Feasibility Study on the Detection of Hypersonic Weapons using Cube-Satellites

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Hypersonic Glide Vehicles are an emerging area of weapons development promising to combine the speed of a ballistic missile with the manoeuvrability of powered flight. This allows these weapons to avoid areas of radar detection and penetrate defences with such speed that current missile defences are inadequate and reaction times are significantly reduced. It is unknown whether current infrared detection methods from low earth orbit can detect, characterise and provide enough information to enable target prediction upon these threats. The objective of this study is to identify the feasibility of using a cluster of cube-satellites to enable the detection, tracking and characterisation of these threats to expand the response time frame.

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Appendix A. Equations


Variables

\begin{align*}
    a &= \text{acceleration} \quad [\text{m} \cdot \text{s}^{-2}] \\
    \alpha &= \text{angle of view} \quad [\text{rad}] \\
    A &= \text{surface area} \quad [\text{m}^2] \\
    \Theta_p &= \text{angular projection} \quad [\text{rad}] \\
    \Theta &= \text{angular resolution} \quad [\text{rad}] \\
    D &= \text{aperture diameter} \quad [\text{m}] \\
    K_B &= \text{Boltzmann constant} \quad [1.381 \times 10^{-23} \text{J} \cdot \text{K}^{-1}] \\
    \rho &= \text{density} \quad [\text{Kg} \cdot \text{m}^{-3}] \\
    d &= \text{distance} \quad [\text{m}] \\
    C_d &= \text{drag coefficient} \quad [\text{Non-Dimensional}] \\
    e_r &= \text{Earth radius} \quad [6371 \text{ km}] \\
    E &= \text{energy} \quad [\text{Wh}] \\
    f &= \text{focal length} \quad [\text{m}] \\
    F &= \text{force} \quad [\text{N}] \\
    v &= \text{frequency} \quad [\text{Hz}] \\
    G_{Rx} &= \text{gain (receiving)} \quad [\text{dB}] \\
    G_{Tx} &= \text{gain (transmitting)} \quad [\text{dB}] \\
    g_0 &= \text{gravitational acceleration} \quad [\text{m} \cdot \text{s}^{-2}] \\
    h &= \text{height} \quad [\text{m}] \\
    s &= \text{image sensor circle size} \quad [\text{m}] \\
    L &= \text{integrated radiance} \quad [\text{W} \cdot \text{sr}^{-1} \cdot \text{m}^{-2}] \\
    m &= \text{mass} \quad [\text{Kg}] \\
    d_p &= \text{pixel pitch} \quad [\text{m}] \\
    p &= \text{Planck constant} \quad [6.626 \times 10^{-34} \text{J} \cdot \text{s}] \\
    P &= \text{power} \quad [\text{W}] \\
    r_v &= \text{resolution (vertical)} \quad [\text{Non-Dimensional}] \\
    r_h &= \text{resolution (horizontal)} \quad [\text{Non-Dimensional}] \\
    \ell &= \text{spatial resolution} \quad [\text{m}] \\
    I_{sp} &= \text{specific impulse} \quad [\text{s}] \\
    B_s &= \text{spectral radiance} \quad [\text{W} \cdot \text{sr}^{-1} \cdot \text{nm}^{-1} \cdot \text{mm}^{-2}] \\
    c &= \text{speed of light} \quad [2.998 \times 10^8 \text{m} \cdot \text{s}^{-1}] \\
    \mu &= \text{standard grav. parameter} \quad [3.986 \times 10^{14} \text{m}^3 \cdot \text{s}^{-2}] \\
    T &= \text{temperature} \quad [\text{K}] \\
    t &= \text{time} \quad [\text{s}] \\
    v &= \text{velocity (orbital)} \quad [\text{m} \cdot \text{s}^{-1}] \\
    \Delta v &= \text{velocity (change)} \quad [\text{m} \cdot \text{s}^{-1}] \\
    \lambda &= \text{wavelength} \quad [\text{m}]
\end{align*}
I. Introduction

The successful development of hypersonic weapons is inevitable. There are many different hypersonic research programs being undertaken throughout the world. It is increasingly clear that sustainable, controlled hypersonic flight is being actualised. Programs such as the European Union’s ‘LAPCAT II’, American-Australian ‘HiFiRE’, and French ‘LEA’ are examples of ongoing hypersonic research programs developing the fundamental technologies enabling hypersonic speeds. While the Indian-Russian ‘BrahMos II’, Chinese ‘Dongfeng-17’ (DF-17), Russian ‘Avangard’ and ‘Kinzhal’, and American ‘HAWC’ are examples of hypersonic research being directly implemented into weapons programs. As of October 2019, three hypersonic payload delivery systems have been announced as operational, being the DF-17, Avangard and Kinzhal.

Hypersonic weapons can be divided into two distinct categories; Hypersonic Glide Vehicles (HGVs) and Hypersonic Cruise Missiles (HCMs). HGVs work in principle by first being launched atop a ballistic missile to gain altitude and lateral velocity, before separating and using both the atmosphere and a lifting body to control a trajectory which ‘skips’ across the upper atmosphere before diving to a target. HGVs are typically released above 50km altitude and travel at an altitude of 100km. Transit speeds for these weapons are between Mach 5 and Mach 30 (about 5,000–25,000 km/hr) which is sustainable in this low drag, high altitude environment. A typical HGV trajectory can be seen in figure 1.

