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I am submitting herewith a thesis written by Michael Anthony Kamp entitled “Development and Testing of a Self-Contained, Portable Instrumentation System for a Fighter Pilot Helmet.” I have examined the final electronic copy of this thesis for form and content and recommend that it be accepted in partial fulfillment of the requirements for the degree of Master of Science, with a major in Aviation Systems.

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Development and Testing of a Self-Contained, Portable Instrumentation System for a Fighter Pilot Helmet

A Thesis
Presented for the
Master of Science Degree
The University of Tennessee, Knoxville

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Abstract

A self-contained, portable, inertial and positional measurement system was developed and tested for an HGU-55 model fighter pilot helmet. The system, designated the Portable Helmet Instrumentation System (PHIS), demonstrated the recording of accelerations and rotational rates experienced by the human head in a flight environment. A compact, self-contained, “knee-board” sized computer recorded these accelerations and rotational rates during flight. The present research presents the results of a limited evaluation of this helmet-mounted instrumentation system flown in an Extra 300 fully aerobatic aircraft. The accuracy of the helmet-mounted, inertial head tracker system was compared to the aircraft-mounted referenced system. The ability of the Portable Helmet Instrumentation System to record position, orientation and inertial information in ground and flight conditions was evaluated. The capability of the Portable Helmet Instrumentation System to provide position, orientation and inertial information with sufficient fidelity was evaluated. The concepts demonstrated in this system are: 1) calibration of the inertial sensing element without external equipment 2) the use of differential inertial sensing equipment to remove the accelerations and rotational rates of a moving vehicle from the pilot’s head-tracking measurements 3) the determination of three-dimensional position and orientation from three corresponding points using a range sensor. The range sensor did not operate as planned. The helmet only managed to remain within the range sensor’s field of view for 37% of flight time. Vertical accelerations showed the greatest correlation when comparing helmet measurements to aircraft measurements. The PHIS operated well during level flight.

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Nomenclature

$\mathbf{a}_{p,k}$	=	k-th acceleration vector measurement in platform frame
$\mathbf{a}_{s,k}$	=	k-th acceleration vector measurement in sensor frame
$\mathbf{a}_{x,AHRS}$	=	x acceleration of aircraft
$\mathbf{a}_{x,cam}$	=	acceleration of helmet in x-axis defined by camera
$\mathbf{a}_{x,gyrocube}$	=	raw x acceleration from Gyrocube (G's)
a_{xc}	=	x acceleration of helmet measured from Gyrocube, corrected to CG (G's)
a_{xx}	=	x-displacement of the x-axis accelerometer, in feet
a_{xy}	=	y-displacement of the x-axis accelerometer, in feet
a_{xz}	=	z-displacement of the x-axis accelerometer, in feet
$\mathbf{a}_{y,AHRS}$	=	y acceleration of aircraft
$\mathbf{a}_{y,cam}$	=	acceleration of helmet in Y-axis defined by camera
$\mathbf{a}_{y,gyrocube}$	=	raw y acceleration from Gyrocube (G's)
a_{yc}	=	y acceleration of helmet measured from Gyrocube, corrected to CG (G's)
a_{yx}	=	x-displacement of the y-axis accelerometer, in feet
a_{yy}	=	y-displacement of the y-axis accelerometer, in feet
a_{yz}	=	z-displacement of the y-axis accelerometer, in feet
$\mathbf{a}_{z,AHRS}$	=	z acceleration of aircraft
$\mathbf{a}_{z,cam}$	=	acceleration of helmet in Z-axis defined by camera
$\mathbf{a}_{z,gyrocube}$	=	raw z acceleration from Gyrocube (G's)
a_{zc}	=	z acceleration of helmet measured from Gyrocube, corrected to CG (G's)
a_{zx}	=	x-displacement of the z-axis accelerometer, in feet
a_{zy}	=	y-displacement of the z-axis accelerometer, in feet
a_{zz}	=	z-displacement of the z-axis accelerometer, in feet
b_a	=	accelerometer bias
C_1	=	cosine matrix for rotation ψ about z-axis

C_2	=	cosine matrix for rotation θ about y-axis
C_3	=	cosine matrix for rotation ϕ about x-axis
C_n^b	=	cosine matrix transformation from reference to body axes
C_b^n	=	inverse cosine matrix transformation from body to reference axes
C_ϕ	=	plane rotation ϕ using Euler angles for reference to body frame
C_θ	=	plane rotation θ using Euler angles for reference to body frame
d_{01}	=	distance between image points i_0 and i_1
d_{02}	=	distance between image points i_0 and i_2
d_{12}	=	distance between image points i_1 and i_2
E	=	diagonally dominant correction matrix
E_g	=	gyroscope diagonally dominant correction matrix
F_b	=	body frame
F_r	=	reference frame
H_1	=	altitude edge of first right triangle of model point orthographic projection
h_1	=	altitude edge of first right triangle of image point orthographic projection
H_2	=	altitude edge of second right triangle of model point orthographic projection
h_2	=	altitude edge of second right triangle of image point orthographic projection
i_0	=	first image point of the image plane
i_1	=	second image point of the image plane
i_2	=	third image point of the image plane
m_0	=	first model point in three-dimensional space
m_1	=	second model point in three-dimensional space
m_2	=	third model point in three-dimensional space
M	=	misalignment matrix
M_g	=	gyroscope full misalignment correction matrix
p	=	center of projection point
p	=	roll angular velocity, radians/s

$\dot{p}$	=	angular roll acceleration, radians/s ²
q	=	pitch angular velocity, radians/s
$\dot{q}$	=	angular pitch acceleration, radians/s ²
q_k	=	quaternion for time step k
r	=	yaw angular velocity, radians/s
$\dot{r}$	=	angular yaw acceleration, radians/s ²
R_{01}	=	distance between model points m_0 and m_1
R_{02}	=	distance between model points m_0 and m_2
R_{12}	=	distance between model points m_1 and m_2
s	=	biquadratic scale factor for projection
S^*	=	sensitivity matrix
S_g^*	=	gyroscope sensitivity matrix
u_a	=	static accelerometer gravity vector
u_g	=	computed gravity vector
$V_{x,cam}$	=	velocity of helmet in X-axis defined by camera
$V_{xc,helmet-GC}$	=	velocity of helmet along X-axis, integration from $a_{xc, helmet-GC}$
$V_{y,cam}$	=	velocity of helmet in Y-axis defined by camera
$V_{yc,helmet-GC}$	=	velocity of helmet along Y-axis, integration from $a_{yc, helmet-GC}$
$V_{z,cam}$	=	velocity of helmet in Z-axis defined by camera
$V_{zc,helmet-GC}$	=	velocity of helmet along Z-axis, integration from $a_{zc, helmet-GC}$
w	=	offset term in direction normal to image plane
X_{cam}	=	X-position of camera location in relation to aircraft datum
x_{CG}	=	x location of body center of gravity, in.
$x_{c,helmet-GC}$	=	X position of helmet center, integration from $V_{xc, helmet-GC}$ in relation to camera lens
x_{loc}	=	vector of x locations of the X, Y, and Z accelerometers in relation to body c.g., in.
Y_{cam}	=	Y-position of camera location in relation to aircraft datum
y_{CG}	=	y location of body center of gravity, in.

$y_{c,helmet-GC}$	=	Y position of helmet center, integration from $V_{yc, helmet-GC}$ in relation to camera lens
y_{loc}	=	vector of y locations of the X, Y, and Z accelerometers in relation to body c.g., in.
Z_{cam}	=	Z-position of camera location in relation to aircraft datum
z_{CG}	=	z location of body center of gravity, in.
$z_{c,helmet-GC}$	=	Z position of helmet center, integration from $V_{zc, helmet-GC}$ in relation to camera lens
z_{loc}	=	vector of z locations of the X, Y, and Z accelerometers in relation to body c.g., in.
Δt	=	time step (seconds)
Δx_{cam}	=	X position of helmet defined by camera in relation to camera lens
Δy_{cam}	=	Y position of helmet defined by camera in relation to camera lens
Δz_{cam}	=	Z position of helmet defined by camera in relation to camera lens
θ	=	Euler rotation angle pitch
θ_{AHRS}	=	aircraft pitch angle
θ_{cam}	=	pitch angle of helmet defined by camera
$\theta_{helmet-GC}$	=	pitch angle of helmet, integrated from $\dot{\theta}_{helmet-GC}$
$\dot{\theta}_{AHRS}$	=	pitch rate of aircraft
$\dot{\theta}_{cam}$	=	pitch angular velocity of helmet defined by camera
$\dot{\theta}_{gyrocube}$	=	raw pitch angular velocity from Gyrocube (degrees/second)
$\dot{\theta}_{helmet-GC}$	=	pitch angular velocity from Gyrocube (degrees/second)
ϕ	=	Euler rotation angle roll
ϕ_{AHRS}	=	aircraft roll angle
ϕ_{cam}	=	roll angle of helmet defined by camera
$\phi_{helmet-GC}$	=	roll angle of helmet, integrated from $\dot{\phi}_{helmet-GC}$
$\dot{\phi}_{AHRS}$	=	roll rate of aircraft
$\dot{\phi}_{cam}$	=	roll angular velocity of helmet defined by camera
$\dot{\phi}_{gyrocube}$	=	raw roll angular velocity from Gyrocube (degrees/second)
$\dot{\phi}_{helmet-GC}$	=	roll angular velocity from Gyrocube (degrees/second)
ψ	=	Euler rotation angle yaw

ψ_{AHRS}	=	aircraft yaw angle
ψ_{cam}	=	yaw angle of helmet defined by camera
$\psi_{helmet-GC}$	=	yaw angle of helmet, integrated from $\dot{\psi}_{helmet-GC}$
$\dot{\psi}_{AHRS}$	=	yaw rate of aircraft
$\dot{\psi}_{cam}$	=	yaw angular velocity of helmet defined by camera
$\dot{\psi}_{gyrocube}$	=	raw yaw angular velocity from Gyrocube (degrees/second)
$\dot{\psi}_{helmet-GC}$	=	yaw angular velocity from Gyrocube (degrees/second)
$\omega_{b/r}^b$	=	angular velocity vector reference frame to body transformation

Acronyms

3D	=	Three Dimensional
6DOF	=	Six Degrees of Freedom
AHRS	=	Attitude and Heading Reference System
API	=	Application Programming Interface
CG	=	Center of Gravity
CPU	=	Central Processing Unit
DAQ	=	Data Acquisition
DAQ-MX	=	Data Acquisition Multi-Threaded Driver
DAS	=	Data Acquisition System
DASH	=	Display and Sight System
EFIS	=	Electronic Flight Instrumentation System
ENU	=	East-North=Up system
FTE	=	Flight Test Engineer
GLOC	=	Gravity Induced Loss of Consciousness
GRT	=	Grand Rapids Technology
GUI	=	Graphical User Interface
HGU	=	Head Gear Unit
HMCS	=	Head Mounted Cueing System
HMD	=	Head Mounted Display
HMDS	=	Head Mounted Display System
HMST	=	Head Mounted Sensory Technology
HUD	=	Heads Up Display
IMU	=	Inertial Measurement Unit
IPP	=	Integrated Performance Primitives
IR	=	Infra-red
JHMCS	=	Joint Helmet Mounted Cueing System

LCD	=	Liquid Crystal Display
LED	=	Light-Emitting Diode
LOS	=	Line of Sight
MEMS	=	Micro Electrical Mechanical System
NED	=	North-East-Down system
NI	=	National Instruments
NiMH	=	Nickel Metal Hydride
OpenCV	=	Open Computer Vision
ORD	=	Objectives and Requirements Document
OS	=	Operating System
PDA	=	Personal Data Assistant
PHIS	=	Portable Helmet Instrumentation System
RAF	=	Royal Air Force
TP	=	Test Pilot
UMPC	=	Ultra Mobile Personal Computer
USB	=	Universal Serial Bus
UTSI	=	University of Tennessee Space Institute
VI	=	Virtual Instrument
VSI	=	Vision Systems International

1. Introduction

FOR an aviator, maintaining an overall level of physical well-being and mental alertness is crucial for the operation of an aircraft. The human head is a large factor when regarding these two requirements. Loss of blood to the head caused by G-loading can lead to GLOC (Gravity Induced Loss of Consciousness). Impacts to the head have the possibility of long term internal injuries such as brain damage. The human head also houses two of the most important organic sensors for navigation, both vision and the inner ear. The mouth and the ears provide the ability to communicate and a sense of hearing can also provide audio clues to impending or abnormal events. For human beings, investing in the protection of the head is also an investment in all of its individual and important components.

A helmet is a form of protective gear worn on the head to protect it from injuries. The oldest use of helmets was by Ancient Greek soldiers, who wore thick leather or bronze helmets for protection from weapon blows. Since then, the use of helmets has proliferated throughout military use. Though the materials involved in their construction have evolved over time, the design elements have not changed much.

In the early age of aviation up to World War II, pilots did not wear hard helmets. Instead they wore headgear to insulate heat to the head and to provide minor protection from bumps. Aircraft that would fly at altitudes of 10,000 to 30,000 feet necessitated pilots to wear headgear that affixed an oxygen mask when required. It wasn't until 1947, when the first United States Army Air Force helmet, the H-1, and the United States Navy helmet, the P-1, for pilots was manufactured by General Textile Mills, today known as Gentex [1].

Fighter helmets are special types of helmets designed to be worn by military aircraft pilots. Typically they include a visor for protection from sunlight and from windblast in case of an ejection, an oxygen mask for protection against hypoxia in high altitude flight, and communications equipment that can interface with the aircraft's radios and intercom. A typical example is shown in Figure 1. In civilian aviation, there are no standards for the certification of helmets for use as an item of aviation life support equipment. Many civilian pilots who require protective headgear normally adopt military flight helmets.

A new revolution in aviation headgear is occurring. Instrumented helmets in the military are becoming more common, with the driving force being Head-Mounted Displays (HMD). With a HMD, a pilot can now aim weapons with the head, receive mission critical information and receive self-protection prompts, all by looking through the

helmet visor [2]. Vision Systems International (VSI), San Jose, California has two HMD models in production: the DASH (Display and Sight System Helmet) and the JHMCS (Joint Helmet Mounted Cueing System). The F-35 Head-Mounted Display System (HMDS), currently in development, will be integrated into the F-35 Lightning II [2]. The F-35 HMDS features a virtual heads-up-display (HUD), magnetic tracker, and integrated Line of Sight (LOS) cueing, integrated de-brief camera, low weight, improved center of gravity, and is quoted as “ejection safe”[3]. The F-35 HMDS is of significant importance because it represents the future standard for helmet mounted displays. It features advanced head tracking capability with near-zero latency, a virtual heads-up display, and full day and night imagery even for extreme off-axis targeting and cueing[4]. The first flight test of the F-35 HMDS was in January of 2007. In October 2007, the HMDS was verified for comfort, fit and stability under high-G conditions by The Royal Air Force Center for Aviation Medicine at the Ministry of Defense, Boscombe Down, an aircraft testing site located in Wiltshire England [5]. The RAF Center flew two modified BAE Hawk T Mk1s in flight regimes ranging from -2G to +9.5G, and test pilots assessed that the F-35 HMDS showed significant promise in comfort and stability [6].

Another company, Ascension Technology, has produced several 3-D head tracking devices. Ascension’s most recent head-tracker, the HY-BIRD shown in Figure 2, combines optical and inertial references to determine the line-of-sight angle of a pilot’s head. It is claimed as the “first [head-tracker] with 360 [degree tracking] without occlusions” [7]. Occlusions are breaks in tracking data caused by the optical references moving out of sight of the optical tracker. This is usually due to an extreme orientation or position of the optical head-tracker. An inertial sensor can provide inertial data to estimate head position when optical references are unavailable. The laserBIRD, an optical tracker, provides the optical references for the HY-BIRD, and inertial measurements are taken by a separate instrument for occlusion removal. Another Ascension product, the phaserBIRD, is a six degree of freedom (6DOF) system with a measurement rate accuracy of 0.1° and an update rate of over 300 Hz. The phaserBIRD has immunity to ambient light and scattering of magnetic and electric field energy from the helmet or cockpit [8].

HMDs and HMCS are both under the umbrella of Head-Mounted Sensory Technology (HMST). There are several unique issues with helmet-mounted sensory technology (HMST):

- These systems create additional weight on the human head, which may lead to stress and strain on pilots and subsequent degradation of pilot performance.
- There is lag present in some HMST systems.

