS tanks. Nominal *ISS* Deorbit Action Plan assumes 6 to 26 months with total propellant cost of 4,250 to 7,450 kg depending on the selected option. Fig. 6 and 7 shows *ISS* debris footprint within SPOUA for option 6 from Table 3 in main and backup day accordingly.

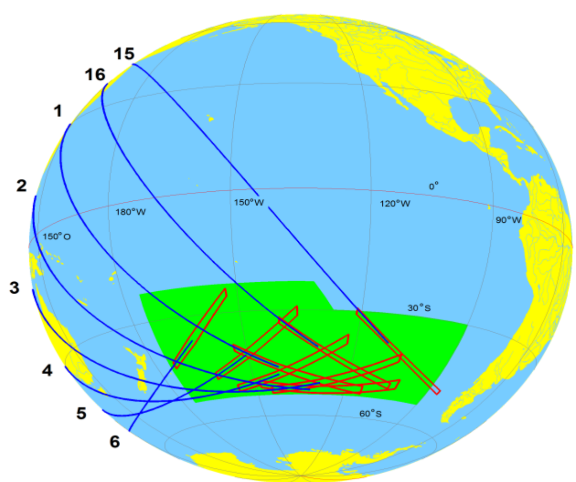

Fig.6: *ISS* debris footprint for option 6 in main day

7. Contingency scenarios

In the event of major *ISS* system(s) failure which the *ISS* Program determines mandates the start of contingency *ISS* deorbit operations, the multilateral *ISS* Program will make a determination of a safe remaining orbital lifetime of the *ISS*, depending on the nature of the contingency and expected functional capability of the *ISS* to perform deorbit operations.

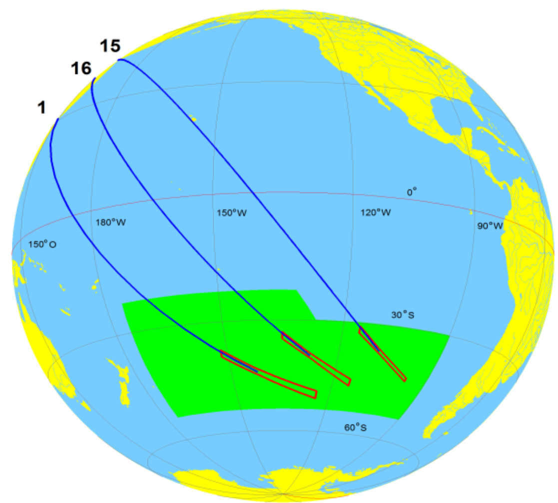

Fig.7: *ISS* debris footprint for option 6 in backup day

In the event of a potentially program ending contingency scenario, all *ISS* Program Partners will come to a consensus of deorbiting the *ISS*. The target time to make this decision is approximately 14 days after said event due to the limited lifetime estimates for vital *ISS* hardware at vacuum. This time is dependent on the propellant resources available on-board the *ISS* as well as the atmospheric drag experienced by the *ISS* at the time of the event (driven by solar cycle activity, atmospheric density, etc.).

This assessment estimates the probability of three major *ISS* contingencies over a 6 months increment which could lead to an evacuation of the crew: Micro Meteoroid Orbital Debris (MMOD) strike causing depressurization (1 in 121), Fire (1 in 227,531), and Toxic release of ammonia (1 in 14,000).

This study focuses on the most likely contingency scenario to cause premature deorbit of the *ISS*: MMOD strike causing irreversible depressurization of the *ISS*.

In the event of a MMOD strike to *ISS*, there is expected to be a significant likelihood of irreversible depressurization of the *ISS*. In this scenario the *ISS* needs to be safely deorbited as soon as practical, since

SM avionics systems are certified to remain operational in vacuum for 180 days. This study assumes that the entire *ISS* crew has evacuated the *ISS* immediately following the MMOD strike event.

The primary objective for USOS operations is to maintain command and control capability and perform drag-based altitude reduction until the final series of deorbit burns can be executed. Key aspects include:

- Maximizing CMG use to conserve propellant to support translational burns
- Maximizing solar array drag to aid in *ISS* altitude decay
- Keeping FGB propellant system components from freezing prior to propellant use

Fig. 8 illustrates the *ISS* engines expected to be available for contingency deorbit based on the current RSOS system limitations at vacuum, specifically lack of capability to simultaneously fire multiple *Progress* engines or dock additional *Progress* vehicles with *ISS* at vacuum.

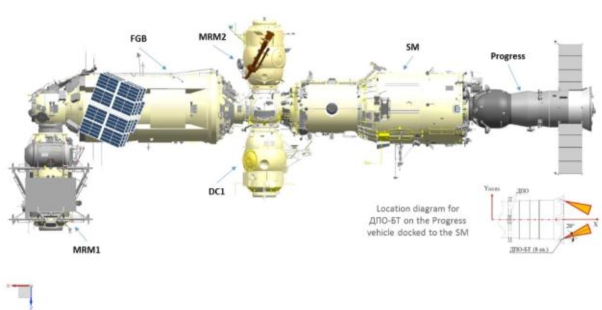

Fig. 8: Current *ISS* contingency engine capability altitude strategy and propellant requirements

With only 180 days remaining to perform all *ISS* deorbit operations, the deorbit planning team estimates that approximately 4,500 kg of propellant will be necessary to de-boost the *ISS* from operational altitude to 345 km**.** This estimate assumes 4000 kg of propellant will be used for de-boost impulse burns and an additional 500 kg of propellant for attitude control. Fig.8 illustrates the un-assisted natural decay of the *ISS* during a potential *ISS* contingency deorbit scenario assuming an *ISS* BN of 107 kg/m^2 and using average solar activity predictions.