II. System Requirements

This study will analyse the feasibility of a satellite system to detect, characterise and provide target predictions of HGV threats in the upper atmosphere. Systems capable of detecting the infrared (IR) signatures from missiles (such as the American ‘DSP’ and Russian ‘OKO’) were first developed during the cold war for intercontinental ballistic missiles (ICBM). These systems have been improved upon with the current American ‘SBIRS’ and Russian ‘EKS’. However, from an unclassified perspective, both experience limitations and can only detect large IR signatures such as the exhaust of an ICBM during its preliminary boost phase. The full capability of these systems is currently unknown to the public. Once the payload separates, it becomes difficult to actively track and characterise these targets due to a low temperature differential against the observation background. In contrast to an ICBM launch, once the HGV has utilised the boost phase’s fuel, it will skip along the upper atmosphere, constantly generating heat, particularly within the thin leading edges. Due to the small area, but intense magnitude of IR radiation emitted, a detector positioned at a lower altitude, closer to the in-flight HGV, is in a better position to return a positive identification. While positioning large systems such as SBIRS High or OKO at a lower altitude would be an ideal solution, the constellation required to provide adequate coverage would lead to an unfeasible total program cost. While being close to the HGV will yield the best chance of a positive identification, systems located in LEO encounter considerable drag which reduces service life considerably. For the purpose of this feasibility study, an operational system lifespan of 4 years is desired.

To provide adequate earth coverage from LEO it is necessary to have multiple satellites to form a cluster between the designated inclination bands, allowing multiple satellites to combine their arcs of observation into an area which provides almost seamless coverage. Due to this need for multiple satellites in a lower Earth orbit, each unit should be optimised, in the smallest form factor possible, to reduce launch complexity and program cost. Therefore, a Cube-Satellite form factor is desirable, with the initial projection of 3U being aimed for. Each 1U accounts for 10cm x 10cm x 10cm volume, weighing typically 1.33kg (3lb) per U.

For the purposes of this final report, a significant portion of the analysis has been omitted to accommodate the 10-page content limit. The complete report can be found as appendix A or by contacting Harrison.Baildham-Parr@defence.gov.au.

References

2 (Speier, et al., 2017)  
3 (China.com, 2019)  
4 (Speier, et al., 2017)  
5 (Lof, 2018)
III. System Components

A. Infrared Detector

Due to the small angular size of hypersonic targets being observed from orbital heights, the two main detection methods are radar and infrared sensor detection using either the earth (overhead) or space (oblique) as the observation background to the target. This study will be analysing infrared detection from the overhead perspective in detail.

There are a range of variables to consider when identifying a detection system for the proposed cube-satellite. One of the most important factors is the wavelength bands the sensor will be conducting surveillance in. To identify these bands, a range of hypersonic vehicle temperatures covering a range of mission profiles must be established which allows for blackbody emission profiles to be established. Once these blackbody profiles have been established, sections which correlate to natural wavelengths of atmospheric absorption can be selected to minimise earth-based interference. The distance between sensor and HGV, combined with the small size of HGV’s pose the greatest limitation, causing very small arc-angles for sensors which need to fit in a cube-satellite form factor. This problem can be addressed through the smallest micrometre magnitude sensor pixel pitch and large optical focal length.

1. Atmospheric Absorption

Extensive research on atmospheric IR spectrum absorbance has been conducted as it has played a vital role in earth-based infrared astronomy. Figure 2 shows atmospheric absorption in the infrared spectrum. Areas of low transmittance can therefore be used by the proposed cube-satellite to identify highly thermal radiating bodies while mitigating earth-based background interference such as reflections.

Therefore, the following wavelengths (marked red in figure 2) are appropriate to be used by the proposed cube-satellite system:
- 1.35 μm – 1.4 μm
- 1.83 μm – 1.91 μm
- 2.55 μm – 2.81 μm and;
- 4.16 μm – 4.41 μm

2. Hypersonic Vehicle Temperature

To detect hypersonic weapons utilising IR sensors, a temperature range must be identified. Due to extreme dynamic pressures exerted on hypersonic vehicles, aerothermal heating is a dominant factor limiting operational speed. Due to this, a range of temperatures between 1500°C and 3000°C will be applied to cover a range of different operating conditions for these HGVs. This temperature range was confirmed to be valid by conducting a customised simulation via the DSMC hypersonic flow software as seen in figure 3. This was modelled on the Chinese DongFeng-17, flying at an estimated 100km altitude at Mach 10. The highest temperatures achieved at the tip being 2000°C, while 11m behind this point, at the rear of the weapon, temperatures of 1770°C were still being observed.

To obtain the most accurate IR radiation footprint for the HGV system, the emittance of the hypersonic boundary layer should be included. However, for a simplified model, the HGVs leading edges will be assumed to act as a blackbody surface with the skin reaching thermal equilibrium to the boundary layer. As shown by simulations above, this equilibrium can be assumed to be above 1500°C across the entire weapon. A material

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6 (Richmond, 2004)
assumption of Carbon-Carbon composite for the HGV body has been made. This material assumption was made due to its prevalence in high temperature resistance – low weight aerospace applications and has an emissivity of 0.8.\(^7\) Radiance curves were then calculated via Planck’s Law, equation 1, and integrated radiances found via Stefan-Boltzmann law, equation 2, to produce figure 4.

3. Resolution and Detection

Typical hypersonic vehicles will only have a small leading-edge area reaching the target temperatures noted above, while the rest of the vehicles larger surface area will emit lower amounts of IR radiation. These lower temperatures may still reach greater than 1,000°C as demonstrated in figure 3 and can therefore still contribute to identify a target. Collecting enough of this light to make a positive identification is a critical aspect of this hypothetical system.