- There is significant proliferation of additional equipment into the cabin environment for these systems. Most HMST require extensive installations. Hence, portability or interchangeability between airframes is lost.
- Aircraft must be initially designed to include these new systems or be upgraded at significant cost.

The introduction of helmet mounted technology is evolving the way fighter aircraft operate, and these systems are changing the bio-dynamics of fighter aircraft cockpits. Since these systems are bulkier and heavier, performing high-G fighter maneuvers or rocket-boosted ejections have a higher probability of pilot injury or fatality. The latest instrumented helmets weigh approximately 4.5 to 6.0 pounds, about 50% heavier than their un-instrumented predecessors [9]. When in a +9-G maneuver, the 4.5 pound helmet will weigh 40.5 pounds. To compensate for the increasing weight of fighter pilot helmets, neck muscle exercise machines and strength-training regimen for fighter pilots may be required. In September of 2008, Survival Innovations Incorporated, Arden, North Carolina received a six million dollar, Air Force contract to develop a “unique head-and-neck restraint system for fighter pilots” that will resolve the growing problem of heavier loads on the pilot’s neck during ejection due to the latest instrumented helmets [10].

Recently the United States Army has been investigating soldier head trauma. Blasts from explosives or grenades may not be fatal to a soldier on the battlefield but can result in traumatic brain injury. To understand this kind of injury, accelerometers are being embedded into a soldier’s helmet. During a blast event, accelerations from shock waves are recorded [11].

Sectors of sports medicine are using similar technology to collect data on the severity and location of head impacts during a sports event. Permanent brain damage can result from head concussions, but the damage can go undetected. Using these systems in the military as well as in sports will alert people to the possibility of severe injury from a head impact so proper care can be administered to prevent further damage [12].

Similar to the premise of safety and health sought by the helmet systems of the United States Army, data collected from a helmet-mounted device in a fighter cockpit environment could further knowledge in improving safety devices for fighter pilots. By gaining a qualitative measure of the dynamics experienced, an improvement and comparison of helmet equipment and piloting techniques may be achieved. A helmet data instrumentation system could also be used in ejection simulations to further knowledge of the physiological effects of ejection [13]. Other design goals may include accurately measuring line-of-sight (LOS) angles. This capability could be used to

permit precision guidance of munitions in a military application, enable intra-cockpit cueing such as an application of 3D Audio, and to maximize situational awareness. Ideally, the system should also be portable between different types of aircraft.

2. Theoretical Background of PHIS

2.1. Reference Frames and Sign Conventions

For motion and navigation, it is necessary to define frames of reference. Reference frames discussed in the present paper assume orthogonal axes using right-handed conventions. For the right-hand convention, the right-hand thumb is pointed in the direction along a positive axis, and the fingers of the same hand curl in the direction of the positive rotation.

The Inertial Frame (i-frame) The origin of this reference frame is fixed to the center of the Earth. Its axes are non-rotating with respect to the fixed stars. The z-axis is parallel to the spin axis of the Earth about the Sun and the x-axis is pointing towards the vernal equinox [14]. The vernal equinox is the point where the celestial equator and the ecliptic of the Earth intersect. The y-axis is determined by completing a right-handed orthogonal frame. Figure 3 shows the inertial frame and the orientation of its axes in relation to the vernal equinox and spin axis of the earth.

The Earth Frame (e-frame) Similar to the inertial frame, the earth frame has its origin at the center of mass of the Earth. The axes are fixed with respect to the Earth itself. The x-axis of the Earth frame points to the mean meridian of Greenwich and the z-axis is parallel to the spin axis of the Earth [15]. The y-axis is orthogonal to the other two axes using right-hand convention. The Earth axes are assumed to be Inertial Axes for most cases in the technical area of aircraft systems identification [16]. This assumption disregards the motion of the Earth relative to the stars.

The Navigational Frame (n-frame) This is a local geographic frame. The origin is located in the origin of the sensor frame. Two coordinate systems exist: the local north-east-down system and the local east-north-up system. Both the north-east-down (NED) system and east-north-up (ENU) system is formed from a plane tangent to the

Earth's surface. For the north-east-down (NED) system, the x-axis is pointing towards the North Pole, the z-axis points downward toward the center of the earth and the y-axis completes a right-handed orthogonal frame. The x-axis is pointed towards the east, the y-axis points towards the North Pole and the z-axis points upward away from the center of the earth[15]. The ENU system is a right-handed orthogonal frame. Using the ENU system allows for altitude readings to increase in the upward z-axis. The NED system considers right handed turns to be positive with respect to the downward z-axis (yaw), and the NED axis system coincides with vehicle-fixed roll, pitch and heading angles when a vehicle is level and pointing to the north. The NED system is the most prevalent n-frame in use for research results [15]. Figure 4 illustrates both the Earth frame and Navigational frame. Axes in this figure with the superscript “e” represent the Earth axes, and axes with the superscript “n” represent the Navigational axes.

The Body Frame (b-frame) The origin of the Body Frame is at the vehicle center of gravity. To serve as an example, if the vehicle is represented as an aircraft, the positive x-axis is pointing forward through its nose. The positive y-axis is pointing out the right wing of the aircraft, and the positive z-axis points downward through the underside of the aircraft, completing an orthogonal set of axes. The body frame is aligned with the roll, pitch and yaw axes of the vehicle [16]. These body axes are fixed to the vehicle body. Figure 5 shows a representation of an aircraft fixed with a body frame. In regards to angular velocities, use of the right-hand rule is the standard sign convention.

2.2. Coordinate Rotations

For coordinate frames, all axis sets are right-handed with positive rotations about each axis taken in a clockwise direction. When defining the attitude of a body with respect to a coordinate reference frame, there are several mathematical representations that can be used. The Euler angle representation is possibly one of the simplest techniques available. When using Euler angles, a transformation from one coordinate frame to another is defined by three successive rotations about different axes that are taken in turn. The three angles of rotation correspond to a set of angles which would be measured by a set of mechanical gyroscopes. These gyroscopes are mounted to a stable body or element, and the axes of the stable element represent the reference frame [17]. It is desired to collect these parameters as measurements of turn rates provided by the gyroscopes and storing them in a computer to be updated as the body or element rotates.

The angles ϕ , θ , and ψ are referred to as the Euler rotation angles. A transformation from reference axes to a new coordinate frame can be expressed as follows: 1) Rotation of angle ψ about the reference z-axis. 2) Rotation of angle θ about the new y-axis. 3) Rotation of angle ϕ about the new x-axis. This representation is very practical because the Euler angles correspond to the angles that could be determined by an angular measurement device. The three Euler rotations can be expressed mathematically as three separate cosine matrices [17].

Rotation ψ about z-axis:

$$C_1 = \begin{bmatrix} \cos \psi & \sin \psi & 0 \\ -\sin \psi & \cos \psi & 0 \\ 0 & 0 & 1 \end{bmatrix} \quad (1)$$

Rotation θ about y-axis:

$$C_2 = \begin{bmatrix} \cos \theta & 0 & -\sin \theta \\ 0 & 1 & 0 \\ \sin \theta & 0 & \cos \theta \end{bmatrix} \quad (2)$$

Rotation ϕ about x-axis:

$$C_3 = \begin{bmatrix} 1 & 0 & 0 \\ 0 & \cos \phi & \sin \phi \\ 0 & -\sin \phi & \cos \phi \end{bmatrix} \quad (3)$$

For a transformation from reference to body axes, the product of the three cosine matrices (1-3) is shown below.

$$C_n^b = C_3 C_2 C_1 \quad (4)$$

Similarly, an inverse transformation from body to reference axes can be found by the product of the three cosine matrices, each transposed [17]. Equation 7 is the direction cosine matrix expressed in terms of Euler angles.

$$C_b^n = C_n^{bT} = C_1^T C_2^T C_3^T \quad (5)$$

$$C_b^n = \begin{bmatrix} \cos \psi & -\sin \psi & 0 \\ \sin \psi & \cos \psi & 0 \\ 0 & 0 & 1 \end{bmatrix} \begin{bmatrix} \cos \theta & 0 & \sin \theta \\ 0 & 1 & 0 \\ -\sin \theta & 0 & \cos \theta \end{bmatrix} \begin{bmatrix} 1 & 0 & 0 \\ 0 & \cos \phi & -\sin \phi \\ 0 & \sin \phi & \cos \phi \end{bmatrix} \quad (6)$$

$$C_b^n = \begin{bmatrix} \cos \theta \cos \psi & -\cos \phi \sin \psi + \sin \phi \sin \theta \cos \psi & \sin \phi \sin \psi + \cos \phi \sin \theta \cos \psi \\ \cos \theta \sin \psi & \cos \phi \cos \psi + \sin \phi \sin \theta \sin \psi & -\sin \phi \cos \psi + \cos \phi \sin \theta \sin \psi \\ -\sin \theta & \sin \phi \cos \theta & \cos \phi \cos \theta \end{bmatrix} \quad (7)$$

It is important to note that the Euler angles are not unique for a given orientation. An example of this would be an aircraft performing a vertical loop. The pitch angle would be in a range of 180 degrees to -180 degrees, with the roll and yaw angles both zero throughout. An alternative representation would be to restrict the pitch angle between -90 degrees to +90 degrees. When the aircraft is nose-up in the vertical the pitch angle of the aircraft reaches +90 degrees, and the roll and yaw angles are changed abruptly by 180 degrees. As the aircraft continues the loop the pitch angle will begin to decrease, reaching zero pitch angle at the top of the loop. When the aircraft is completely

nose-down in the vertical at a pitch angle of -90 degrees, the roll and yaw angles change back to zero degrees. For this representation, the pitch angle is restricted to ± 90 degrees or $\pm \pi/2$ radians. The roll and yaw are considered as undefined during the abrupt transition at the ± 90 degree pitch angles [18].

2.3. Transformation from an Arbitrary Point of Body to the Center of Gravity of a Body

When a series of inertial measurements are taken from an arbitrary point on a body, it is desired to correct the translational accelerometer measurements to the body center of gravity. Rotation-based measurements are property of the rigid body independent of choice of coordinate system and therefore require no corrective transformation. The helmet in this paper is represented as a rigid body. Acceleration measurements are corrected from accelerometers located at the three-dimensional coordinate x_{loc} , y_{loc} , and z_{loc} to the body center of gravity located at x_{CG} , y_{CG} , and z_{CG} . If the accelerometers are oriented at an angle, then they must be corrected using an inverse Euler coordinate transformation as shown in equation 8.

$$\begin{bmatrix} x_2 \\ y_2 \\ z_2 \end{bmatrix} = \begin{bmatrix} \cos \theta & 0 & \sin \theta \\ 0 & 1 & 0 \\ -\sin \theta & 0 & \cos \theta \end{bmatrix} \begin{bmatrix} x_1 \\ y_1 \\ z_1 \end{bmatrix} \quad (8)$$

x_{loc} = vector of X positions of the X, Y, and Z accelerometers, in.

y_{loc} = vector of Y positions of the X, Y, and Z accelerometers, in.

z_{loc} = vector of Z positions of the X, Y, and Z accelerometers, in.

x_{CG} = X Center of Gravity position, in.

y_{CG} = Y Center of Gravity position, in.

z_{CG} = Z Center of Gravity position, in.

$[\ddot{p}, \ddot{q}, \ddot{r}]$ = Angular accelerations, radians/s²

$[p, q, r]$ = Angular velocities, radians/s

$$a_{xx} = \frac{-(x_{loc(1)} - x_{CG})}{12}; \quad a_{xy} = \frac{(y_{loc(1)} - y_{CG})}{12}; \quad a_{xz} = \frac{-(z_{loc(1)} - z_{CG})}{12} \quad (9)$$

$$a_{yx} = \frac{-(x_{loc(2)} - x_{CG})}{12}; \quad a_{yy} = \frac{(y_{loc(2)} - y_{CG})}{12}; \quad a_{yz} = \frac{-(z_{loc(2)} - z_{CG})}{12} \quad (10)$$

$$a_{zx} = \frac{-(x_{loc(3)} - x_{CG})}{12}; \quad a_{zy} = \frac{(y_{loc(3)} - y_{CG})}{12}; \quad a_{zz} = \frac{-(z_{loc(3)} - z_{CG})}{12} \quad (11)$$

Negative signs for the x and z displacements are used because the fuselage station and waterline positive directions are opposite to the x and z vehicle body axes. The term “waterline” refers to the vertical location of items on an aircraft. The term “fuselage station” refers to the longitudinal location of items on an aircraft. The fuselage station has an x-axis with positive direction facing the aft of the aircraft. The waterline positive direction is positive in the upward direction. The next step is to implement the position correction for the accelerometer measurements.

The X acceleration measurement, corrected to the C.G., in G's:

$$a_{xc} = \frac{(g \cdot a_{p,x} + (q^2 + r^2) \cdot a_{xx} - (pq - \dot{r}) \cdot a_{xy} - (pr + \dot{q}) \cdot a_{xz})}{g} \quad (12)$$

The Y acceleration measurement, corrected to the C.G., in G's:

$$a_{yc} = \frac{(g \cdot a_{p,y} + (p^2 + r^2) \cdot a_{yy} - (pq + \dot{r}) \cdot a_{yx} - (qr + \dot{p}) \cdot a_{yz})}{g} \quad (13)$$

Z acceleration measurement, corrected to the C.G., in G's:

$$a_{zc} = \frac{(g \cdot a_{p,z} + (q^2 + p^2) \cdot a_{zz} - (pr - \dot{q}) \cdot a_{zx} - (qr + \dot{p}) \cdot a_{zy})}{g} \quad (14)$$

2.4. Position and Orientation of Three Corresponding Points in Three-Dimensional Space

It is possible to determine the orientation and position of a model in three-dimensional space by performing model-based object recognition. This form of object recognition takes a minimized set of model points and matches them with image points to compute the orientation and position of the model in the image coordinate system [18]. The orientation and position of a three-dimensional object is computed from a two-dimensional image. A bi-quadratic equation is used for geometry-based, solution that is graphically interpreted.

A projection model, called weak-perspective projection, is used. This type of projection is an orthographic projection that is scaled to approximate perspective projection. The scaling assumes that all points on the three dimensional object are roughly the same distance from the observing sensor. An unmodified perspective projection example is shown in Figure 7. This perspective projection can be compared geometrically with the weak-perspective projection example shown in Figure 8. The effect of scaling is demonstrated in Figure 8. The benefit of using the weak-perspective projection is that the camera focal length and center point are not needed. For both

perspective and weak-perspective projections, a minimum number of three points are required to compute the orientation and position of a model. This does not however, provide an infinite number of pose solutions [18]. The difficulty is determining the orientation and position of the three points in space when provided with the corresponding three image points. Using the bi-quadratic computation, the solution provides a direct expression of the three matched three-dimensional model points, but in image coordinates. Ideally, there would be a low-level process looking for points that correspond to the three-dimensional model points that produces the two-dimensional points from the image. Referring to Figure 7, the model point capturing process is illustrated for a perspective projection. The concept shown for the perspective projection provides a clearer understanding of the perspective three-point problem that is necessary for utilizing the weak-perspective projection method. The three model points, m_0 , m_1 , and m_2 , which exist in three-dimensional space, are perceptively projected onto the image plane creating image points, i_0 , i_1 , and i_2 . These projections are created via lines that travel to a center of projection point, p . The side lengths and the angles of the tetrahedron provide the information necessary to determine the locations of m_0 , m_1 , and m_2 .

Now referring to Figure 8, the weak-perspective projection is shown using two tetrahedrons. The smaller geometry is a scaled-down version of the larger one. This figure shows the three image points, i_0 , i_1 , and i_2 , in the smaller tetrahedron. These three image points are located on the image plane. The model points, m_0 , m_1 , and m_2 , are projected orthographically onto a plane that contains m_0 and is parallel to the image plane. This projection is then scaled down by a factor “ s ” and projected again onto the image plane. Again, it is the side lengths of the tetrahedrons and the scale factor that provide the information necessary to determine the locations of m_0 , m_1 , and m_2 .