As illustrated in Fig.9, limiting natural decay to approximately 5 months (allowing an additional month for the decision process and final deorbit burn sequence) requires an altitude of approximately 345 km at the start of the natural decay period. The propellant required to lower *ISS* from operational altitude of 400 km to 345 km is estimated to be approximately 4500 kg. This scenario assumes a burn schedule of 5 m/s de-boost burns per day. Once 345 km is reached, the *ISS* will naturally decay to an altitude 279 km over the course of approximately 5 months, depending on specific orbital parameters, *ISS* ballistic configuration, and atmospheric density.

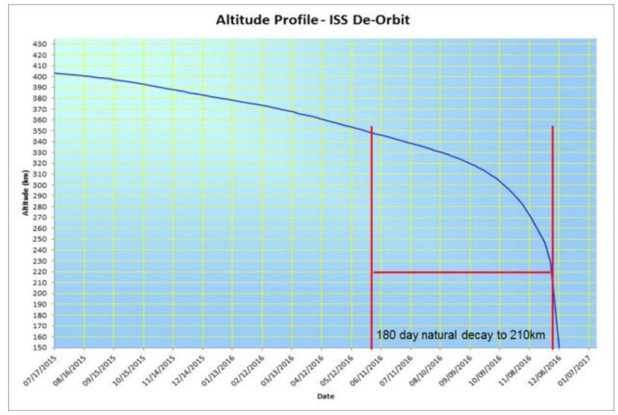

Fig. 9: Natural atmospheric decay estimates for *ISS* contingency deorbit

Due to the time sensitive nature of this burn sequence and the expected remaining lifetime of *ISS* avionics systems in vacuum, these initial de-boost burns must be agreed on and performed as soon as possible. The *ISS* deorbit planning team recommends that these de-boost burns should begin between 14-45 days post contingency event. Because of the higher drag experienced by the *ISS* at lower altitude ranges, performing the initial de-boost burns early provide additional time and propellant margin for the final setup and deorbit burn sequences.

In the event that sufficient propellant is not available to support both the final deorbit burn sequence and assisted decay from operational altitudes, the *ISS* Program will weigh risks and make a determination how to utilize available propellant. Two options are identified:

- Maintaining 3000 kg of propellant in *ISS* tanks for use in the final deorbit burn sequence, thus extending the timeframe for *ISS* deorbit operations beyond 180 days.
- Utilizing available propellant to lower *ISS* altitude such that the *ISS* will deorbit within 180 days, and accepting risk due to smaller final burn sequence and extended footprint. In this case, the *ISS* Program will make a determination of the particular daily orbits that will cause the lowest Expectation of Casualty.

Selection of one of the above options will be dependent on propellant resources available on *ISS*, as well as *ISS* functionality (capability to dock additional *Progress* vehicles / propellant & *ISS* systems capability to perform the final deorbit burn) and solar cycle / atmospheric conditions at the time of the contingency.

In this scenario the *ISS* Program has determined that CMG control use will be the best option until the *ISS* reaches an altitude of 230-250 km. 100 kg is the current estimate for required per day attitude control propellant once CMGs are no longer effective, while the *ISS* naturally decays to the penultimate perigee. This is estimated to take 14-28 days, resulting in 1,400-2,800 kg of propellant being used for attitude control alone.

The *ISS* Program believes that in this scenario, the *ISS* should utilize natural atmospheric drag to decay to a penultimate circular orbit of 135-145 km to preclude the additional use of propellant.

The final burn scenario, and thus the propellant required for the final burn, will be determined by the configuration and capabilities at the time of the contingency. The best case sequence assumes a *Progress* vehicle with 800 kg in main tanks docked to SM aft. Refer to Table 5 for potential fuel capability of the final burn sequence.

Table 5: Fuel capability for final burns

Vehicle:	SM	Aft Progress		Total fuel, kg
Engine:	Main	R&D	Main	
0 *Progress*	800			800
1 *Progress*		670	800	1470

Contingency *ISS* Deorbit Action Plan assumes total propellant requirement to be between 7,200 and 9,270 kg depending on the selected deorbit scenario and Solar activity at the time of deorbit. It is important to point out that the final deorbit burn executed by a single *Progress* vehicle does not guarantee that all *ISS* debris will remain within SPOUA. To meet SPOUA constraint, it is anticipated that either 1) 3 *Progress* vehicles will be required to execute the final burn or 2) SM main engine firings in addition to an aft docked Progress main engine firing will be required for the final burn sequence. The *ISS* altitude at the time of the execution of the final deorbit burn should remain within 130 to 140 km. Fig. 9 shows footprints relative to SPOUA for 140 km of initial *ISS* altitude at the time of the final deorbit burn execution.

8. Conclusions

The results of the *ISS* nominal and contingency deorbit capability study have shown that nominal deorbit can be performed by the current capabilities in current *ISS* configuration. In that case 4,250 to 7,450 kg of propellant will be required to perform the deorbit operations and can support three to four back up orbits for contingency operations.

In case of a contingency deorbit, the expected propellant requirement can vary from 7,200 to 9,270 kg and cannot guarantee all debris to be contained within SPOUA with current *ISS* capabilities.

Fig. 9 Footprints after deorbit from 140 km circle orbit

In order to improve the *ISS* thrust capability and ensure guaranteed *ISS* safe deorbit in case of a contingency, the *ISS* Program is actively investigating other options.

Acknowledgements

Authors express great appreciation to Dr. Jack Bacon and Jim Cooney for their efforts in developing *ISS* nominal and contingency deorbit strategies.

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