There are two ways the system could handle HGV target identification. If it required that the HGV is to be characterized using resolved images, the image passed through the optics and impressed upon the sensor must cover at least 4-8 pixels. This requirement would increase complexity, weight, size and cost of the system drastically as seen in the full report analysis. The alternative is to pursue the acquisition of an unresolved image, occurring when the incident image of a HGV upon the detecting sensor is the size of one pixel or smaller. While this information would not be able to adequately characterize all aspects of a HGV, key parameters would be observed. Using distance covered per frame and time, the velocity of the object could be obtained, allowing initial target prediction calculations to be iterated. If a second detector was able to observe the object more information such as height could be determined, further narrowing target predictions. This would also allow an atmospheric density lookup to be conducted, which if combined with velocity data, can be used to find the characteristic spectral radiance. Comparing the estimated spectral radiance against the measured values, a size estimate could be established, allowing for rough characterisation while still utilising an unresolved image. This would allow the system’s complexity, size and cost to be reduced while still maintaining the key detection properties of the system. For the purpose of the summary report, an unresolved system will be pursued.

4. Pixel Pitch and Sensor

A key parameter in detecting potential HGV threats is the pixel pitch of the sensor being utilised. When considering IR cameras, pixel pitch is the length and height of a light-collecting pixel, usually square. The size of this pixel is a major contributing factor to the construction of the surrounding lens, defining variables such as the focal length and aperture required when trying to detect objects of different apparent sizes at varying ranges. Due to the size constraints placed upon cube-satellites, for the proposed system, it is a requirement to select a sensor with the smallest pixel pitch available, as this reduces the focal length required to collect the same number of photons per pixel\(^8\). Recently, IR sensors have entered the 10μm x 10μm size with companies such as Sofradir\(^9\) and Raptor Photonics offering sensors in this size, with a large array of companies offering larger 12μm -15μm detectors. Due to the requirements listed above, a suitable candidate for the system sensor could be the OWL 1280 VIS-SWIR. With a spectral response of 0.4μm -1.7μm it can comfortably detect the highest emission peak range of 1.35μm – 1.4μm for typical HGVs.

The most important measure of the system is the signal to noise (S/N) ratio. This ensures that the key information, the HGV ‘signal’ is not being obscured by background ‘noise’. While the selection of wavelength allows the atmosphere to absorb almost all detectable light, some light sources within the 1.4μm region may be observed, potentially becoming false positives and obscuring the signal of interest. Sunlight reflection off high altitude clouds will be particularly troublesome due to the lack of atmospheric density (and therefore atmospheric absorption properties) between cloud and satellite. Potential solutions to this issue would most likely be software based, ensuring speed and altitudes of identified signals align with a typical HGV profile. Another solution would be to analyse the characteristic emission spectrum of high-altitude hypersonic vehicles, create a profile based on the ionic dissociation emissions caused by aerothermal and shock effects, and then incorporate a remote spectral sensor into the system. This would allow for observation of this particular emission spectrum and provide a high degree of detection certainty. This method however would bring its own challenges due to upper atmospheric condition and HGV altitude variations. A further analysis would be required to determine feasibility of such a

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\(^7\) (Li & Strieder, 2009)  
\(^8\) (Clark, 2005)  
\(^9\) (Reibel, et al., 2015)
system which lies outside the scope of this investigation, which will be focusing on a low-cost infrared imaging approach.

5. Optics

The sensor mentioned above is currently enclosed and produced as a camera system. While this allows it to be integrated easily in some products, it will have to be heavily modified to become a solution in the proposed detection system. Due to the target system size of 3U (10cm x 10cm x 30cm), the aperture is restricted to a maximum size of 10cm, with a straight-line focal length of 30cm if the entire cube-satellite is utilised. The geometry of the problem can be seen in figure 5, demonstrating how equation 3 works to imprint a resolved image upon the imaging sensor. Due to our analysis revolving around an unresolved image, the target size would increase with focal length decreasing. For the proposed system, a purely refractive systems should be utilised as the desired focal length is reduced to 59mm as seen in later analysis.

![Figure 5. Simplified Optical Geometry (not to scale)](image)

6. Angle of View: Unresolved Images

Utilising unresolved images to obtain useful information is not an unknown concept in astronomy. If our system is designed to produce unresolved images, it can be expected to return dark images with a bright point source of light which could be the potential HGV. This would occur due to the HGV being smaller than the minimum size as mentioned in equation 3. If that equation is rearranged as shown in equation 4, the minimum size a HGV would have to be can be calculated.

While utilising unresolved imaging causes a loss of HGV detail, the key aims of the constellation could still be achieved. To calculate angle of view (AOV, \(\alpha\)), first the size of the image circle (s) imprinted upon the sensor must be found. This is done utilising Pythagoras’ theorem as demonstrated in equation 5.\(^{10}\) In this equation Resolution \((r_0\) and \(r_1\)) is being measured in number of pixels (non-dimensional), and pixel pitch is \(\mu m\). The system will utilise a rectilinear projection. Once the circle size is obtained, the AOV can be calculated through equation 6,\(^{11}\) derived from trigonometry of the camera’s internal geometry. Using a sensor size of 16.39mm, and focal length of 59mm, an AOV of 15.8° will result. For a satellite at height \(h\) of 400km, this correlates to a ground coverage of approximately 12,300km\(^2\) using equation 7.

7. Diffraction Limit

The ability for an optical system to ultimately resolve detail is limited by diffraction. This is largely dependent on the distance between the sensing satellite and target HGV (causing a small angular diameter), wavelength utilised for detection, sensor and aperture size. With the currently imposed system limitations of an aperture 0.1m and detection wavelength \(\lambda\) of 1.4μm, a minimum focal length of 0.059m, or 59mm is necessary to eliminate diffraction and airy disk effects.