2.5. Computing the Weak-Perspective Solution

The side lengths from the geometry of the weak-perspective projection shown in Figure 8 are necessary to determine the three-dimensional pose of the model. The distances between the model points are represented as lengths R_{01} , R_{02} , and R_{12} . The corresponding distances between the image points are d_{01} , d_{02} , and d_{12} . Tao D. Alter derives the three-dimensional pose solution from the basic geometry of the weak-perspective three-point problem leading to a biquadratic in the scale factor “ s ” [18]. The parameters of the geometry in Figure 8 are provided computationally in equations (15-22).

$$s = \sqrt{\frac{b + \sqrt{b^2 - ac}}{a}} \quad (15)$$

$$(h_1, h_2) = \pm \left(\sqrt{(sR_{01})^2 - d_{01}^2}, \sigma \sqrt{(sR_{02})^2 - d_{02}^2} \right) \quad (16)$$

$$(H_1, H_2) = \frac{1}{s} (h_1, h_2) \quad (17)$$

$$(h_1, h_2) = \pm \left(\sqrt{(sR_{01})^2 - d_{01}^2}, \sigma \sqrt{(sR_{02})^2 - d_{02}^2} \right) \quad (18)$$

Where:

$$a = (R_{01} + R_{02} + R_{12})(-R_{01} + R_{02} + R_{12})(R_{01} - R_{02} + R_{12})(R_{01} + R_{02} - R_{12}) \quad (19)$$

$$b = d_{01}^2(-R_{01}^2 + R_{02}^2 + R_{12}^2) + d_{02}^2(R_{01}^2 - R_{02}^2 + R_{12}^2) + d_{12}^2(R_{01}^2 + R_{02}^2 - R_{12}^2) \quad (20)$$

$$c = (d_{01} + d_{02} + d_{12})(-d_{01} + d_{02} + d_{12})(d_{01} - d_{02} + d_{12})(d_{01} + d_{02} - d_{12}) \quad (21)$$

$$\sigma = \begin{cases} 1 & \text{if } d_{01}^2 + d_{02}^2 - d_{12}^2 \leq s^2(R_{01}^2 + R_{02}^2 - R_{12}^2) \\ -1 & \text{otherwise} \end{cases} \quad (22)$$

The solution to these equations (15-22) has a two-way ambiguity except when both h_1 and h_2 are zero. This is because there is a reflection about the plane parallel to the image plane. When h_1 and h_2 are both equal to zero, the model triangle is parallel to the image triangle. There is a need to account for this ambiguity when turning right and h_1 crosses zero. This can be done by preventing the sign change when one point is close to the mirror plane and the other is not.

The solution fails when the model triangle becomes a single line. This causes the value of “a” to equal zero and a solution may not exist. Given image points, $i_0 = (x_0, y_0)$, $i_1 = (x_1, y_1)$, and $i_2 = (x_2, y_2)$. These image point coordinates can be used to determine the three-dimensional locations of the model points in camera-centered coordinates [18].

$$m_0 = \frac{1}{s} (x_0, y_0, w) \quad (23)$$

$$m_1 = \frac{1}{s} (x_1, y_1, h_1 + w) \quad (24)$$

$$m_2 = \frac{1}{s} (x_2, y_2, h_2 + w) \quad (25)$$

An unknown offset in a direction normal to the image plane is represented as the variable “w.” Since a three-dimensional rigid transform is desired, it is computed from the three camera-centered coordinates of equations (23-25) and the original three model points. The offset term “w” drops out when computing the rotation of the model.

2.6. Matrix Kinematic Relationships for Rotation

It is necessary to determine the components of the angular velocity vector. This can be done with a given set of time-varying Euler angles that describe a rotating frame. To show this mathematically, let a coordinate system in a body frame F_b be relative to a system in reference frame F_r . The orientation of the system in body frame F_b is described by the roll, pitch and yaw sequence of Euler angles. These Euler angles also have derivatives $\dot{\phi}$, $\dot{\theta}$, and $\dot{\psi}$. Beginning at the reference frame F_r , using two intermediate frames with angular velocities that are the Euler angle rates, and using the additive property of angular velocity, the following relationship is obtained[19]:

$$\omega_{b/r}^b = \begin{bmatrix} \dot{\phi} \\ 0 \\ 0 \end{bmatrix} + C_\phi \left(\begin{bmatrix} 0 \\ \dot{\theta} \\ 0 \end{bmatrix} + C_\theta \begin{bmatrix} 0 \\ 0 \\ \dot{\psi} \end{bmatrix} \right) \quad (28)$$

C_ϕ and C_θ represent the plane rotations using the Euler angles for reference frame to body frame transformation. These plane rotations can be replaced with the cosine matrices in equations (2-3). After multiplying out the matrices, the final result is shown in equation 29.

$$\omega_{b/r}^b \equiv \begin{bmatrix} P \\ Q \\ R \end{bmatrix} = \begin{bmatrix} 1 & 0 & -\sin \theta \\ 0 & \cos \phi & \sin \phi \cos \theta \\ 0 & -\sin \phi & \cos \phi \cos \theta \end{bmatrix} \begin{bmatrix} \dot{\phi} \\ \dot{\theta} \\ \dot{\psi} \end{bmatrix} \quad (29)$$

P, Q and R are the standard symbols for roll, pitch and yaw rate components of the aircraft angular velocity vector, respectively. The inverse transformation for components of an angular velocity vector in the body frame to the reference frame is in equation 30.

$$\begin{bmatrix} \dot{\phi} \\ \dot{\theta} \\ \dot{\psi} \end{bmatrix} = \begin{bmatrix} 1 & \tan \theta \sin \phi & \tan \theta \cos \phi \\ 0 & \cos \phi & -\sin \phi \\ 0 & \sin \phi / \cos \theta & \cos \phi / \cos \theta \end{bmatrix} \begin{bmatrix} P \\ Q \\ R \end{bmatrix} \quad (30)$$

2.7. Second Order Discrete Time Integration Step Formula – Attitude Integration Algorithm

For a time step when attitude is unavailable, it is possible to utilize attitude information from a previous time step along with known rotational motion to determine the current attitude. Rotational motion may come from an

inertial sensing source. A second order discrete time integration step formula can use this available information to solve for current attitude. This form uses Euler angles and standard aeronautics conventions. ϕ , θ , and ψ represent roll, pitch and yaw respectively. The continuous time nonlinear differential equation which must be integrated by the attitude computer is given as equation 31, with its parts expanded in equations 32-34.

$$\dot{\theta}(t) = W_B(\theta(t))\omega(t) \quad (31)$$

$$\theta(t) \equiv \begin{bmatrix} \phi(t) \\ \theta(t) \\ \psi(t) \end{bmatrix} \quad (32)$$

$$\omega(t) \equiv \begin{bmatrix} \omega_x(t) \\ \omega_y(t) \\ \omega_z(t) \end{bmatrix} \quad (33)$$

$$W_B(\theta(t)) \equiv \begin{bmatrix} 1 & \sin \phi(t) \tan \theta(t) & \cos \phi(t) \tan \theta(t) \\ 0 & \cos \phi(t) & -\sin \phi(t) \\ 0 & \sin \phi(t) / \cos \theta(t) & \cos \phi(t) / \cos \theta(t) \end{bmatrix} \quad (34)$$

A discrete time attitude computation will be derived from the continuous time nonlinear differential equation. Eric Foxlin states that the evolution of $\theta(t)$ can be approximated over a short time interval using a Taylor series expansion, as shown in equation 35 [20].

$$\theta(t + \Delta t) = \theta(t) + \dot{\theta}(t)\Delta t + \ddot{\theta}(t)\frac{\Delta t^2}{2} + \dots \quad (35)$$

Deciding the number of terms to keep from the series expansion is largely dependent on the size of the length of the time step Δt . A first order integration algorithm would have an error per step that is mostly due to the third term of equation 35:

$$\omega^2 \Delta t^2 / 2 \quad (36)$$

So the error rate for a first order integration of the Taylor series expansion would be:

$$error\ rate = \frac{1}{2} \omega^2 \Delta t \quad (36)$$

This would give an error rate of 0.05 radians per second or about three degrees per second for a peak head velocity of 6 radians per second and a time step of 0.0003 seconds. This error rate is unacceptable for the purpose of this paper, due to a need for tighter tolerances. However, if the third term of the Taylor's series expansion in equation 35 is retained, then the error rate would be dominated by the fourth term:

$$\omega^3 \Delta t^3 / 6 \quad (37)$$

So then the error rate for a second order integration algorithm would be:

$$error\ rate = \frac{1}{6}\omega^3\Delta t^2 \quad (38)$$

Using the same peak head velocity and time step for the error rate of the first order integration, the new error rate would be 0.0003 radians per second or about one degree a minute. The bias drift stated by the manufacturer for the gyro package used in this research is about one degree per second [21]. For this reason, a second order integration algorithm is desired. If the continuous time nonlinear differential equation (31) is differentiated using the chain rule for partial derivatives, the result is equation 39.

$$\ddot{\theta}(t) = \frac{\partial}{\partial \theta} [W_B(\theta(t))\omega(t)]\dot{\theta}(t) + \frac{\partial}{\partial \omega} [W_B(\theta(t))\omega(t)]\dot{\omega}(t) \quad (39)$$

From equation 39, $V_B(\theta, \omega)$ is defined in equation 40.

$$V_B(\theta, \omega) \equiv \frac{\partial}{\partial \theta} [W_B(\theta(t))\omega(t)] = \begin{bmatrix} \left[\frac{\cos \phi(t) \sin \theta(t)}{\cos \theta(t)} \omega_y - \frac{\sin \phi(t) \sin \theta(t)}{\cos \theta(t)} \omega_z \right] & \left[\frac{\sin \phi(t)}{\cos^2 \theta(t)} \omega_y + \frac{\cos \phi(t)}{\cos^2 \theta(t)} \omega_z \right] & 0 \\ \left[-\sin \phi(t) \cdot \omega_y - \cos \phi(t) \cdot \omega_z \right] & 0 & 0 \\ \left[\frac{\cos \phi(t)}{\cos \theta(t)} \omega_y - \frac{\sin \phi(t)}{\cos \theta(t)} \omega_z \right] & \left[\frac{\sin \phi(t) \sin \theta(t)}{\cos^2 \theta(t)} \omega_y + \frac{\cos \phi(t) \sin \theta(t)}{\cos^2 \theta(t)} \omega_z \right] & 0 \end{bmatrix} \quad (40)$$

The derivative of $\omega(t)$ is approximated as equation 41.

$$\dot{\omega}(t) \approx \frac{\omega(t+\Delta t) - \omega(t)}{\Delta t} \quad (41)$$

Equations 40 and 41 are then substituted into equation 39 to yield the following in equation 42.

$$\ddot{\theta}(t) = V_B(\theta(t), \omega(t))W_B(\theta(t))\omega(t) + W_B(\theta(t))\frac{\omega(t+\Delta t) - \omega(t)}{\Delta t} \quad (42)$$

Then plugging equations 31 and 42 into the Taylors Series expansion of equation 35 and rearranging the terms leads to equation 43 below.

$$\theta(t + \Delta t) = \theta(t) + W_B \frac{\omega(t) + \omega(t+\Delta t)}{2} \Delta t + V_B W_B \omega(t) \frac{\Delta t^2}{2} \quad (43)$$

To clarify for equation 43, $W_B = W_B(\theta(t))$ and $V_B = V_B(\theta, \omega)$. Equation 43 is the second order discrete time integration step formula. The time step Δt is an explicit parameter in this formula, so usage in an attitude computer does not require a constant step size. There is no need for an interrupt driven program to have a constant sampling rate for data acquisition [20].

2.8. In-Field User Calibration of Inertial Measurement Units

For measurement purposes, it is necessary to validate the accuracy of an inertial measurement unit in relation to known reference information. From this relationship, calibration coefficients are determined that force the output to agree with the reference information over a range of output values. There are many ways of improving the accuracy by calibrating an inertial measurement unit, however to accurately determine all parameters provided by an inertial measurement unit, special calibration devices such as rotating “turn tables” and three-axis rotation machines are necessary.

Researchers from the Mechanical Engineering Department of the National University of Singapore determined a new method to calibrate an inertial measurement unit without any special calibration devices [22]. This new method calibrates and compensates for “non-zero biases, non-unit scale factors, axis misalignments and cross-axis sensitivities of both the tri-axial accelerometer and gyroscopic setups in a micro-electrical-mechanical systems (MEMS) based inertial measurement unit (IMU)” [22]. The Earth’s gravity is the physical reference used in this calibration method. Two properties regarding the relationship of the accelerometer package and the local gravity are the basis for calibrating the tri-axial accelerometers. The first property states that “the magnitude of the static acceleration measured must equal that of the gravity” [22]. The second property states “the gravity vector measured using a static tri-axial accelerometer must equal the gravity vector computed using the IMU orientation integration algorithm, which in turn uses the angular velocities measured using the gyroscopes” [22]. Acknowledging these properties, this new method of calibration makes possible the ability to calibrate an IMU simply by holding it in the hands and moving it for a few minutes. It is important to explicitly define all elements of the calibration function. All the following elements, matrices and mathematical relationships exist in software used in this research. In calibrating the inertial measurement unit, the following sensor errors are resolved: the non-zero bias, the non-unit scale factor, the non-orthogonal misalignment factor and the cross-axis sensitivity. The misalignment matrix is defined in equation 46. The sensitivity matrix is also defined in equation 47.

$$M = \begin{pmatrix} 1 & -\alpha_{yz} & \alpha_{zy} \\ 0 & 1 & -\alpha_{zx} \\ 0 & 0 & 1 \end{pmatrix} \quad (46)$$

$$S^* = \begin{pmatrix} S_{xx} & S_{xy} & S_{xz} \\ S_{yx} & S_{yy} & S_{yz} \\ S_{zx} & S_{zy} & S_{zz} \end{pmatrix} \quad (47)$$

For the misalignment matrix, the terms α_{ij} are for the small rotation of the sensor i th axis about the platform j th axis, in order to align with the platform i th axis. For the sensitivity matrix, the variable s_{ij} is the sensitivity of the i th axis accelerometer to the accelerations of the j th axis. The matrix S^* in equation 47 is an identity matrix. With this representation, there are no scaling errors along the axis and the sensor is not sensitive to cross-axis acceleration. Matrix M is multiplied with matrix S^* in order to combine the effects of the cross-axis sensitivity and the sensor misalignment.

$$E = \begin{pmatrix} S_{xx} - S_{yx}\alpha_{yz} + S_{zx}\alpha_{zy} & S_{xy} - S_{yy}\alpha_{yz} + S_{zy}\alpha_{zy} & S_{xz} + S_{yz}\alpha_{yz} + S_{zz}\alpha_{zy} \\ S_{yx} - S_{zx}\alpha_{zx} & S_{yy} - S_{zy}\alpha_{zx} & S_{yz} - S_{zz}\alpha_{zx} \\ S_{zx} & S_{zy} & S_{zz} \end{pmatrix} \quad (48)$$

The off-diagonal terms in matrix E , shown as equation 48, are known to have small values, and so they are ignored. The result matrix E with these terms removed is shown in equation 49.

$$E \approx \begin{pmatrix} S_{xx} & S_{xy} - S_{yy}\alpha_{yz} & S_{xz} + S_{zz}\alpha_{zy} \\ S_{yx} & S_{yy} & S_{yz} - S_{zz}\alpha_{zx} \\ S_{zx} & S_{zy} & S_{zz} \end{pmatrix} = \begin{pmatrix} e_{00} & e_{01} & e_{02} \\ e_{10} & e_{11} & e_{12} \\ e_{20} & e_{21} & e_{22} \end{pmatrix} \quad (49)$$

The matrix E is a diagonally dominant correction matrix. The proposed error model for a tri-axial accelerometer setup is shown in equation 50. The term b_a is the accelerometer bias. The term $a_{s,k}$ is the k th acceleration vector measurement in the sensor frame, or raw acceleration. The term $a_{p,k}$ is the k th accelerometer vector measured in the platform frame.

$$a_{p,k} = E(a_{s,k} - b_a) \quad (50)$$

The model parameters in matrix E and the vector b_a are collected to form θ_a .

$$\theta_a = \{e_{00}, e_{01}, e_{02}, e_{10}, e_{11}, e_{12}, e_{20}, e_{21}, e_{22}, b_x, b_y, b_z\} \quad (51)$$

Using θ_a , the function in equation 52 can be defined.

$$h(a_{s,k}, \theta_a) = E(a_{s,k} - b_a) = a_{p,k} \quad (52)$$

Now assuming that the reference of local gravity has a magnitude of unity, a cost function can be created that is based on this reference and the error model. This measures the amount of deviation from the ideal 1 G for k sets of measurements.

$$L(\theta_a) = \sum_{k=0}^{K-1} \left(1 - \|h(a_{s,k}, \theta_a)\|^2 \right)^2 \quad (53)$$

The misalignments, scale factors and cross-axis sensitivities are modeled next. Matrix M_g is the full misalignment correction matrix.

$$\omega = (\omega_x, \omega_y, \omega_z)^T \quad (55)$$

$$\omega_{p,k} = M_g S_g^* (\omega_{s,k}) \quad (56)$$

Where:

$$S_g^* = \begin{pmatrix} S_{xx} & S_{xy} & S_{xz} \\ S_{yx} & S_{yy} & S_{yz} \\ S_{zx} & S_{zy} & S_{zz} \end{pmatrix} \quad (57)$$

$$M_g = \begin{pmatrix} 1 & -\alpha_{yz} & \alpha_{zy} \\ \alpha_{xz} & 1 & -\alpha_{zx} \\ -\alpha_{xy} & \alpha_{yx} & 1 \end{pmatrix} \quad (58)$$

Just like the tri-axial accelerometer error model of equation 50, the minor misalignments and cross-axis sensitivities are not distinguishable, so equation 56 becomes the new function in equation 59. The matrices S_g^* and M_g are multiplied to form E_g .

$$\omega_{p,k} = E_g (\omega_{s,k}) \quad (59)$$

The gravity vector measured by the static tri-axial accelerometer must equal the gravity vector computed using the IMU orientation integration algorithm. This algorithm uses the angular velocities measured from the gyroscopes. With this relationship a new cost function is proposed. An operator, Ψ , converts the angular velocity $\omega_{p,k}$ from $k = 0$ to $k = n$ and the initial gravity vector u_0 to the new gravity vector u_g , which is found from the gyroscopes. This operator, Ψ , can be an orientation algorithm which determines orientation by the integration of the angular velocity. A new relationship using the operator Ψ is defined in equation 60.

$$u_g = \psi[\omega_{p,k}, u_0] \quad (60)$$

The orientation is represented with a quaternion, q in equation 61. The current quaternion q_k is related to the previous quaternion q_{k-1} for each time step k . The time interval between each time step is represented by Δt . The

matrix “I” in equation 61 is a 4x4 identity matrix. The angular velocity that is measured by the gyroscopes and represented as $\omega_{p,k}$ across the time interval Δt , is represented in equation 62 as $\delta\beta$. Matrix B is shown in equation 63.

$$q_k = \left[\cos(0.5|\delta\beta|) I + \frac{1}{|\delta\beta|} \sin(0.5|\delta\beta|) B \right] q_{k-1} \quad (61)$$

$$\delta\beta = \omega_{p,k} \Delta t = \begin{pmatrix} \delta\beta_x \\ \delta\beta_y \\ \delta\beta_z \end{pmatrix}, \quad |\delta\beta| = \sqrt{\delta\beta_x^2 + \delta\beta_y^2 + \delta\beta_z^2} \quad (62)$$

$$B = \begin{pmatrix} 0 & \delta\beta_x & \delta\beta_y & \delta\beta_z \\ -\delta\beta_x & 0 & \delta\beta_z & -\delta\beta_y \\ -\delta\beta_y & -\delta\beta_z & 0 & \delta\beta_x \\ -\delta\beta_z & \delta\beta_y & -\delta\beta_x & 0 \end{pmatrix} \quad (63)$$

The quaternion $q_n = (a, b, c, d)^T$ at the time step $k=n$ can be obtained by using the initial quaternion q_0 and the sequence of angular velocities from $k=0$ to $k=n$. The rotation matrix is obtained from q_n . The computed gravity vector u_g is obtained from the starting gravity vector u_0 multiplied by the rotation matrix.

$$R = \begin{pmatrix} a^2 + b^2 - c^2 - d^2 & 2(bc + ad) & 2(bd - ac) \\ 2(bc - ad) & a^2 - b^2 + c^2 - d^2 & 2(cd + ad) \\ 2(bd + ac) & 2(cd - ab) & a^2 - b^2 - c^2 + d^2 \end{pmatrix} \quad (64)$$

$$u_g = Ru_0 \quad (65)$$

Next, u_a is set as the gravity vector measured by the static accelerometer. The nine elements of matrix E_g from equation 59 are used to form θ_g . This is used for the definition of the cost function in equation 66.

$$L(\theta_g) = \sum_{k=0}^{K-1} \|u_a - u_g\|^2 \quad (66)$$

The difference between u_g and u_a will be equal to zero when the IMU is static, assuming there are no sensor errors or algorithmic errors in the orientation algorithm ψ . This is true for all arbitrary motions between static states. This enables the gyroscopes to be calibrated without any accurate turn rate or precise maneuvers. Since the accelerometers have already been calibrated, they provide the required gravity vector measurements. The cost functions $L(\theta_a)$ and $L(\theta_g)$ are both minimized using the Downhill Simplex method[22]. The Downhill Simplex method, or Nelder-Mead method is a common nonlinear optimization technique, and it is a well-defined numerical method for function evaluation and multidimensional [23].

A search is made across a range of quasi-static threshold values and varying lengths of time for static motion because the motion profile is uncontrolled. The residue error is used to find the least noisy set of static gravity vector measurements. The quasi-static threshold is varied from $5.8 \times 10^{-5} \text{ g}^2$ to $3.5 \times 10^{-4} \text{ g}^2$ in steps of $5.8 \times 10^{-5} \text{ g}^2$ and the static time is varied from 0.05 to 0.5 seconds in steps of 0.05 seconds. This is done in order to find the best set of data. This process simultaneously finds the accelerometer error model parameters, the quasi-static threshold and the length of the static time.

The result of the accelerometer cost function $L(\theta_a)$ gives an estimation of the residue error and the quality of the data. The next step is to separate the gyroscope data into sets using the quasi-static threshold and static time determined from the accelerometer measurements. There is an initial long pause in data collection that is used to measure the gyroscope biases. The gravity vectors are obtained by compensating the non-random errors in the accelerometer readings by using the accelerometer error model parameter values. The bias-free gyroscope measurements and gravity vectors are used to compute the parameter values of the gyroscope error model.

This new method of IMU calibration is based on the local gravity vector and mathematical reasoning. A level of confidence in this method must be exercised by comparison to true values or to another method. Unfortunately, no certified equipment for IMU calibration was available. To verify the results of the in-field user calibration of an IMU without external equipment, another multi-position method for MEMS IMUS was used. The six-position static and rate test is among the most common used calibration methods for IMUs. This method does not require any special aligned mounting, and it has been adapted to compensate for primary sensor error. As low-cost IMUs are unable to measure the rotation of the earth, such a rotation cannot be used as a reference. To compensate, a turntable was used to provide the reference rotation rate. The six-position method requires the inertial system to be mounted on a level surface with each sensor axis pointed alternatively up and down. For a triad of accelerometers, this requires a total of six positions, hence the name “six position method”.

3. Development and Construction of the PHIS

3.1. Initial Design

In 2006, Gentex and the Department of Aviation Systems at UTSI proposed the construction of “a self-contained data system [...] that can be attached to a fighter pilot helmet or other aviation head protection equipment” [13]. The data system was required to be capable of recording linear accelerations and angular rates experienced by the pilot’s head. The system would be self-contained and portable, and all its components attached to the wearer’s helmet or person. The objectives were to:

- Develop and test a self-contained portable data acquisition device for a fighter pilot helmet.
- Record motion of the wearer’s head during various flight maneuvers.
- Utilize acquired data to increase the understanding of the human head in a fighter flight environment.

The self-contained portable data acquisition device for a fighter pilot helmet was built. A low cost inertial measurement unit was mounted on the top of the helmet shell, as to minimize intrusiveness to the wearer. This was a complete three axis silicon MEMS (Micro-Electrical Mechanical System) inertial measurement module with analog outputs and an enhanced low pass filter. Attached to the inertial measurement unit was a handheld recorder, or combination Palm Pilot Tungsten T3 and Datastick data acquisition system. The inertial measurement module was connected to the handheld recorder using a wire with a breakaway quick disconnect to provide a level of safety during operation. Figure 9 provides a visual description of the initial design for the Portable Helmet Instrumentation System.

The inertial measurement unit measures seven quantities: tri-axial linear accelerations with corresponding angular rates and device temperature. The module is an O-Navi ONI-23504 motion pack, also known as a Gyrocube 3F tri-axial MEMS IMU. The unit is very compact with a volume of less than one cubic inch and it only requires 5 volts of excitation. The Gyrocube 3F is mounted to the top of the helmet shell. The helmet is a Gentex HGU-55/P advanced lightweight fighter/attack helmet, shown in Figure 10. The Gyrocube 3F was mounted so that it would not interfere with the operation of the helmet visor nor disrupt normal movement of the head and neck. A customized carbon fiber panel was mounted over the Gyrocube 3F to protect the sensor and eliminate snag hazards. This carbon

fiber panel installation is shown in Figure 10. The Gyrocube 3F, pictured in Figure 11, is connected to a Datastick DAS-1294 unit through a quick-disconnect cable.

The Datastick DAS-1294 is a mobile data acquisition module with eight single-ended analog inputs. It is attached to the helmet wearer by use of pilot kneeboard platform. To power the Gyrocube 3F and Datastick DAS-1294, an external battery (Silver Datastick Systems 6V 1800mAh NiMH manufactured by MAHA Group Corp P/N MH-DPB180M) was used. The Datastick DAS-1294 connects to the Palm Tungsten T3 handheld and uses the proprietary Datastick Connection software to view and store analog measurements. The software monitors and records measurements in real time and stores them in a Palm Operating System database on the Palm handheld. The database has a 50 MB storage limitation. Recorded data could be retrieved using the Datastick Reporting System software, which relocates measurements to a Microsoft Excel spreadsheet. The Gyrocube 3F has an output range of zero to five volts per channel, with a 2.5 volt bias for a nominal sensor reading (linear acceleration of zero G's, angular rate of zero degrees per second, and temperature of 25 degrees Celsius). This initial system, which consisted only of the GyroCube IMU and the Palm/Datastick recorder, was tested in flight. The Gyrocube was mounted to the instrument panel dust cover of the UTSI Department of Aviation Systems' Extra 300 aircraft. The aircraft was flown in several different aerobatic maneuvers. From that dataset, it was determined that the system needed to be calibrated. Other factors contributing to the obscure results would have to be investigated.

3.2. Gyrocube Calibration

Before conducting a series of tests to evaluate the performance of an accelerometer or gyroscope, preliminary tests had to be undertaken to ensure that the sensors were functioning as designed by the manufacturer. Tests included observation of the inertial measurement unit output for a short period of 10 to 20 minutes after activation to monitor the warm-up trend and to try determining the threshold acceleration and angular rate levels which produce the output signal. The method for establishing an operational PHIS IMU is as follows:

- 1) Power-On Power-Off Repeatability Test
- 2) Six Position and Rate Calibration Method
- 3) In-Field User Calibration Method

Determining the error parameters in the observed angular rate and linear acceleration is done through calibration [17]. Two forms of calibration will be applied to the PHIS MEMS IMU. The results of the two different methods will be compared for validation purposes. These two forms of calibration are the Six Position Method and the new In-Field User Calibration Method.

The traditional method of calibration is the Six-Position Test and Rate Test. The Six-Position static and Rate Tests are the most commonly used methods [17]. The Six Position Method requires the inertial system to be mounted on a level surface with each sensitive axis of every sensor pointing alternatively up and down. For a triad of orthogonal sensors, such as on the GyroCube, this results in a total of six positions. Mounting the Gyrocube on a level table with each sensitive axis pointing alternatively up and down makes it possible to determine estimates of the accelerometer biases, scale factor errors and the sensitive axis misalignments. The series of rotations are shown in Figure 12. These estimates are computed by summing and differencing various combinations of accelerometer and gyroscope measurements [24]. Bias is a displacement from a zero reading of an instrument measuring a reference condition. It is a constant error over the full range of measurements. Scale Factor is a unit-less amount of the variation in an instrument's sensitivity as ambient conditions are changed. For calibrating the gyroscopes, it is necessary to use a rate table. Since MEMS sensors are not very accurate to begin with, and the angular velocity error of a MEMS gyroscope is greater than the terrestrial velocity rotation, an artificial rotational plane must be used. It is important to note that the Department of Aviation Systems has an Aerosmith Rate Table that was last calibrated in the year 1962, which could possibly lead to significant error. The setup used to perform the Six Position Calibration Method is shown in Figure 12. To quickly perform the Six Position calibration method calculations, a Labview script was created with sections for accelerometer calibration and gyroscope calibration. Files from the Datastick software are downloaded to a personal computer (PC) and read in by the Labview script. The script computes the scale factors and biases for each axis.

The Palm Tungsten T3 is connected to the Datastick DS-1294 through the PALM special connector at the bottom of the Palm PDA. Pin number 16 of the special connector is for the charge signal, which draws a +5 volt charge supply input at about 500 to 700 mA. During the combined use of the Palm Tungsten T3 and the Datastick DS-1294, there was a power fluctuation caused by the Palm Operating System accessing Pin 16 to recharge the Palm Tungsten T3 when its internal battery went below 100% charge capacity. Since the +5 volt charge supply input is also shared by the Datastick DS-1294 and the Gyrocube 3F, each time the Palm Tungsten T3 drew power from Pin

16, it caused voltage drops on both the Datastick and Gyrocube. This resulted in extreme fluctuating measurements for accelerations and angular velocities from the Gyrocube 3F. This was solved by placing a tiny piece of duct tape over Pin 16 on the Datastick DS-1294's PALM special connector interface.

Over time the voltage output of the external nickel-metal-hydride battery decreases during use of the Datastick. This is normal for nickel-metal-hydride batteries. The discharge curve of a NiMH has a negative slope, with the voltage output level decreasing over the range of amp hours of use. Since the measurement channels are excited by the +5V power supply, measurements would decrease in amplitude over time. A +5V fixed-voltage regulator would need to be installed in-between the external NiMH battery and the Datastick DS-1294 if extended usage was planned.

The time stamps being issued by the Datastick software are of the format 'hh:mm:ss', with 'hh' representing the hour of the day, 'mm' representing the minutes of the hour and 'ss' representing the seconds of the minute. There are multiple measurements being taken per second and these measurements are given the same time stamp. A Labview script was written to differentiate between multiple data points with identical time stamps and assign new time stamps with the format hh:mm:ss.sss, with the last 'sss' representing a fractional second. The new time stamp is not representative of the true 'exact' time of the measurement, as the Datastick is incapable of recording milliseconds as part of the time stamp.

The Datastick software uses the Palm OS database structure, which has a limitation of 10,000 records for a database. The original PHIS Objectives and Requirements Document requested a 48 Hz measurement rate for the instrumentation system. At this rate, a database in the Datastick software would max out in 3.43 minutes of recording time. A more modest measurement rate of 15 Hz was used for the Six Position Calibration Method. Even at 15 Hz, a database would reach its size limit after 11.11 minutes of use.

Calibration can be challenging with a lack of certified equipment. However, the requirements of human scale inertial measurement require some form of calibration technique. In regards to low-cost MEMS IMU sensors, procuring expensive equipment that costs many times more than the cost of the actual sensors is not economical. Luckily, there is a new method of calibration that does not require any special equipment. The In-Field User Calibration Method can be used to calibrate an IMU even if it is misaligned with the local reference frame. Similar to the Six-Position method, the local gravity is used as a reference for calibration. The methodology and

mathematical background of the In-Field User Calibration Method were discussed in full in Chapter II Section 2.8. With this method, it is possible to calibrate the IMU while holding it in the hands and moving it for a few minutes.