8. Sensor Tiling

To increase the amount of information gathered by each satellite, sensors can be ‘tiled’ into a panel increasing the image circle, this allows a greater spatial resolution for a given AOV. The drawbacks of this system include an increased computational load, increasing both processing power and therefore onboard computer costs, as well as additional sensor and integration costs in the imager. On a cost-analysis basis, the tiling of sensors would provide diminishing returns if scaled above a 4x4 array.

The results in table 1 can be seen for a range of tiling options. The target resolution represents how big a HGV would have to be to completely fill one pixel of the sensing array. If the focal length of 59mm (limited by diffraction effects) calculated previously is used, a maximum AOV of 58° can be seen when utilising a 4x4 sensor array. This 59mm focal length also permits a reasonable target resolution of 51m per pixel.

<table>
<thead>
<tr>
<th>Sensor Array</th>
<th>1 x 1</th>
<th>2 x 2</th>
<th>3 x 3</th>
<th>4 x 4</th>
</tr>
</thead>
<tbody>
<tr>
<td>Resolution</td>
<td>1280 x 1240</td>
<td>2560 x 2048</td>
<td>3840 x 3072</td>
<td>5120 x 4096</td>
</tr>
<tr>
<td>Sensor Size (mm)</td>
<td>12.8 x 10.24</td>
<td>25.6 x 20.48</td>
<td>38.4 x 30.72</td>
<td>51.2 x 40.96</td>
</tr>
</tbody>
</table>

\(^{10}\) (Quay Cameras, n.d.)

\(^{11}\) (McCollough, 1983)

Final Project Report 2018, UNSW Canberra at ADFA
9. Final Recommendation

It is recommended that each unit’s imaging system utilises an unresolved image topography and possesses a 4x4 array of OWL 1280 VIS-SWIR sensors with a custom lens and camera unit. This unit should function using a focal length of 59mm, allowing a spatial resolution of 51m per pixel when observing a target 300km away. It will also allow the system to possess an AOV of 58° which will reduce the total satellite count significantly.

B. Processor

The computational load for the system is heavily dependent on the output of the imaging system. This can be fully characterised through framerate, resolution and output format. The OWL 1280 VIS-SWIR is a 12-bit sensor of 1280x1024 pixels, causing each frame to result in 15.7Mbits of data. The minimum frame rate as specified by the manufacturer is 10fps resulting in an absolute minimum data rate of 157.3 Mbit/s

\[
1280 \times 1024 \times 12 = 15.7\text{Mbit per frame} \quad 15.7 \times 10 = 157.3\text{Mbit per second}
\]

<table>
<thead>
<tr>
<th>Sensor Array</th>
<th>1 x 1</th>
<th>2 x 2</th>
<th>3 x 3</th>
<th>4 x 4</th>
</tr>
</thead>
<tbody>
<tr>
<td>Resolution</td>
<td>1280 x 1240</td>
<td>2560 x 2048</td>
<td>3840 x 3072</td>
<td>5120 x 4096</td>
</tr>
<tr>
<td>Raw Data Rate (10fps)</td>
<td>157.3 MB/s</td>
<td>629.2 MB/s</td>
<td>1.42 GB/s</td>
<td>2.52 GB/s</td>
</tr>
<tr>
<td></td>
<td>19.7 MB/s</td>
<td>78.7 MB/s</td>
<td>177 MB/s</td>
<td>314.6 MB/s</td>
</tr>
</tbody>
</table>

Table 2. Minimum required data rate

There are a number of commercial off the shelf (COTS) solutions that can be utilised within the system. A leading candidate however is the Innovative Solutions in Space (ISIS) on board computer (OBC) which has a particularly high clock rate compared to other OBC’s. While it is too early to properly determine complete computing architecture, a higher clock rate strongly correlates to better performance, lending it to deal with the handling of data rates mentioned in table 2. It will also allow the system to perform target location and velocity predictions based on processed images onboard instead of having to offload this data to ground based systems.

C. Communications

There are two main topologies that can be utilised for communications between satellites and ground stations, being geostationary relay and direct to ground link. However, the summary report will only cover the direct link methodology. Both communication methods require intersatellite communication to allow HGV information to be relayed throughout the network while also allowing communications with any unit at any given time. This can be achieved with relatively low power requirements if an appropriate frequency is selected. While free space loss is increased in accordance with the inverse square law at higher frequencies and distances, high frequencies may be utilised in this application as there are no atmospheric interference. Data rate transfer is also increased with higher frequency at the expense of additional power and module size requirements. If HGV position estimation can be conducted aboard satellites, total data transmission volumes will be minimal.

The biggest restriction on data transfer rate within the satellite array will be power consideration, with power available being in the 1W magnitude. Although an accurate estimation of data rate required can’t be ascertained without delving into the software aspects more thoroughly, at least 10kbps of network bandwidth would be required.

12 (Ortega Varela de Seijas, 2016)
1. **Intersatellite Communications**

Omnidirectional intersatellite communications is required for not only constellation management reasons, but to also increase system capability. Having satellites in the vicinity will allow the initial detector to notify surrounding assets that a HGV is in their adjacent sector. Without an individual target assigned, these nearby satellites can utilise attitude control systems to orientate and provide a more accurate HGV location prediction by observing from a different angle. Ideally, a minimum of three satellites will be able to observe the same HGV to provide positional triangulation, however this raises concerns of positively identifying the same HGV as well as complications arising if multiple HGV’s are in flight. This will lead to complexification of the software package, increasing system development time and cost.