A Labview program was created to collect calibration data and to compute the non-zero bias, and the unit-less scale factor for each of the accelerometers and gyroscopes. The front end of this Labview VI was created to interface solely with a National Instruments NI-6008 DAQ and the Gyrocube using the DAQ-MX programming interface. However, the small front end can be altered within the software to work with any six degree of freedom IMU. When the user first runs the program, the front panel for the main VI is shown, followed by the Data Collection VI. There are instructions available to the user on the Data Collection front panel. Following the methodology from Chapter II Section 2.8, the first step is to hold the IMU still for twenty seconds. This is to determine the static biases for the gyroscopes. Once the IMU has remained in a stable and static state for twenty seconds, an indicator light illuminates and the user proceeds to the next step shown on the front panel. The user must rotate the IMU in any arbitrary direction and then hold the IMU still for two seconds, repeating these two steps for thirty iterations. Each time that the user holds the IMU still for two seconds, at the end of the two second period a static state counter increments by one. The static state counter is to help the user keep count of the number of iterations that have been completed. It has been determined through experience that at least thirty movement iterations are required for a successful convergence for a solution using the Downhill Simplex Method. Once the user is satisfied with the number of static state counts and the quality of the measurements, the “Stop – Finish Recording” button is pressed. The Data Collection panel disappears and the user is returned to the main front panel. The VI will proceed automatically to determine the calibration coefficients. It will first determine the optimum threshold and static time, then build a function for the tri-axial accelerometers. This function will be minimized using the Downhill Simplex Method until the scale factors and biases for the accelerometers are determined. This VI requires MATLAB. The Downhill Simplex Method, also known as the Nelder-Mead Simplex Method, is a necessary and built-in function call from the MATLAB software. The next step for the VI is to build a function for the gyroscopes. Using the Downhill Simplex method, this gyroscope function is minimized and the gyroscope scale factors are determined. The gyroscope biases have easily be determined earlier using the twenty second static period in the measurement data. Once all calibration coefficients have been found, the VI will display both the original measurement data in white for all three accelerometers and gyroscopes in separate plots on the main front panel. Additionally, for each sensor, a red line representing the measurement data corrected using the calibration

coefficients is plotted in the same plot window. The calibration coefficients are then recorded. A sample of resultant calibration values for the In-Field User Calibration method is provided in Table 3, along with their average value and standard deviation.

Having completed the In-Field User Calibration method and the Six Position Calibration method, a comparison was made between the two. Table 4 shows a comparison of scale factors and biases from the two calibration methods. The reported scale factors and biases for the two methods are averages calculated from multiple calibration attempts for each method. Of significant note is the close similarity for resultant calibration coefficients for the gyroscopes. The calibration results for the accelerometers had varying biases, between 1.88 to 4.62 G's in variation. Scale factors for the tri-axial accelerometers determined by the Six Position Method were off from the result of the In-Field User Method by a 42% difference for the X accelerometer, a 34% difference for the Y accelerometer and a 41% difference for the Z accelerometer. Knowing that the gyroscope calibration coefficients are almost identical is significant in determining the correct calibration coefficients for the accelerometers. For the Six Position Calibration Method, the accelerometer measurements are mathematically independent from the gyroscope measurements when determining the coefficients for each. However, the calibration of the gyroscopes using the In-Field User Calibration Method is very much dependent on the previously calibrated accelerometer measurements. The cost function shown in equation 66 mathematically illustrates this relationship. If the calibration coefficients for the accelerometers determined using the In-Field Calibration Method were wrong, then the calibration coefficients for the gyroscopes determined using the In-Field Calibration Method would be wrong as well. Since two different calibration methods provide identical gyroscope calibration coefficients, it can be assumed that the gyroscope values are correct. Again, for the In-Field Calibration Method, the calibration of the gyroscopes is determined by comparing the outputs of the calibrated accelerometers and the IMU orientation integration algorithm after each rotation. It is with great confidence that all calibration coefficients determined using the In-Field User Calibration Method are correct. In regard to the accelerometer coefficients determined using the Six-Position method, a number of defects in the calibration process had to have led to the incorrect values. The level surface that the calibration was performed on was determined to be level using a bubble level, which introduced error into the system.

3.3. Finding Helmet Center and Transformation of IMU Measurements

Based on rigid body dynamics, the use of homogenous transformations will allow for the interpretation of the Gyrocube 3F measurements as linear accelerations and angular rates of and about the helmet center. The mathematical process on how to correct the translational accelerometer measurements to the body center of the helmet is provided in Chapter II Section 2.3. The Portable Helmet Instrumentation System must accommodate many different head sizes, so basing the center point as a function of mass; similar to the method of determining the center of gravity, would not be optimal. The center of the helmet was defined geometrically. Figure 13 provides a side view of the HGU-55 helmet with Gyrocube IMU installed. The helmet fairing has been temporarily removed. The helmet was considered to be a perfect sphere. The Gyrocube IMU provides all measurements corrected to the center of Gyrocube module, as shown in Figure 13. The distance from the center point of the superimposed circle to the reference center of the Gyrocube IMU was measured to be 5.5 inches. For component distances from the helmet center, the Gyrocube was determined to be 4.44 inches behind the center point along the x-axis and 3.228 inches above the center point along the z-axis. There is no displacement in the y-direction as the Gyrocube was installed on the line of symmetry of the helmet shell. The level or pitch angle of the helmet as it is worn is ambiguous from person to person, so it is very difficult to discern the appropriate Euler angle to correct for orientation of the IMU. If an imaginary dotted line is drawn through the helmet as shown in Figure 13, then the angle formed between that line and the line drawn from the IMU to the center would be 36 degrees. That means that a 54 degree pitch-down rotation for the Gyrocube is required to make it level with the helmet reference frame.

3.4. Head Tracking Relative to a Moving Platform using a Range Sensor

Depending on what information is desired, this system may not operate correctly on a moving platform such as an aircraft. The Gyrocube IMU acting alone will only provide measurements for head motion relative to the ground. It was decided that a full description of helmet motion should be made available from the measurement system. To tackle this design objective, it is necessary to track the head position and orientation relative to the vehicle platform. Due to the flexibility of the human neck, the helmet motion will be different from the aircraft. It then becomes a requirement to track the aircraft relative to the navigation frame. Both the aircraft and the navigation frame are moving relative to the inertial reference frame. Measuring aircraft motion can be done similarly to the way the helmet motion is measured. Eric Foxlin suggests in his paper “Head-tracking relative to a moving vehicle or

simulator platform using differential sensors” to compute the tracking IMU mounted on the helmet relative to a reference IMU rigidly attached to the vehicle. Fortunately, an Attitude and Heading Reference System (AHRS) exists already on the test aircraft, the Extra 300. An AHRS is a tri-axial sensor that provides heading, attitude and yaw information for the aircraft. By collecting data from the AHRS and relating it to the measurements from the helmet, the motion of the helmet internal to the aircraft cockpit can be determined. However, knowing the exact relationship for relating orientation and accelerations between the aircraft and helmet requires knowledge of the head position and orientation relative to the vehicle. A range sensor could measure this. There are many forms for range sensors, including acoustic ranging, magnetic sensors, and optical sensors. One method of optical range tracking is the use of a camera monitoring a collection of points and determining pose from their coordinates.

To cheaply accomplish a form of point tracking, a common web camera could be used to track infrared points of light attached to the body in motion. There are several forms of software available in the open market that uses this kind of setup. However, all forms that utilize traditional cameras burden the computer with additional tasks such as blob tracking and coordinate determination. A camera with an on-board processor to accomplish these additional tasks would be desired. Very few cameras that include on-board processing exist on the open market. The two most prominent are NaturalPoint’s TrackIR and Nintendo’s Wii-Remote. The Nintendo Wii Remote is shown in Figure 15. Of the two, the Nintendo Wii-Remote was the most cost-effective solution. Both operate with 100 Hz image sampling. However, the Wii-Remote has a sub-pixel reporting resolution of 1024 by 768 pixels while the NaturalPoint TrackIR only operates with 355 by 288 pixels [25]. When deciding on hardware, the ability to integrate with Labview also became a necessity. There were no known ways of recording data from a NaturalPoint device into Labview. It was by luck however that a Labview programmer Sam Shearman from National Instruments had created a Labview interface to a Wii Remote [26]. This enabled collection of the raw coordinates of the image points for use with other Labview VI’s. Due to the better performance characteristics and easier form of integration, the Nintendo Wii-Remote was chosen as the range sensor for measuring head pose relative to the aircraft frame. Figure 15 shows an exterior view of the Wii-Remote and its IR-sensing camera. The architecture of the Wii-Remote does not allow for the original image from the camera to be accessed by an outside party.

Point tracking allows for three-dimensional pose determination. By monitoring infrared emitting lights on an object and knowing their relative distance from one another, it is possible to compute the position and orientation using a method of perspective projection. A model of weak-perspective projection was used to determine

geometrically the pose of the helmet within the cockpit of the aircraft. Further details on calculating the weak-perspective projection can be found in Chapter 2 Section 2.5. For the three points, infrared LEDs powered by two AA batteries were installed on the front of the helmet as shown in Figure 17. Infrared LEDs are used because they can be better isolated from visible light using filters. As a human factors issue, IR-LEDs are also discrete as they emit no visible light that could be distracting to a pilot or aircrew. The 5mm infrared LEDs were used because they are larger than the 3mm variety and so provide better sub-pixel tracking accuracy for the camera. Since the viewing angle of the IR-LEDs is too narrow for point-tracking, the epoxy section of the LEDs was manually filed down. This gives the IR-LEDs a wider emission angle, but also reduces brightness. The dimensions of the LED triad are described by three measurements. The center point, denoted by the letter “R” in Figure 17 is centered laterally by 4 inches in the Y direction, and sits 1 inch forward in the longitudinal direction of the two side points. The center point also is 3.9 inches in the Z direction above the two side points.

A problem with the IR-LED configuration shown in Figure 17 was discovered where the plane defined by the 3 points would get close to being parallel with the camera plane when pitching upward. The tracking solution would become extremely unstable. Tracking would also become unreliable when a forward point moved behind the middle point, particularly when the helmet was yawing. These conditions could be avoided by adjusting the LED point model such that the middle point would be set deeper than its height and width from the outer points.

3.5. Adjustment to Range Sensor for Sunlight

While testing the IR-tracking capabilities of the camera, it became apparent that refracted and direct sunlight created false tracking-points that were detected by the software. Although the Wii-Remote has its own visible light filter over the IR-sensor, additional filtration was needed. A readily available resource was ordinary 35mm photographic color film negatives. Using exposed color film; the film negative material can block out most visible light but allow 880 nm or higher infrared light to pass through, as shown in Figure 16. The IR-LEDs installed on the helmet emit 940 nm infrared light. Through trial and error, a total of 32 layers of exposed color film negative were applied over the IR-sensing camera. Any further layering resulted in degradation of the point-tracking capability of the camera. The inexpensive fix effectively removed the false tracking points when operating the helmet and camera in sunlit conditions.

3.6. Determining Mounting Location of Range Sensor in Aircraft

The camera must be mounted somewhere in the cockpit of the test aircraft, such that the helmet LEDs are within the camera field of view. Consequently, the method of installation of the camera in the aircraft will determine how the LED Point-Model will be attached to the helmet and adjusted from its configuration shown in Figure 17. A large flat surface is available on the pilot's instrument panel dust cover in the Extra 300 aircraft. Though no mounting hardware exists, the large flat surface provided a simple mounting solution that allows for either a forward facing or rear facing camera, as shown in Figure 18. The camera would be attached to the dust cover with an adhesive layer of industrial grade Velcro. The camera has a vertical 23 degree field of view, which can present problems viewing the point-model in very short range applications. A field of view cone of 23 degrees has been superimposed over the images shown in Figure 18. The mounting option illustrated in the right image of Figure 18 shows the possibility of mounting the camera in a rear-facing position, should the pilot wear the helmet. It can be seen in the left image of Figure 18 how there is an increased possibility of the point-model leaving the field of view of the camera. This might be partially averted by increasing the mounting height of the camera along the z-axis. This option was avoided because of structural concerns. Despite the limited effective field of view for a forward facing camera configuration, it was decided that a test subject sitting in the passenger seat would wear the helmet for flight testing. This ensures a level of safety as the PHIS system will not interfere with the pilot and aircraft operations. Measuring the distance of the mounted camera from the aircraft datum, the displacement in the longitudinal direction was 5.25 feet behind the datum, 0.42 feet to the left of the aircraft centerline in the lateral direction, and 0.83 feet above the waterline.

Since a configuration change was made for how the camera was positioned in relation to the LED point model installed on the helmet, a change had to be made to the LED point model itself. The location of the three LEDs shown in Figure 17 could no longer be seen by the IR-sensing camera, which was now located behind the helmet. A new point model had to be built that would be visible to the camera. To best avoid a situation where the plane defined by the three LED points would become parallel with the camera plane, the point model plane needed to be in a near perpendicular orientation in relation to the camera plane when the helmet is sitting "straight and level". The best option for this was a clip-configuration with the LEDs installed in a small triad on either the left or right side of the helmet. Since hardware already existed on the left side of the helmet to accommodate the installation of a microphone, it was decided to manufacture a LED mount that would attach here. Figure 19 shows the final

placement of the LED triad on the HGU-55 helmet. Furthermore, the dimensions for this new triad are shown in the same figure. The final dimensions of the LED triad in inches are given relative the center LED “R” and using the helmet reference frame. The top-most LED is 0.6 inches above the center LED and 0.5 inches behind in the longitudinal direction. The bottom-most LED is 1 inch below the center LED and 1.1 inches behind it in the longitudinal direction. A custom-made mount was produced to accommodate the new LED triad, as shown in Figure 20. This mount was made using an aluminum plate and a Styrofoam core to hold the LEDs and their resistors. A layer of epoxy was used to seal the Styrofoam and to adhere the LEDs in place.

3.7. Consolidating the Data Acquisition System

Maintaining a common timing source was important for data collection. It was found to be very difficult to correlate data from the Palm Pilot Tungsten T3 and Datastick in combination with the Samsung Q1 UMPC. The Palm Pilot and Datastick were being used to record data solely from the Gyrocube IMU mounted on the helmet. The Samsung Q1 UMPC was being used to record aircraft data from the AHRS using Christopher Ludwig’s GRT Realtime Data Acquisition Software [27]. AHRS data was accessed from a pre-installed optically-isolated Serial-to-USB hub located in the aircraft, as shown in Figure 21. In addition, the Samsung Q1 was the only machine available that could connect using Bluetooth wireless with the Nintendo Wii-Remote. To simplify the system setup, all measurement responsibilities were transferred to the Samsung Q1 UMPC. The Palm Pilot and Datastick hardware were abandoned.

The entire system is self-powered. The Samsung Q1 UMPC is powered by its own internal lithium-ion battery. The National Instruments NI-6008 DAQ energizes the Gyrocube IMU with a +5 Volt analog supply, which is provided by a USB port on the Samsung Q1. The Gyrocube to NI-6008 to UMPC connection is illustrated in Figure 22. Likewise, the Serial-to-USB hub is powered by a USB port on the Samsung Q1.

The UMPC alone does not have a strap or mounting bracket. To rectify this, a leg-mount system from RAM Mounting Systems Incorporated specifically made to fit the Samsung Q1 UMPC model was used. This leg-mount system featured two adjustable straps for securing the UMPC to the leg during flight. In addition a sealable enclosure, also from RAM Mounting Systems, was converted to serve as housing for the NI-6008 DAQ and the battery pack for the IR-LED Point Model. This setup can be seen in the lower right-hand corner of Figure 22. A demonstration on how the PHIS would be worn using the mounting system is shown in Figure 23.

Upon design finalization, the PHIS was considered a self-contained data system that can be attached to a fighter pilot helmet. The preliminary design was to measure linear accelerations and angular rates of the helmet, displaying this data in real-time. The objectives were to develop and flight-test this system. The design was amended to provide pose and motion data of the helmet in the aircraft frame and inertial frame. To do this, data regarding helmet orientation and position relative to the aircraft and data regarding aircraft orientation relative to the ground had to be made available. The helmet orientation and position are related to the aircraft by use of an optical-based point tracking system supplemented by inertial navigation equations to compensate for occlusions. The aircraft is related to the earth or inertial frame by accessing the aircraft's Attitude and Heading Reference System (AHRS). The PHIS data system is shown in its completed form in Figure 22, and the system interfaces are illustrated in Figure 24. The Data Acquisition software was programmed in Labview. This software combines point-tracking, using three dimensional pose determination algorithms, with an underdeveloped form of inertial navigation. The DAS has a bench test mode and an aircraft mode. The bench test mode must be selected when operating in a stationary environment. Aircraft mode must be selected when operating in a moving platform or vehicle. Data regarding platform or vehicle motion, such as output from an AHRS, must be available for the DAS to operate in aircraft mode. The software structure is outlined in Figure 25. The software can capture the following data:

- Helmet angular rates (Provided in: Raw Format, Helmet Frame, Aircraft Frame, Inertial Frame)
- Helmet linear accelerations (Provided in: Raw Format, Helmet Frame, Aircraft Frame, Inertial Frame)
- Helmet Attitude: Roll Angle, Pitch Angle, and Yaw Angle (Provided in: Aircraft Frame, Inertial Frame)
- Helmet Position: X position, Y position, Z position (Provided in: Aircraft Frame, relative to camera)
- Inertial tracking mode and Point-tracking mode flags
- Aircraft angular rates (Provided in: Aircraft Frame)
- Aircraft Linear Accelerations (Provided in: Aircraft axis)
- Aircraft attitude: Roll angle, Pitch angle, and Yaw angle (Provided in: Inertial axis)
- Additional aircraft parameters available from the EFIS Third Party Interface

Stand-alone calibration software for a 6DOF IMU that provides full calibration of the accelerometers and gyroscopes without need of a physical "truth source" or additional hardware was developed for use with the PHIS.