Intersatellite communications will have to be omnidirectional to enable links between network nodes without altering each unit’s attitude to establish a connection. Free space loss (FSPL) due to the inverse square law is the largest source of loss within the system and is calculated using $d$ as distance, $\nu$ as frequency and speed of light constant as $c$ via equation 10.\(^{13}\) To quantitatively analyse the parameters of the intersatellite communications network, a link budget can be conducted. This allows relative signal strength and overall network effectiveness to be calculated and can be found in the extended report.

2. **Ground Link**

One of the benefits of utilising direct satellite to ground station communications is that the distance being travelled is significantly smaller (400km minimum compared to the 35,386km minimum of geostationary transmission), allowing power requirements to be significantly reduced. This would result in a FSPL of 162dB compared to the geostationary value of 201dB for the X-Band frequency. Another benefit is the ability to have a larger geographical distribution of ground-based nodes which can receive data, allowing for communication redundancy instead of relying on the single geostationary Optus C1. Drawbacks to this approach include the necessity to select communication frequencies which are not impacted heavily by atmospheric events and possible gaps in communication if constellation nodes are not in range of a ground station. This latter point could be minimized with strategic base station placement. To minimise power requirements for satellite transmission, earth-based transceiver stations can utilise steerable parabolic antennas with higher gains. However, a cheaper alternative is to use omnidirectional dipole antennas. If implemented, earth-based power requirements would be increased, however the reduction in system complexity would offset this cost, being an ideal solution.

D. **Propulsion**

For the purposes of this analysis, the assumption is being made that the satellite will be in a low enough orbit (in order to gain the closest proximity to HGV activity) that atmospheric drag will be the dominant factor limiting mission lifespan. To overcome this and extend mission time, each unit will have to be equipped with a propulsion system. Key factors to consider while analysing and comparing propulsion methods include size and power constraints, thrust produced, and the additional mission time offered.

To begin calculating the method of propulsion needed, it is important to know the drag of the satellite profile. This is dependent on a number of factors as shown in equation 12. Variations in the upper atmospheric density are the main source of unreliability in orbit lifetime predictions. Generally atmospheric density is modelled using an exponentially decreasing profile which while being simple, is not a true representation of conditions. This is due to atmospheric density being complex and exhibiting spatial and temporal variations. Analysis conducted in the full report provides a drag force of $3.82 \times 10^{-7}$ N.

Due to the minute amount of drag being exerted upon the cube-satellite, an electrospray propulsion system can be utilised. This system operates on the principle of electrostatic extraction and acceleration of charged particles (ions) from a liquid (propellant) surface to produce thrust. This system would be appropriate for this mission due to its high specific impulse and low thrust. The biggest drawback of using electric based propulsion revolves around ensuring that enough power is being generated onboard to supply the module. However due to the low amounts of thrust required, power consumption should also be minimal.

Upon reviewing an overview of cube-satellite propulsion systems, including comparison and analysis\(^{14}\), units within the constellation could utilise the flight heritage (Aero-Cube-8) S-iEPS system produced by the Massachusetts Institute of Technology (commercially Accion Systems)\(^{15}\). Through analysis seen in the extended report, this system could provide up to 7 years life extension. Realistically this value would be less due to the

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\(^{13}\) (EverythingRF, 2015)

\(^{14}\) (Tummala & Dutta, 2017)

\(^{15}\) (Vitug, 2017)
spacecraft only needing periodic orbital adjustments and altitude boosts. The time calculated above is in addition to the nominal time of flight which is determined in section IV.B.

E. Attitude Control

To properly maintain satellite orientation in space, ensuring the camera is observing the earth at all times, each unit will have to have an attitude control system which can make small adjustments to the rate of rotation. This system will also prove particularly useful for orbital parameter adjustments, entering “safe” mode where solar panels are pointed directly at sunlight and potentially orientate satellites for communications purposes. It will also allow for satellites nearby a HGV detection event to orientate in such a way that additional information about the HGV’s position can be ascertained, assisting in position triangulation. An attitude control system (ACS) such as the NewSpace Systems (NSS) cube-satellite ACS would need to be used in conjunction with any of the solutions discussed below. This allows processing and management of attitude to be managed by a dedicated unit, freeing main OBC processing power for image interpretation. This ACS possesses a GPS receiver which is necessary for constellation maintenance and positional accuracy when identifying potential HGV’s. It also possesses a Three-Axis magnetic field sensor suitable for coarse attitude measurement as well as a micro electromechanical system (MEMS) inertial rate sensor.16 High fidelity attitude data will be provided during sunlight hours by the CubeSpace CubeSense module.17 “Dark” attitude control will be provided by the ACS integrated MEMS and magnetic field sensors.

1. Momentum Wheels

Momentum wheels allow for excellent positional control, however this comes at the expense of component size, mass and power consumption. This type of system also relies on moving parts to operate which have the potential to fail. The smallest commercial example of this type of system is the Nano Avionics CubeSat Reaction Wheels Control System SatBus 4RWO.18 It is recommended that an ACS control bus is used in conjunction with the CubeSpace CubeSense module and NanoAvionics 4RWO SatBus.

F. Chassis

The system will be housed within a 3U chassis to provide adequate room for all components. The next most common chassis form factor is 6U, essentially being 2 3U chassis’ side by side. This larger chassis was explored at the commencement of this project; however, it was deemed the main payload (HGV detector) and support systems could be housed within the 3U frame. After conducting a comparison between popular cube-satellite component providers, it was determined that the EnduroSat 3U CubeSat Structure II offers the best support for the selected components at the lowest cost and weight.19

G. Power Requirements

To maintain night-time observation of Earth, batteries must be implemented to power the satellite when solar panels are non-functional. Solar panels will be used as the primary source of power regeneration as it is light, compact and reliable in LEO. Table 3 below outlines the best fit system components identified for the proposed mission and attempts to estimate power consumption based on the available technical data. Cells highlighted green are confirmed manufacturer data, while grey is estimated or inferred data.