Though programmed to calibrate the Gyrocube, this software can also calibrate any 6DOF Inertial Measurement Unit (IMU), provided the end user makes the appropriate alterations to allow the IMU to interface with Labview.

4. PHIS Testing

4.1. Bench Testing

After a complete prototype was built, it was necessary to validate the hardware and software functionality. The prototype configuration, shown in Figure 26, was used for the bench testing. The PHIS was tested using the data acquisition software and calibration software, installed on a Samsung Q1 UMPC. A number of tests had to be conducted to validate the final IR-LED point-model, check the Gyrocube output, determine the battery life, and monitor software operating speed and stability.

It was important to determine if any bias or scaling was required to produce accurate distance measurements from the helmet position in relation to the lens of the IR-camera. The maximum operating range within the camera's field of view would also be determined. A flat surface was covered with a tape grid with increments measured out to the nearest inch. The IR-camera was placed on the center of the edge of the grid. With the IR-LEDs activated, distance measurements for the X and Y position of the LED triad were recorded from the table and from the UMPC. This was done at several different locations over the tape grid. The resultant calibration coefficients for distances are available in Table 5.

The biases and scale factors were also determined for attitude measurements. A routine for calibration was performed for determining helmet attitude in relation to the IR-camera. A rotating platform allowed for attitude calibration. This platform, shown in Figure 26, was machined to allow for mounting of a measurement system and it is engraved with angle measurements from negative sixty to positive sixty degrees. The helmet was mounted to the platform. To measure roll angle, the helmet was rotated incrementally through several angles listed on the rotating platform. The UMPC roll measurements were recorded as well and compared. Roll increments were ten degrees apart for measurements from zero roll to +/- 60 degree roll. Similarly, the helmet was repositioned on the rotating platform and the same method of measurement was applied for pitch and yaw. The resultant attitude calibration coefficients are listed in Table 5.

Similar to the IR-LED to camera distance test, a flat surface was covered with a tape grid with increments measured out to the nearest inch. The IR-camera was placed in the center at one edge of the grid. The helmet with

attached IR-LEDs was allowed to rest on the table at an arbitrary location on the grid. With the LEDs activated, distance measurements for the X and Y location relative to the camera lens were recorded from the table and from the UMPC. Then, obscuring the LEDs, the helmet was moved to a new location on the grid. Distance measurements were again taken from the table and the UMPC. This process was repeated for several times at various locations.

The helmet was mounted on the rotating platform once again to determine if errors were present in attitude measurements using the inertial sensing Gyrocube. To measure the roll angle, the helmet and platform were rotated incrementally through several angles while the IR-LEDs were obscured. The UMPC roll measurements were recorded and compared to the roll measurements from the rotating platform. This was done in increments of ten degrees from zero to +/- 60 degrees of roll. The helmet was then resealed and the same process was repeated for measuring pitch and yaw.

For both tests that observed Gyrocube inertial measurements, there was an expected exponential increase of error over time. The error accumulation over several time steps was observed. These measurements were post-processed and a conclusion was made on the maximum acceptable operating time for the helmet instrumentation system when in inertial navigation mode.

The maximum possible operating time for the PHIS had to be determined. Since actual flight times would vary based upon each mission and ground operations must be considered, it is important to know the limitations of the on-board batteries. The method for determining battery life was to charge the UMPC to full battery capacity. Fully charged batteries were placed into the IR-LED battery pack and the IR-camera. A timer was started, and the PHIS system was turned on. Next, the IR-camera was connected via Bluetooth wireless to the UMPC and the the IR-LEDs were turned on. The Data Acquisition Software was started on the UMPC. The system was left to run until there was an uncommanded shutdown due to loss of power of the UMPC or the camera or the LEDs.

The functionality barring any uncommanded shutdown or lockup in software of the PHIS was verified. The reliability of the software had to be proven prior to flight testing. While performing bench testing, any lockups or uncommanded shutdowns would have been recorded, including all error messages and observable effects. The operating speed was also monitored during bench testing. Any speed variation in data collection would have been noted. However all equations use Δt , or time for each computer cycle, which makes a uniform time period for all

computational cycles unnecessary. For every ten minute mark of use, any apparent and progressively degrading slow down of the software would have been recorded.

4.2. Aircraft Ground Test

After completion of the PHIS testing in the lab environment, the system was tested in the aircraft to verify the same functionality, including the connection of the aircraft AHRS during PHIS operation. A complete setup of the PHIS hardware being worn by a flight test engineer in the test aircraft is shown in Figure 27. This setup represents the system at the beginning of aircraft ground testing.

The first of several ground tests was the determination of field of view limits for the IR-camera. Too small of a field of view would result in higher reliance on inertial navigation, likely to result in more error accumulation over the length of time of the flight. The reliability of the inertial navigation functionality had already shown poor results in the lab environment. An adjustment to camera position and orientation may offer some relief for field of view limitations. To conduct this test, while sitting in the passenger seat of the aircraft, the flight test engineer positioned his head while wearing the PHIS in various locations around the passenger section of the cockpit. This was done without being physically strapped into the aircraft with seatbelts. During flight the test engineer would be strapped down, so the results of this test would reveal limits that would not be reached during flight.

Interference due to visible light or infrared light from the sun or other external light sources was investigated for the IR-camera's functionality in the aircraft cockpit. Interference from sunlight or even the EFIS display screen situated in the passenger's instrument panel could result in bad position and attitude readings from the IR-camera. While the test engineer sat in the passenger seat with the PHIS activated and the hanger door open, the IR-LED display in the Data Acquisition software front panel would be monitored for sporadic jumps in LED position as had been noticed prior to installation of the visible light filter. The aircraft canopy was lowered and locked in order to mimic in-flight conditions. The EFIS was turned on so that the LCD screen was energized. The camera was pointed in the direction of the sun and the DAS was monitored to see if the camera picked up the additional light source. No random or sporadic changes in LED position were noted, confirming the installation of the visible light filter was successful.

With the AHRS plugged into the PHIS through the serial-port-to-USB hub, a test was conducted to see if accurate attitude angles were available during operation. This test was simply verifying the orientation of the helmet

with respect to the inertial frame. Since the aircraft is a tail dragger, the incline of the aircraft center line should have been evident when the pilot was looking straight forward.

To ensure safety and to investigate human factor issues, a run-through of wire and cable locations was performed. It was investigated whether any equipment or wiring would cause snags or prevent a successful egress in the event of an emergency. While sitting in the passenger seat with the PHIS fully installed, the test engineer evaluated potential snag locations and practiced an egress. Inference of the PHIS with control stick movement was also investigated. Efforts making sure the test engineer could operate the EFIS and aircraft controls were made. Simply securing the wiring for the PHIS underneath the existing seatbelts ensured flight safety.

Prior to the first flight test, a series of radio frequency tests were performed. The aircraft was rolled out of the hanger. The PHIS was turned on and all elements were activated, including connection of the Bluetooth camera, connection to the Gyrocube, and connection to the aircraft AHRS. A normal startup of the aircraft was performed following a slightly altered checklist. Interference with radio, aircraft systems and power were checked. Packet loss was also checked for the AHRS and EFIS system. No issues were reported. Since it would be the first time that the PHIS recorded data while the aircraft engine was operating, vibrational effects were investigated. Possible effects could have been unacceptable camera-shake and vibration noise in the Gyrocube measurements. While the aircraft engine was running, PHIS data was recorded and any variation that may have resulted from vibrations was later checked for.

4.3. Aircraft Flight Test

The overall objective was to perform a limited evaluation of the PHIS applied in the aviation environment. A total of 2 sorties were flown for about 30 minutes each for a total of 1.1 hours of flight time on June 30th, 2009 and July 14th, 2009. The PHIS was flight tested on the UTSI Department of Aviation Systems' aerobatic aircraft, the Extra 300. The PHIS was worn by a flight test engineer during the flight test. The completed tasks, or aerial maneuvers, for each of these tests are summarized in Table 7.

The PHIS evaluation was accomplished by exposing the system to various types of accelerations and angular rates. Roll, pitch and yaw were recorded by the PHIS as well as linear accelerations during each maneuver conducted in Table 7. The aircraft's AHRS was used as a truth source for evaluation. The flight test engineer attempted to keep a forward-facing, eyes-level head position during each maneuver. This was to keep the IMU

readings in the helmet measurement frame in alignment with the aircraft AHRS. Minor deviations between the helmet IMU and AHRS measurements were to be expected since the human neck is flexible. The following literature will explain in greater detail how each maneuver listed in Table 7 was flown along with the expected excitations along the aircraft frame and their respective measured quantities.

Level flight was used as the baseline maneuver for evaluation of the PHIS. The aircraft was considered to be in level flight when longitudinal and lateral accelerations were zero, the vertical acceleration was a positive one G acceleration, and all rotational rates were zero. While in level flight, the flight test engineer kept the helmet in a straight-and-level position in line with the aircraft center line. Ideally, acceleration and rotational measurements for the helmet during level flight would be very similar to the measurements recorded in a straight-and-level sitting position on the ground.

A G-warmup maneuver involving steep turns was flown to build G-tolerance for the aircrew, but it also served as a wonderful opportunity to validate the PHIS system. A steep turn was performed with a bank angle of sixty degrees or more and a pull on the control stick that increased the pitch angle of the aircraft. The acceleration was towards the center of the turn. A positive acceleration along the vertical axis of the aircraft and helmet was expected to be measured.

The barrel roll is a combination of a loop and a roll. For a barrel roll maneuver, one loop is completed while the aircraft also completes one full roll. The flight path during a barrel roll has a shape of a horizontal corkscrew. During a barrel roll, the pilot always experiences positive G's along the vertical axis of the aircraft.

The next maneuver was a knife-edge, flown with both left and right bank. The aircraft is rolled 90 degrees, with the wingtips perpendicular to the ground. While rolling into this position, opposite rudder was applied to keep the nose high and the plane flying level. More rudder throw may have been required. The elevator is used to keep the aircraft flying straight. The purpose of the knife-edge maneuver was to excite lateral acceleration for the helmet and aircraft measurement systems.

The aircraft was then rolled to inverted flight. The aircraft was flown inverted to allow the measurement of a negative one G acceleration along the vertical axis of the helmet and aircraft. In addition to simple inverted flight, the aircraft performed a bridge maneuver. The pilot pitched up the aircraft from level flight. He then proceeded to make a 180 degree roll. The aircraft was now in a 45 degree inverted up-line. The stick was pulled to bring the aircraft's longitudinal axis level with the ground. The aircraft was flown in inverted flight for a period of time

before the stick was pulled to bring the aircraft on a 45 degree inverted down-line. The aircraft was rolled wings-level and the stick pulled back to resume straight-and-level flight.

Two flat turns were flown. A flat turn is a turn without any banking. This involves yawing the aircraft around using the rudder. The aircraft is prevented from any bank by using aileron inputs. The purpose of the flat turn was to create an isolated yawing motion.

A slow roll was also performed. The slow roll is flown on a straight line. The roll rate has to be constant and the longitudinal axis of the aircraft has to remain straight. This requires constantly changing the rudder and elevator control inputs throughout the roll. The slow roll was flown to generate a rolling motion along the longitudinal axis that could be measured. In addition, a hesitation 4-point roll was flown. The hesitation 4-point roll is similar to the slow roll, but there are four separate stops. Each stop is ninety degrees apart from the previous stop. The fourth stop should place the aircraft back at a straight-and-level position. This was another opportunity to measure rolling motion and also to pick out variations in accelerations for both the lateral and vertical axes.

The square loop is a maneuver similar to a traditional loop. This maneuver was flown to generate a pitching motion and to vary accelerations for the longitudinal and vertical axes. The aircraft is pulled into a vertical line. There is another pull into an inverted horizontal line. Next there is a pull into a vertical down-line. The two vertical lines and the horizontal line on top have to be the same length. The exit line at the bottom of the loop has to be at least as long as the other three sides. The quarter loops that connect the four sides have to be of the same radius at each corner.

The last aerobatic maneuver performed during the flight was the hammerhead. The hammerhead is started with a quarter loop to an up-line vertical. As the airspeed decreases, rudder is applied and the aircraft is rotated about its yaw axis. The nose will fall through the horizon and point toward the ground. The aircraft then travels along a vertical down line before a quarter loop to level flight. The cartwheel portion of the hammerhead is performed with full rudder and opposite aileron. The purpose of the hammerhead was to measure an acceleration change along the longitudinal axis as the aircraft traveled along the vertical up and down line. Yaw measurements were also to be recorded during the yawing portion of the maneuver.

A major question that required investigation was if there would be successful functionality of the PHIS in conjunction with the aircraft AHRS and data collection using Christopher Ludwig's GRT Data Acquisition Logger. Ground testing had proven all elements of the PHIS, including the Gyrocube, IR range sensor, and UMPC software

package, were functional. However, no testing had been done to verify that data could be collected from the aircraft AHRS in conjunction with data collection operations from other PHIS instruments. During the first flight, “Not-a-number” values (NaNs) were reported by the handheld display for all AHRS-dependant variables. These values included all pose, linear accelerations and rotational rates for the helmet reported in the inertial frame. Alterations had to be made to the code to accommodate several indexing errors when recording data from Christopher Ludwig’s GRT Logger VI. A short ground test was run to verify proper use of data from the GRT Logger VI.

The LED IR range sensor did not operate as planned either. Only 10,609 out of 28,645 data points or 37% of all data collected was in LED mode. This required a configuration change of the LED triad location. These displacements are necessary to account for the small field of view of the range sensor. The FOV of the range sensor was 23 degrees vertical and 31 degrees horizontal. The probability of the LED triad traveling outside the camera field of view was very high, so a physical correction was necessary. An extension bracket was added to lower the LED triad into the field of view of the range finder in hopes of increasing its effectiveness. Another reason for the extension bracket was to fix the LED model into a locked position. Previously, the LED model would be bumped into, thus causing a displacement of the LED model and generation of incorrect data regarding helmet pose and position. The final offset of the LEDs from the helmet center in centimeters were 7.214 in the x-direction, 16.916 in the y-direction and -15.612 in the z direction.

The maneuvers and flight plan for the second flight were identical to the previous test flight. The second flight served to validate the software fix in-flight and to verify that all real-time calculations were successfully performed in flight. Sadly, despite efforts to adjust the LED point model location using the extension bracket, there were still issues keeping the LED model within the limited field of view of the range sensor. The total number of data points collected during the second flight in LED mode was 10,288 points out of 29,548 points total, or 35% of the total data.