<table>
<thead>
<tr>
<th>Component</th>
<th>Brand</th>
<th>Model</th>
<th>Minimum</th>
<th>Average</th>
<th>Maximum</th>
</tr>
</thead>
<tbody>
<tr>
<td>Imaging</td>
<td>Raptor Photonics</td>
<td>OWL 1280 SWIR</td>
<td>2</td>
<td>2</td>
<td>3</td>
</tr>
<tr>
<td>Processor</td>
<td>ISIS</td>
<td>OBC</td>
<td>0.3</td>
<td>0.4</td>
<td>1</td>
</tr>
<tr>
<td>Communications</td>
<td>ISIS</td>
<td>Full Duplex Transceiver</td>
<td>0.48</td>
<td>0.48</td>
<td>4</td>
</tr>
<tr>
<td>Reaction Wheel</td>
<td>Nano Avionics</td>
<td>4RWO</td>
<td>0.1</td>
<td>0.6</td>
<td>6</td>
</tr>
<tr>
<td>ACS</td>
<td>NSS</td>
<td>NACS-001</td>
<td>1</td>
<td>1.5</td>
<td>3</td>
</tr>
<tr>
<td>Attitude Sensors</td>
<td>CubeSpace</td>
<td>CubeSense</td>
<td>0.1</td>
<td>0.1</td>
<td>0.36</td>
</tr>
<tr>
<td>Propulsion</td>
<td>Accion</td>
<td>Tile 500</td>
<td>0</td>
<td>1.5</td>
<td>2</td>
</tr>
<tr>
<td><strong>Total</strong></td>
<td></td>
<td></td>
<td>3.98</td>
<td>6.58</td>
<td>19.36</td>
</tr>
</tbody>
</table>

Table 3. Satellite System Power Utilisation

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16 (NewSpace Systems, n.d.)
17 (CubeSpace, 2019)
18 (NanoAvionics, 2019)
19 (EnduroSat, 2019)
1. Solar Panels

There are three dominant companies which provide power solutions for cube-satellites; Endurosat, ISIS and GOMSpace. Each offers fundamentally the same product; however, pricing, weight and additional features somewhat vary. Table 4 below outlines these variations to enable easy comparison.

<table>
<thead>
<tr>
<th>Brand</th>
<th>Additional Features</th>
<th>Cost (AUD)</th>
<th>Power (W)</th>
<th>Mass (g)</th>
</tr>
</thead>
<tbody>
<tr>
<td>GOMSpace</td>
<td>Sun &amp; temperature sensors</td>
<td>$3,000</td>
<td>$20,000</td>
<td>2.3</td>
</tr>
<tr>
<td>EnsuroSat</td>
<td>Sun &amp; temperature sensors, gyroscope</td>
<td>$2,500</td>
<td>$17,500</td>
<td>2.4</td>
</tr>
<tr>
<td>ISIS</td>
<td>Harnessing, sun &amp; temperature sensors</td>
<td>$4,000</td>
<td>$24,000</td>
<td>2.3</td>
</tr>
</tbody>
</table>

Table 4. Solar Panel Comparison

Table 4 highlights that although power generation of the GOMSpace offering is lower than EnduroSat, it still provides more power than required at the maximum power load identified in table 3 and has a 40% decreased mass over the complete 9U system. The reduction in launch mass and cost will therefore offset the increased initial cost of the panels and it is therefore recommended that the GOMSpace panels are used.

2. Power Storage

To conduct night operations, the system must possess enough battery power to run all systems while in the Earth’s shadow. The time of eclipse ($t_e$) is calculated via equation 18 to be 36.04 mins, full equation explanation is provided in the full report. Due to a 20W peak sustained power being required, the system will require 12 Wh of energy ($E$) without being recharged through the 36-minute eclipse phase of its orbit according to equation 20. To provide adequate margin of error, a 20Wh COTS system can be incorporated into the satellite. A battery solution that fits these requirements while also incorporating a power supply bus is the EnduroSat EPS I Plus, which contains a total power storage capacity of 20.4Wh at 292g of mass.

H. Thermal Control

The thermal design of a satellite is essential to ensure sustained operation within the space environment. The objective of the thermal subsystem is to keep every component within their operating temperature ranges. Measurements taken from the International Space Station demonstrate sun facing surfaces reaching 121°C with dark sides being -157°C. For the purposes of this mission, the utilisation of a passive thermal control system would be ideal in reducing cost and complexity. A complete analysis would require detailed simulation to validate and lies outside the scope of this study.

IV. Orbital Parameters

A. Applicable Orbit Types

For the proposed system’s desired purpose, a low inclination, low earth orbit is desirable. If an inclination of 27.5° is chosen, a band of coverage wrapping around the globe below Brisbane, Australia and above Taiwan can be observed, as shown in figure 7. Launch to this orbit also benefits from comparatively lower fuel requirements due to a lower magnitude of plane change, further efficiency can be achieved if launched domestically.

B. Orbital Height Selection

The largest hazard while operating satellites in low earth orbit is the magnitude of the atmospheric drag constantly acting upon the constellation. Atmospheric drag is dependent on surface area, mass, orbital velocity, drag coefficient and atmospheric density at its given altitude. A study conducted by the Australian Centre for Space Engineering Research at UNSW, Kensington demonstrates that the estimated lifetime for such a system in an unassisted state

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20 (GOMSpace, 2018)
21 (EnduroSat, 2019)
22 (ISIS, 2013)
23 (EnduroSat, 2018)
24 (Price & Dr. Phillips, 2019)
would be 600 days or 1.64 years according to figure 8. Therefore, to meet the program lifetime requirements at such a low altitude, a propulsion system is required as discussed in section III.D.