5. Experimental Results

5.1. Level Flight

To produce a baseline for the flight data, the aircraft was flown at level flight for about one minute. A small section of that data, consisting of 20 seconds, was plotted. The first plot for level flight data is shown in Figure 39. This is a plot for the longitudinal accelerations versus time. Before proceeding with the analysis of this plot, it is

important to explain how all plots are set up for results of both test flights. In every plot, the blue circles represent the helmet IMU data represented in the helmet frame. The red triangles represent the aircraft AHRS data with respect to the aircraft frame. The green crosses represent helmet IMU data, but transformed to the aircraft reference frame. Ideally, if there is deviation between the helmet IMU data in the helmet reference frame and the aircraft AHRS data, the transformed helmet IMU data should line up exactly with the aircraft AHRS data. If all elements of the PHIS are correctly functioning and the flight test engineer is able to maintain a head position in line relative to the aircraft reference frame, then all three time histories should line up. Looking at Figure 39, the longitudinal accelerations versus time for level flight, the ideal acceleration measurement would be zero G's. This is because the aircraft is experiencing little to no acceleration in the longitudinal direction. The AHRS data reflects this desired state for straight and non-accelerating flight. In this time history, the helmet IMU data in the helmet reference frame is off by 0.6 G's. This measurement fluctuates down to 0.3 G's, but returns to the average value of 0.6 G's. This could very well be attributed to an offset of the helmet IMU and reference frame from that of the aircraft's. A slightly positive longitudinal acceleration suggests the flight test engineer was looking slightly upward. The acceleration is a component of the earth's gravity. This offset and prediction could be further verified if the vertical acceleration measured by the helmet IMU was analyzed. In fact, in Figure 43, the helmet IMU vertical acceleration measurement is reported at 0.8 G's for its average value, as opposed to the expected value of 1 G. This further suggests either the flight test engineer's head was looking upwards, or the helmet itself was being worn slightly upwards during data capture for level flight data. The great news is that the helmet IMU longitudinal acceleration data was successfully transformed to the aircraft reference frame. Looking at Figure 39, it is apparent that the helmet IMU data in the aircraft frame overlays the aircraft AHRS data. Deviation between the helmet IMU data, represented as green crosses, and AHRS data, represented as red triangles, is less than 0.2 G's. The successful alignment of this data is due largely to the operation of the range sensor. The dark green squares along the bottom and top edges of the plot correspond to the LED operation mode of the PHIS. When the LED Mode has a value of one, it means the LED point model is within the field of view of the range sensor and the PHIS is able to track helmet pose and position. When the LED mode flag has a value of zero, it means that the LED point model is out of range and the PHIS system cannot determine helmet pose and position using the range sensor. However, a simplistic form of an inertial navigation system algorithm is used to determine helmet pose and position using dead-reckoning. The accuracy of the INS algorithms will become apparent during further discussion of flight test data. In

Figure 39, for the longitudinal acceleration, the PHIS operated in LED mode for 62% of the twenty seconds of plotted data. Inoperability of the LED mode had no visible effect on IMU to AHRS comparison. No drift of the transformed helmet IMU data was evident. Further details on range sensor functionality can be found for both the first and second test flights in Table 8 and Table 9, respectively.

Figure 40 shows roll rate for level flight over the same time period as Figure 39. The aircraft AHRS reported zero roll. Likewise, both helmet IMU readings in the helmet and aircraft reference frames were zero, but minor fluctuations varying from positive 100 to negative 100 degrees per second of roll were evident. This may be due to minor head moments. Lateral accelerations in Figure 41 also proved a successful lineup for helmet IMU and aircraft AHRS data. No lateral accelerations should have been present during straight-and-level flight. Assuming the flight test engineer kept his head level, the baseline data presented here is consistent with expected lateral motion for straight-and-level flight. Figure 42 shows pitch rate versus time for level flight. Again, following the trend set by previous plots for level flight data, all helmet data and aircraft AHRS data overlay very well. At the ten second mark, a short two second pitch up and pitch down motion can be seen. This may be due to a small bump or other unknown physical phenomenon resulting in a minor deflection of the helmet.

The vertical accelerations observed by both the helmet and AHRS are shown in Figure 43. For straight-and-level flight, the vertical acceleration experienced by the aircraft should be one positive G. This is shown in the level flight data results. However, the vertical acceleration reported for the helmet IMU in the helmet reference frame is only an average of 0.8 G's. As discussed earlier, this may be due to a slight pitching-up of the flight test engineer's head or simply the way the helmet was worn. The PHIS software transforms the vertical acceleration experienced by the helmet and almost always overlays the helmet acceleration in the aircraft reference frame to the AHRS data. There is a small deviation between the two of 0.1 G's.

Lastly, yaw rate data during level flight was compared between the helmet and aircraft. Without any yawing motion, all data should be zero. The AHRS yaw rate shows this, as well as the helmet IMU data. There are four fluctuations of about +/- 150 degrees of yaw occurring over the twenty seconds of data. Three of these occur in LED mode, suggesting that they are very real phenomenon and are likely slight movements of the helmet due to the flight test engineer moving his neck.

The resultant level flight data from flight two tells a slightly different story than from comparable data from flight one. In Figure 45, the longitudinal acceleration of the aircraft was kept at a constant 0.2 G's. This is different

from flight one, where the reference acceleration from the AHRS was less than 0.1 G's. The time period of level flight data for flight two is longer at one hundred seconds; versus the twenty seconds plotted for flight one. Head orientation relative to the aircraft, as shown in Figure 45, seemed satisfactory due to their overlaying. The helmet acceleration in the helmet reference frame was within 0.1 G's of difference to the aircraft acceleration in the aircraft frame. When the range sensor provided helmet orientation in reference to the aircraft frame, the helmet acceleration measurements in the aircraft frame were reported at an average of 0.6 G's. The range sensor successfully functioned from the thirty second mark to about the eighty second mark. The completed 0.2 G increase for longitudinal helmet acceleration from helmet frame to aircraft frame has no explanation.

Figure 46 shows the roll rate during level flight for flight two. The gyroscopic measurements for both the helmet IMU and the aircraft AHRS appear consistent. The lateral accelerations recorded during this time, however, do not maintain consistency, as shown in Figure 47. Helmet accelerations in the helmet frame match up pretty well with the reference AHRS accelerations. Although when transferring helmet lateral accelerations to the reference frame, measurements are driven upward by an average of 0.3 G's. No explanation is available. The first thirty seconds of lateral acceleration in the aircraft frame can be dismissed because it was computed inertially and not with the range sensor. It drifts from 0.2 G's to 1.0 G's in fifteen seconds. When the range sensor is available at the thirty second mark, the helmet lateral acceleration measurement returns immediately to 0.2 G's. The pitch rates recorded for the aircraft and helmet are shown in Figure 48. No noticeable major deviations are present in this plot. Small pitch motions of the helmet, measuring no more than fifty degrees per second and lasting about a second in length are in all likelihood caused by the flight test engineer moving his head slightly.

The vertical accelerations for the helmet in the helmet frame and the AHRS in the aircraft frame line up perfectly, as seen in Figure 49. These two measurements fluctuate around 1.0 positive G's, an expected value for vertical acceleration in straight-and-level flight. Strangely, just as described in longitudinal and lateral accelerations for the level flight baseline maneuver, the helmet accelerations that have been transformed by the computer to the aircraft frame show a consistent and noticeable displacement. The vertical acceleration for helmet in the aircraft reference frame has an average value of 0.7 G's, about 0.3 G's less than the other two measurements. No explanation is available, but it is believed the offsets of tri-axial accelerations for the helmet in the aircraft reference frame are correlated. Figure 50 shows the yaw rates for both the helmet and the aircraft. Helmet yaw rates are reported consistently at zero degrees per second, with minor deviations likely from head motion induced by the

flight test engineer. The aircraft is reported to be yawing at a constant ten degrees per second to the left, over a period of one hundred seconds. This is not consistent with expected values for straight-and-level flight. This constant yaw displacement was not observed in the first test flight's yaw data.

5.2. G-warmup Maneuver

Following level flight, the test pilot performed a G-warmup maneuver, also known as a steep turn. Two noticeable effects occurred. There was an increase in positive G-forces in the vertical and also the longitudinal axis of the aircraft while in the turn. When the range sensor was producing good data, the helmet acceleration in the aircraft frame followed along the same consistent overlay of the other accelerations. Figure 51 shows the longitudinal accelerations for two steep turns. This is produced from data from flight one. Significant drift was shown when the range sensor was not available for periods of five seconds or more. The range sensor operated for 41% of the length of time of the G-warmup maneuvers. The first steep turn was flown at a shallower bank angle than the final steep turn, resulting in a lower induced G-force. The first steep turn begins at the ten second mark and exits at the twenty second mark. The maximum induced vertical acceleration was 2.5 G's. The second steep turn, with its larger maximum vertical acceleration of 3 G's, begins at the twenty-five second mark and exits at the forty-five second mark. For both steep turns, the reported longitudinal acceleration from the AHRS exhibited a more linear increase during the turn. The longitudinal acceleration of the helmet, both in regards to the helmet and aircraft reference frames, shows a larger maximum G-force during the second turn at 0.7 G's, approximately 0.3 G's greater than the baseline AHRS. The second test flight also featured a number of steep turns for G-warmup. It is important to note that the range sensor did not work during the maneuvers. Again, helmet acceleration in the aircraft frame drifts significantly when the range sensor was not available. The transformed helmet vertical acceleration overlaid the original helmet vertical acceleration during this brief five second period. The results from the second flight's G-warmup maneuvers gave similar results to those from the first flight. Data for the G-warmup maneuvers from the second flight can be found in Appendix B.

5.3. Barrel Roll Maneuver

Following a series of G-warmup steep turns, a barrel roll was performed. The barrel roll from flight one is represented in Figure 52, Figure 53, and Figure 54. The longitudinal accelerations are shown in Figure 52. The range sensor did not work at all during the performance of this maneuver. It is strongly believed that the LED point model fell outside of the field of view of the range sensor. During the first flight's barrel roll, the range sensor only tracked the LED point model for 12% of the total time. Referring to Figure 52, the longitudinal acceleration of the helmet experiences a noticeable variation from the aircraft AHRS. Both measurements follow the same trend of increasing and decreasing values, but at the twenty-three second mark the helmet continues to show a decreasing trend in longitudinal acceleration, with the maximum deviation of -0.7 G's from the baseline AHRS. At the thirty-two second point the aircraft is completely inverted at the top of the barrel roll. No explanation for the helmet's measurements is available. This has to be due to a deviation of the helmet frame from being in line with the aircraft frame. Figure 53 also shows similar deviations for the measured lateral accelerations. The helmet acceleration is a maximum -0.7 G's less than recorded aircraft values. There are two humps that occur in the helmet data that is not present in the aircraft AHRS data. Figure 54 shows the vertical accelerations of the helmet and aircraft for the barrel roll maneuver. It is quite clear that the measurements overlay one another very well, sans the transferred helmet accelerations.

The second flight includes two barrel rolls. It is evident from both flights that the helmet is expressing an offset from the aircraft reference frame. The deviations present in the first flight's barrel roll are not a singular event. It is likely that it was difficult for the flight test engineer to keep a steady head position that was aligned with the aircraft's reference frame during the barrel roll maneuver. Similar to the lateral accelerations from the first flight's barrel roll, there were deviations from the baseline AHRS measurements in the second flight's lateral accelerations. Through process of elimination, it was determined that the deviations present in the longitudinal and lateral accelerations are coupled. This is probably because the flight test engineer kept his head level with the aircraft reference frame without an offset in roll and pitch, but with an induced left yaw offset. The flight test engineer may have been inclined to look out towards a center reference point in the distant sight picture that the pilot uses to fly a well coordinated barrel roll.

5.4. Left Knife-Edge Maneuver

The purpose of flying the aircraft in a knife-edge maneuver is to put the aircraft and helmet lateral axes in line with the earth's gravity vector. There were two variations of the knife-edge, both left and right. Ideally, both helmet and aircraft lateral measurements should line up if the flight test engineer manages to keep the helmet reference frame in line to that of the aircraft reference frame. Figure 55 shows a successful alignment of helmet lateral accelerations with the aircraft AHRS lateral accelerations from flight two. The range sensor did not record enough data during this maneuver, so helmet lateral acceleration data transformed to the aircraft reference frame has large variations from the baseline AHRS data. The range sensor only worked for 18% of the maneuver in the first flight, and 5% of the maneuver from the second flight. The left knife-edge maneuver was repeated for flight two, with very similar results. The negative accelerations of Figure 55 correspond with the observed event. Positive accelerations in the lateral direction would be out the right ear while looking forward. The vertical accelerations for the left knife-edge maneuver from flight two are shown in Figure 56. It is important to note that for the first time of several events, there is an apparent delay in helmet measurements from the aircraft measurements. This delay can be determined by measuring the peak-to-peak distance in time. The first peak-to-peak distance is an offset of about one second. This is observable at the thirty second mark. The second definable peak-to-peak offset is 1.5 seconds in length. This is observable at the forty second mark. So in the length of about ten seconds, the timing of the helmet measurements with aircraft measurements has deviated by about an additional 0.5 seconds. This timing offset was not observed during the first test flight. Being that the second test flight was the first to fully record all parameters from both the helmet systems as well as the aircraft EFIS hardware, it is likely that there is a hardware bottleneck in the UMPC preventing data from all measurement sources being written to file at the same exact time.

5.5. Right Knife-Edge Maneuver

Similarly to the left knife-edge maneuver, the purpose of the right knife-edge was to, at best effort; excite a measureable acceleration isolated to the lateral axis. The comparison between helmet and aircraft lateral acceleration measurements is satisfactory, as shown for the first flight in Figure 57. Again, the range sensor did not provide good data due to the LED point model location relative to the range sensor's field of view. A second attempt at the right knife-edge maneuver was performed in the second flight. The vertical acceleration results are

shown in Figure 58. The helmet and aircraft accelerations in this figure show effects of recording time displacement that began in the left knife-edge maneuver earlier from the second flight, as shown in Figure 56. For the right knife-edge maneuver, the offset has increased to two seconds, with the helmet measurements being recorded after the aircraft measurements. However, it would seem that if the two curves could be realigned with one another, a very acceptable overlay would be seen. This realignment would have given another piece of strong evidence that the Gyrocube IMU is functioning to the standards of the baseline AHRS. However, this growing time offset between helmet and aircraft measurements during the second flight is discouraging. It would be easy to implement a function in the software that would compensate for the delay of helmet measurement recording, but since the time offset is increasing at a non-constant rate, no compensating function can be used.

5.6. Inverted Flight

The main purpose of inverted flight for helmet to aircraft comparison was to investigate negative G-loading along the vertical axes by isolating G-forces to only this axis and only in the negative direction. Looking at Figure 59 from the first flight, both helmet and aircraft measurements are aligned. The range sensor did not provide sufficient data, so helmet measurement transformation to the aircraft axis was not possible. For the second flight, the same measurements for vertical accelerations in inverted flight were taken. The time offset issue is escalating as the offset is increasing from its first appearance in the left knife-edge maneuver of flight two. At the beginning of the inverted flight maneuver, the offset is three seconds. By the end of the inverted flight maneuver, helmet measurements are lagging behind aircraft measurements by ten seconds.

5.7. Bridge Maneuver

The bridge maneuver was somewhat of a repeat of the inverted flight maneuver. The vertical acceleration measurements are shown in Figure 60. Everything lines up well, except for the helmet accelerations in the aircraft frame, due to loss of the range sensor effectiveness during the bridge maneuver. As expected, the vertical accelerations shown in Figure 60 measure around -1 G while the aircraft is inverted. The 45 degree up-line and 45 degree down-line are evident at the ten second mark and thirty-three second mark, respectively. A bridge maneuver was flown in the second flight. The results are very similar to those described in Figure 60. During the bridge

maneuver of flight two, the data abruptly ended mid-maneuver. This is because there was a system lockup. It is proposed that the ongoing time offset that had been exponentially growing during the second flight was caused by insufficient system resources in the UMPC. No evidence is available to determine if the bottleneck was caused by software or hardware. Once the UMPC and the Labview VI in execution became too overloaded, the lockup event occurred. The Labview VI was restarted in flight and the test plane for the second flight continued.

5.8. Left Flat Turn Maneuver

The purpose of the left flat turn maneuver was to generate a noticeable yaw and lateral acceleration that could be recorded by both the helmet IMU and aircraft AHRS. In Figure 61, the lateral accelerations measured from the first flight are plotted. There is a great consistency between the helmet and aircraft measurements. Secondly, the range sensor was working, so helmet lateral accelerations were properly transformed to the aircraft reference frame. The range sensor effectively tracked the LED point model for 62% of the maneuver time during the first flight. Also, for the second flight, the sensor tracked the point model for 75% of the total maneuver time. These transformed accelerations line up with the aircraft accelerations. Figure 61 is a very satisfying plot for showing lateral acceleration measurements for the helmet and aircraft, in correlation to one another. The left flat turn maneuver from the second flight was another successful exhibit of correlating helmet and aircraft lateral acceleration data together. The range sensor was working and the transformed helmet accelerations overlaid the baseline aircraft AHRS data.

5.9. Right Flat-Turn Maneuver

This maneuver is the same as the previous left flat turn maneuver, but in the opposite direction. The results from the right flat turn were also very similar to the left flat turns, except for both right flat turns the range sensor did not provide the data necessary for helmet to aircraft reference frame transformation. Comparing flights one and two, there was some improvement. For the right flat turn from flight one, the range sensor only worked 25% of the time. For the second flight, the range sensor worked for 54% of the length of the maneuver. The lateral accelerations for the right flat turn from the second flight are shown in Figure 62. This plot shows a good overlay of the helmet lateral acceleration in the helmet reference frame with the aircraft lateral acceleration in the aircraft frame.