C. Constellation Parameters
To construct, visualize and iterate constellation designs, the SaVi and Geomview packages were utilized in a Linux environment. The constellation will be of a Ballard Rosette (Walker delta) design constrained to inclinations of 27.5°, an altitude of 400km, and the 58.1° angle of view calculated in section III.A.6. Footprint is outlined as yellow circles with areas of coverage overlapping being a darker orange.

Figure 9 shows a less than 95% coverage, but also possesses a lower satellite count compared to other orbits, this reduces overall program cost significantly. This constellation configuration has the 16 planes of satellites phased in such a manner that the 27.5° latitude band around the globe is scanned completely every three minutes, the ground coverage over this time period is highlighted green. This infers that at most, a three-minute delay between launch and detection will occur. From a ground station perspective, there will constantly be 10 to 15 satellites in view at any given point in time within the 27.5 latitude band, allowing great flexibility in ground station placement.

V. System Summary & Cost

A. Development Cost
This cost accounts for the further detailed technical work required to make such a system operational. It has been estimated through using the salaries of a team of 5 engineers on a $100,000AUD annually, working for 3 years, costing $1.5M AUD in for detailed design and analysis work. A further $1.5M AUD has been estimated as miscellaneous costs being required by this team to conduct project work. This includes costs to buy and test components, conduct small scale laboratory testing and any specialised design and analysis software.

B. Construction Cost
This accounts for the costs involved with the manufacturing of the minimum 256 units required in orbit to complete the constellation. This will be estimated through the expectation that all satellites will be constructed by the end of a two-year time frame. Full rate production can be estimated as producing a cube-satellite unit each week using a team of four engineers. Therefore, to produce 256 units over 90 weeks (accounting for leave and public holidays) only three production teams will be required to complete the constellation over two years. At $100,000 AUD annual salary per engineer, being approximately $2.4M AUD in wages. A further $7.6M AUD is allocated to either the refurbishment or construction of an assembly facility and all associated equipment.

C. Test & Evaluation Cost
These costs are associated with the verification of the final design, the flight qualification of any space hardware which has not been certified (primarily the optics system) and testing of complete cube-satellite units to ensure functionality. Quality assurance on cube-satellites units should be conducted through temperature, vibrational, radiation and pressure testing. As values for this type of testing are not readily available for a similar 3U system, a placeholder estimate of $10M AUD has been allocated.

25 (Qiao, et al., n.d.)
D. Launch Costs

New Zealand based Rocket Lab has achieved successful commercial launches with its Electron rocket, costing US$6M per launch while boosting 225kg to LEO\textsuperscript{26}. To complete the constellation completely, four Electron rockets would be required, each carrying 64 satellites to completely furnish four orbital planes. This would cost $24M AUD. A full analysis of other launch providers can be found in the full report.

E. Ground Station

Due to the selection of VHF/UHF communication bands for the constellation, commercial systems can be used. ISIS offers a complete ground station kit with steerable VHF/UHF antennas for $100,000AUD\textsuperscript{27}. To attain constant constellation communication and redundancy, a minimum of 5, but ideally 10 ground stations ($1M AUD) should be established. To account for additional unforeseen expenses and any additional infrastructure required, a total ground station cost of $5M AUD is estimated. Ongoing manning costs to maintain and observe the constellation have not been included for the purpose of this analysis.

F. Summary

<table>
<thead>
<tr>
<th>Component</th>
<th>Brand</th>
<th>Model</th>
<th>Notes</th>
<th>Supplier Cost (AUD)</th>
<th>AU Cost</th>
<th>Mass (kg)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Chassis</td>
<td>EnduroSat</td>
<td>Structure II</td>
<td>3U</td>
<td>$2,800</td>
<td>$4,500</td>
<td>285g</td>
</tr>
<tr>
<td>Imaging</td>
<td>Raptor Photonics</td>
<td>DWE, 1280 SWIR 4x4 Array (estimated)</td>
<td></td>
<td>$40,000</td>
<td>1000g</td>
<td></td>
</tr>
<tr>
<td>Processors</td>
<td>ISIS</td>
<td>OBC</td>
<td>System placeholder pending analysis</td>
<td>$4,400</td>
<td>$7,500</td>
<td>94g</td>
</tr>
<tr>
<td>Communications</td>
<td>ISIS</td>
<td>Full Duplex Transceiver</td>
<td></td>
<td>$18,500</td>
<td>$14,000</td>
<td>75g</td>
</tr>
<tr>
<td>Antenna</td>
<td>EnduroSat</td>
<td>UHF Antenna (mounted, incorporates 2 solar cells</td>
<td>3,000</td>
<td>$5,000</td>
<td>85g</td>
<td></td>
</tr>
<tr>
<td>ACS</td>
<td>NSS</td>
<td>NACS-001</td>
<td>Includes GPS (estimated)</td>
<td>$10,000</td>
<td>150g</td>
<td></td>
</tr>
<tr>
<td>Reaction Wheel</td>
<td>Nano Avionics</td>
<td>4RWO</td>
<td>Redundant 3-axis precision control</td>
<td>$8,000</td>
<td>$13,500</td>
<td>700g</td>
</tr>
<tr>
<td>Attitude Sensors</td>
<td>CubeSpace</td>
<td>CubeSense</td>
<td>180 deg FOV, Sun &amp; Earth sensors</td>
<td>$4,800</td>
<td>$8,000</td>
<td>80g</td>
</tr>
<tr>
<td>Thermal Rig</td>
<td></td>
<td></td>
<td>Mass and cost estimated</td>
<td>$5,000</td>
<td>200g</td>
<td></td>
</tr>
<tr>
<td>Power Generation</td>
<td>GOMSpace</td>
<td>NanoPowerP110</td>
<td>9U worth of solar panels</td>
<td>$20,000</td>
<td>234g</td>
<td></td>
</tr>
<tr>
<td>Power Storage</td>
<td>EnduroSat</td>
<td>EPS 1 Plus</td>
<td>Integrates power supply board</td>
<td>$3,000</td>
<td>$5,500</td>
<td>292g</td>
</tr>
<tr>
<td>Propulsion</td>
<td>Accion</td>
<td>Tile 500</td>
<td>Capillary ion drive, Cost estimated</td>
<td>$15,000</td>
<td>100g</td>
<td></td>
</tr>
<tr>
<td>Misc. Cables</td>
<td></td>
<td>Misc.</td>
<td>Estimated</td>
<td>$2,000</td>
<td>100g</td>
<td></td>
</tr>
</tbody>
</table>

| Unit Total | $150,000 | 3295g |
| Constellation Total (526 units) | $38,400,000 | 843kg |

Table 5. Program cost breakdown

VI. Conclusions

Throughout this feasibility study thus far, it appears that a low earth orbit cube-satellite constellation to detect hypersonic vehicles is feasible within current technical and physical limits. This has been observed through the macroscopic analysis of HGV characteristics to obtain optimal observation parameters, allowing the requirements, comparison and component selection of a cube-satellite system to be conducted. The constellation will require at least 256 units to function adequately costing an estimated $150M AUD each. The total program cost is estimated to be under $100M AUD, however, this could vary significantly due to unexpected costs. If this system was to be pursued by the RAAF, it would increase Australian expertise in the space sector and provide a considerable boost to the Australian space industry, particularly if domestic or regional launch and component suppliers are utilised.

VII. Recommendations

If this line of investigation should continue, it is recommended that the sensing suite is re-analysed, with a more detailed trade-off analysis between radar systems and imaging systems being conducted. Scaled-down testing of the selected infrared sensor in hypersonic wind tunnels should be conducted to determine core viability of the system. While all other systems explored in this feasibility study would need further analysis, areas requiring particular attention include a detailed analysis of the software required (with accompanying on board computer), thermal management simulations and a long-term space survivability study of the system.

\textsuperscript{26} (Mann, 2017)

\textsuperscript{27} (ISIS, 2019)
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Final Project Report 2018, UNSW Canberra at ADFA
Appendix A – Equations

1. \[ B_\lambda(\lambda, T) = \frac{2pc^2}{k^3} \frac{1}{e^{\frac{k\lambda}{kT}} - 1} \]

2. \[ L = \int_{e^{v_{Lower}}}^{e^{v_{Upper}}} \frac{x}{e^x - 1} \, dx \text{ where: } x = \frac{mv}{kBT} \]

3. \[ f = \frac{d_p}{A} \]

4. \[ A = \frac{d_p}{f} \]

5. \[ s = \sqrt{(r_n d_p)^2 + (r_v d_v)^2} \]

6. \[ \alpha = 2\tan^{-1} \frac{s}{2f} \]

7. \[ A = \left[ 2h \tan \left( \frac{\alpha}{2} \right) \right]^2 \]

8. \[ \theta = 1.22 \frac{\lambda}{\rho} \approx 1.22 \frac{\lambda}{\rho} \rightarrow f \approx \frac{fD}{1.22} \]

9. \[ FSPL(dB) = 20 \log_{10} (d) + 20 \log_{10} (v) + 20 \log_{10} \left( \frac{4\pi}{c} \right) - G_{Tx} - G_{Rx} \]

10. \[ v = \sqrt{\frac{\mu e}{e + h}} = \sqrt{3.986e10^5} = 7.67 \text{ km/s} \]

11. \[ F_d = \frac{c_d \rho v^2}{2m} = \frac{2.2 \times 0.01 \times 2.36e10^{-12} \times 7670^2}{2 \times 4} = 3.82e10^{-7} N \]

12. \[ \Delta v = l_{sp} g_0 \ln \left( \frac{m_a}{m_f} \right) = 1717 \times 9.81 \times \ln \left( \frac{4}{3.995} \right) = 21.06 \frac{m}{s} \]

13. \[ F = ma \rightarrow a = \frac{F}{m} = \frac{3.82e10^{-7}}{4} = 9.55e10^{-8} \frac{m}{s^2} \]

14. \[ t = \frac{\Delta v}{a} = \frac{21.06}{9.55e10^{-8}} = 2.2e10^8 \text{ s} = 6.98 \text{ years} \]

15. \[ t_p = 2\pi \sqrt{\frac{(e_r+h)^3}{\mu}} = 2\pi \sqrt{\left( \frac{6371+400}{3.986e10^5} \right)^3} = 92.4 \text{ minutes} \]

16. \[ \theta_p = \sin^{-1} \left( \frac{e_r+h}{e_r} \right) = \sin^{-1} \left( \frac{6371}{6371+400} \right) = 70.2^\circ \]

17. \[ t_e = t_p \left( \frac{2\alpha}{360} \right) = 36.04 \text{ minutes} \]

18. \[ t_s = t_p - t_e = 92.4 - 36.04 = 56.37 \text{ minutes} \]

19. \[ E = P \frac{t}{60} = 20 \times 0.6 = 12 Wh \]