5.10. Slow Roll Maneuver

The slow roll maneuver was performed to investigate both lateral and vertical accelerations. A slow roll was performed during the first flight but the data acquisition VI had a lockup and the slow roll data record was incomplete. To make up for this deficiency, two slow rolls were performed in the second test flight. Figure 63 shows the lateral accelerations for the first slow roll performed in flight two. Both helmet and aircraft lateral accelerations lined up, except for the helmet acceleration in the aircraft frame. This is due to the absence of range sensor data for the first slow roll. There was a good overlay of data for both helmet and aircraft accelerations. Additional plots can be found in Appendix B. Again, the transformed helmet accelerations were unreliable without proper range sensor data.

The second slow roll took place immediately following the first slow roll. Figure 64 shows the roll rates recorded by both helmet and aircraft AHRS during the second slow roll. At the point of entry to the maneuver, the aircraft was rolled sharply to the left, as evident in Figure 64 at the twenty second mark. The aircraft is banked sharply to exit the slow roll and re-level the aircraft. Both helmet and aircraft roll rates seem to be overlaid very well except for the fact that the helmet roll rates do not reach the maximums of the aircraft during the two noticeable banking motions.

5.11. Hesitation Roll Maneuver

The hesitation roll was flown to investigate roll rate and vertical acceleration elements. The hesitation roll is a full roll but with four separate hesitations, or stopping points. Each hesitation is preceded by a ninety degree roll. For the first flight, the roll rates are shown in Figure 65. The roll rates in Figure 65 are of particular interest because the four rotations are clearly evident at about the six, eight, ten and eleven second marks. The helmet roll rate is about an average of ten degrees per second less than the aircraft roll rate. However, the general shape and trend of both curves show an acceptable overlay. For the actual maneuver, which began at the four second mark and ended at the thirteen second mark, the range sensor data was not available. For the second flight, the vertical accelerations are shown in Figure 66. Just like the hesitation roll from the first flight, the LED point model was out of the range sensor's field of view, so helmet measurements transformed to the aircraft frame were not available during the second flight's hesitation roll. The time offset from previous maneuvers from the second flight is evident in Figure

66. The helmet roll rate data is lagging behind aircraft recorded measurements by about one second, as evidenced in the peak-to-peak displacements. If the helmet data could be fixed and shifted back in time by one second, it looks as though an acceptable alignment of helmet and aircraft data would occur. If helmet data could be readjusted to be in sync with aircraft data, it appears as though a good alignment or satisfactory comparison would exist.

5.12. Loop Maneuver

The loop maneuver was only flown in the second test flight. It was a good maneuver for examining helmet and aircraft relationships for pitching motion and vertical accelerations. The range sensor did not provide helmet attitude data for the extents of the loop maneuver. The vertical accelerations for the loop maneuver are shown in Figure 67. The maneuver begins at the twenty second mark and concludes at the fifty second mark. A very concise overlay for helmet measurements and aircraft measurements is apparent. Both helmet IMU and aircraft AHRS report the two peak accelerations of about 3.5 G's and also the zero-G vertical acceleration at the top of the loop while the aircraft is inverted at the thirty-eight second mark.

5.13. Square Loop Maneuver

The square loop maneuver is a variation of the previously discussed loop maneuver. The square loop is unique because the pitching motion of the aircraft is separated into four quarter loops. Between each quarter loop is a vertical line or a horizontal line. The maneuver is flown such that an observer sitting perpendicular to the plane of the square loop would observe the flight path of the aircraft to be almost a square. The range sensor did not provide good data during the square loop maneuver of the first flight or for the second flight. The pitch rate from the second flight's square loop maneuver is shown in Figure 68. The four quarter loops and their respective pitch rates are evident as four small humps, each no more than twenty-five degrees per second at their maximum rates. Reported helmet pitch rates and aircraft pitch rates align perfectly. Data from flight two's square loop is very similar to the data from the same maneuver of flight one.

5.14. Hammerhead Maneuver

The hammerhead maneuver was the final aerobatic maneuver performed for each test flight. The main objective of flying the hammerhead maneuver was to generate a yawing motion of the aircraft that could be measured by both the helmet IMU and aircraft AHRS. The range sensor did not supply substantial data during the hammerhead maneuver for flight one. The yaw maneuver at the top of the vertical line occurs at the forty second mark, as shown in Figure 69. Both helmet yaw rate data and aircraft yaw rate data overlay one another for the duration of the maneuver. The yaw rate went from zero yaw to a maximum yaw rate of negative fifty degrees per second, indicating a counter-clockwise yawing motion which also reflects the actual observed event. This data from flight one also is representative of the hammerhead event in flight two. Further flight data is available in Appendix B.

6. Conclusions and Future Research

6.1. Conclusions

The primary objective of the present research was to develop and flight test a self-contained, portable, inertial and positional measurement system for an HGU-55 fighter pilot helmet. Secondary objectives included recording the motion of the helmet during various flight maneuvers and utilizing the data to increase the understanding of the human head in a maneuvering flight environment. The portable helmet instrumentation system was built to record helmet inertial data using a head-mounted IMU, and helmet positional data using an optical range sensor. The aircraft AHRS was integrated into the PHIS to provide aircraft inertial data.

The PHIS was tested on two flights. The IR LED range sensor did not operate as planned. Only 10,609 out of 28,645 data points, or 37% of all data, from the first test flight were collected while the helmet was within the field of view of the range sensor. The total number of data points collected during the second flight when the range sensor was available was 10,288 points out of 29,548 points total, or 35% of the total data. Of all three axes, the vertical axis had the greatest correlation for helmet IMU measurements and the baseline aircraft AHRS

measurements. The PHIS operated well during level flight and during the left flat turn maneuver. This was the case for both flights.

The safety and feasibility of the Portable Helmet Instrumentation System was tested during two aerobatic flights in the UTSI Aviation Systems Extra 300 aircraft. A configuration change so that the PHIS is worn by the pilot would be helpful in evaluating the full functionality of the system. The physical limitation of the camera's field of view does not accommodate use for a test subject in the front seat of the Extra 300 aircraft. As the system was proposed to be worn by a pilot, it is the next logical step in testing its full operability. If this is difficult to implement, the pilot can wear the helmet and an additional crew member can monitor and control the UMPC device from the front seat.

It is believed that the delayed recording of the helmet data was due to hardware limitations imposed by the Samsung Q1 UMPC. The current handheld computer has a CPU speed of 600 MHz and one gigabyte of RAM. In order to fix this issue, faster hardware is required if the current Labview software is not modified. Otherwise, significant optimization of the Labview code must be made to increase its computational efficiency. Special programming techniques could be implemented to reduce the runtime of each iteration in the Labview code main loop.

If equipping the pilot with the PHIS is not a desirable design goal, then in order to make the PHIS fully operational for a front seat passenger, a whole new camera system will have to be developed. The Nintendo Wii-Remote Controller may have to be replaced because of its limited field of view. A candidate replacement would be the NaturalPoint TrackIR 5, newly released in 2009, as it is the only other commercially available camera that offers the same frames per second and sensor resolution of the Wii-Remote. However, all NaturalPoint products require some CPU usage, which may be a problem for the already over-loaded processing work required on the Samsung Q1 UMPC. In addition, there is no known existing interface between the TrackIR application programming interface (API) and Labview. Additional work must be undertaken to program some kind of interface between the TrackIR data and the existing Labview code.

The original PHIS concept proposed a self-contained data system that can be attached to a fighter pilot helmet that would "measure linear accelerations and angular rates of the helmet/head, displaying and recording the data in real time." All system components were required to be attached to the helmet and/or helmet wearer. This requirement was not met because an additional element such as an aircraft-mounted IMU or AHRS was required.

6.2. Future Recommendations

A method of point-tracking is required to define the helmet attitude, if not only for orientation correction, but possibly for attitude measurement in its entirety. One solution to meet this requirement is reversing the locations of the range sensor and infrared LEDs. The IR-LEDs would be installed in the cockpit and the range sensor would be installed on the helmet. Using available algorithms, similar to Intel's OpenCV (Open Computer Vision), now called Intel IPP (Integrated Performance Primitives), a method of motion estimation can be used to determine orientation and position. An image sequence (video) from a head-mounted camera can be processed to produce an estimate of the velocity of points in the image, both rotational and translational, which in turn could provide a velocity estimate of the camera and thus the helmet. There are many available algorithms for a visual tracking system. Two for example are blob tracking, which makes correlations of large groups of pixels, and contour tracking, which looks for object boundaries. LED installation in the cockpit would not have to be permanent. These could be clip-on points and a calibration routine could be devised to quickly re-teach the helmet tracker for each new cockpit setup.

The current system requires integration with the aircraft AHRS in order to perform differential inertial measurements. The necessity of being connected to the aircraft could be redesigned for by an addition of a second IMU that is worn by the pilot between the thigh and the seat cushion. This option may not prove to be as accurate as a hard-mounted IMU or AHRS, as the pilot may shift his weight or sitting position during flight. This might be avoided by clamping the second IMU to the seat and not to the pilot.

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Appendix A

Table 1: Measured Helmet Parameters [13]

Parameter	Limits	Resolution*	Rate (Hz)
Linear Accelerations (X, Y, Z)	$\pm 10\text{ g}$	0.005 g	TBD (up to 48 Hz)
Angular Rates (p, q, r)	$\pm 400\text{ deg/ sec}$	0.2 deg/ sec	TBD (up to 48 Hz)

*Based on 12 bit Data Acquisition System

Table 2: Six Position Method Calibration Results

	X Scale Factor	X Bias (G's)	Y Scale Factor	Y Bias (G's)	Z Scale Factor	Z Bias (G's)
Calibration #1	0.740963	3.93079	0.82722	2.36314	0.76891	-4.86698
Calibration #2	0.735972	4.01074	0.833072	2.43108	0.768259	-4.80203
Average	0.7384675	3.970765	0.830146	2.39711	0.7685845	-4.834505
	Wx Scale Factor	Wx Bias (deg/s)	Wy Scale Factor	Wy Bias (deg/s)	Wz Scale Factor	Wz Bias (deg/s)
Calibration #1	1.0395	-20.7426	1.05945	8.29838	1.00813	-14.9054
Calibration #2	1.06121	-20.7074	1.05989	8.10747	1.0038	-15.1895
Average	1.050355	-20.725	1.05967	8.202925	1.005965	-15.04745

Table 3: Calibration Results for In-Field User Calibration Method

Accelerometers	X Scale Factor	Y Scale Factor	Z Scale Factor	X Bias (G's)	Y Bias (G's)	Z Bias (G's)
Average	1.289864333	1.265327667	1.310005333	0.631651333	0.510811333	-0.212636333
Standard Deviation	0.035145065	0.01515855	0.003632722	0.015270447	0.00784443	0.005889968
	1.322887	1.247913	1.306396	0.615431	0.517301	-0.209691
	1.252925	1.272509	1.313661	0.64575	0.513039	-0.2088
	1.293781	1.275561	1.309959	0.633773	0.502094	-0.219418
Gyroscopes	Wx Scale Factor	Wy Scale Factor	Wz Scale Factor	Wx Bias (deg/s)	Wy Bias (deg/s)	Wz Bias (deg/s)
Average	1.012666667	1.032951333	1.003547667	-19.583175	6.856642333	-12.431584
Standard Deviation	0.01751376	0.044017267	0.017801677	0.172974876	0.263907091	0.241342008
	0.998571	1.042803	0.98467	-19.728543	7.072772	-12.455755
	1.007156	1.071208	1.020031	-19.391868	6.934623	-12.179066
	1.032273	0.984843	1.005942	-19.629114	6.562532	-12.659931

Table 4: In-Field and Six Position Calibration Methods Comparison

	X Scale Factor	Y Scale Factor	Z Scale Factor	X Bias (G's)	Y Bias (G's)	Z Bias (G's)
In-Field Method	1.289864333	1.265327667	1.310005333	0.631651333	0.510811333	-0.212636333
Six Position Method	0.7384675	0.830146	0.7685845	3.970765	2.39711	-4.834505
Difference	0.551396833	0.435181667	0.541420833	3.339113667	1.886298667	4.621868667
	Wx Scale Factor	Wy Scale Factor	Wz Scale Factor	Wx Bias (deg/s)	Wy Bias (deg/s)	Wz Bias (deg/s)
In-Field Method	1.012666667	1.032951333	1.003547667	-19.583175	6.856642333	-12.431584
Six Position Method	1.050355	1.05967	1.005965	-20.725	8.202925	-15.04745
Difference	0.037688333	0.026718667	0.002417333	1.141825	1.346282667	2.615866

Table 5: Calibration Coefficients for IR Range Sensor

Reported Distances	Scale Factor	Bias (centimeters)
Longitudinal Position	1.0192	0.1754
Lateral Position	-1.9627	-0.3974
Vertical Position	-1.3854	0.003
Reported Angles	Scale Factor	Bias (radians)
Roll Angle	1.1771	0.006
Pitch Angle	1.5193	0.0008
Yaw Angle	2.2793	0.0371

Table 6: Flight Test Aircraft Load and Airspeed Limits [28]

Extra 300 Aircraft Limits			
Airspeed MIN	Airspeed MAX	G-Loading	Design Maneuvering Speed
60 kts	220 kts	+8 g / -8 g	158 kts

Table 7: List of Flight Test Maneuvers

Name	Load Factor (Z-Axis)	Airspeeds (kts) min	Airspeeds (kts) MAX ENTRY	Attitudes	Head-Position	Desired "Excitation"					
						X Axis	Y Axis	Z Axis	Wx	Wy	Wz
Level Flight	-	Vs	Vne	LVL	FWD and LVL						
G-Warmup (Steep Turn)	1 -> 4 g's	100	Vne	60 DEG BANK	FWD			X		X	
Barrell Roll	0.5 -> 3 g's	120	158	LVL, INVERTED, ROLL 360	FWD			X	X		
Knife Edge	1 -> 2 g's	100	158	90 DEG BANK	FWD		X		X		
Inverted Flight	-1 g's	100	158	INVERTED	FWD			X			
Bridge Maneuver	-1 g's	100	158	INVERTED	FWD			X	X		
Flat Turn	-	Vs	100	LVL	FWD		X				X
Slow Roll	-1 -> 1 g's	120	158	INVERTED, 90 DEG BANK	FWD		X	X	X		
Hesitation 4-Point Roll	-1 -> 1 g's	120	158	INVERTED, 90 DEG BANK	FWD		X	X	X		
Looping	-0.25 -> 3, 4 g's	80	158	PITCH UP, INVERTED	FWD & LEFT	X		X		X	
Square Loop	-0.25 -> 3, 4 g's	80	158	PITCH UP, INVERTED	FWD & LEFT	X		X		X	
Hammerhead / Stall Turn	-0.25 -> 2,3 g's	0	158	VERT	FWD & LEFT	X					X

Table 8: Range Sensor Functionality for Test Flight 1

Range Sensor Functionality for Test Flight 1			
Maneuver	Number of points recorded with range sensor	Total recorded points	Percent time range sensor in use during maneuver
entire flight	10609	28644	37%
level flight	181	294	62%
G-warmup	303	737	41%
barrel roll	91	766	12%
knife edge (left)	79	442	18%
knife edge (right)	88	334	26%
inverted flight	35	480	7%
bridge maneuver	113	708	16%
flat turn (left)	453	735	62%
flat turn (right)	158	644	25%
slow roll	27	462	6%
hesitation roll	77	289	27%
square loop	76	707	11%
hammerhead	367	1470	25%

Table 9: Range Sensor Functionality for Test Flight 2

Range Sensor Functionality for Test Flight 2			
Maneuver	Number of points recorded with range sensor	Total recorded points	Percent time range sensor in use during maneuver
entire flight	10288	29547	35%
level flight	745	1496	50%
G-warmup	212	1202	18%
G-warmup #2	6	213	3%
G-warmup #3	74	427	17%
barrel roll	112	1280	9%
knife edge (left)	38	824	5%
knife edge (right)	86	439	20%
inverted flight	128	630	20%
bridge maneuver	311	752	41%
flat turn (left)	762	1013	75%
flat turn (right)	464	858	54%
slow roll	80	559	14%
slow roll #2	118	793	15%
hesitation roll	0	264	0%
square loop	91	718	13%
loop	507	1093	46%
hammerhead	592	1284	46%



Figure 1: F-16 Fighter Pilot Wearing Gentex HGU-55 Helmet [29]

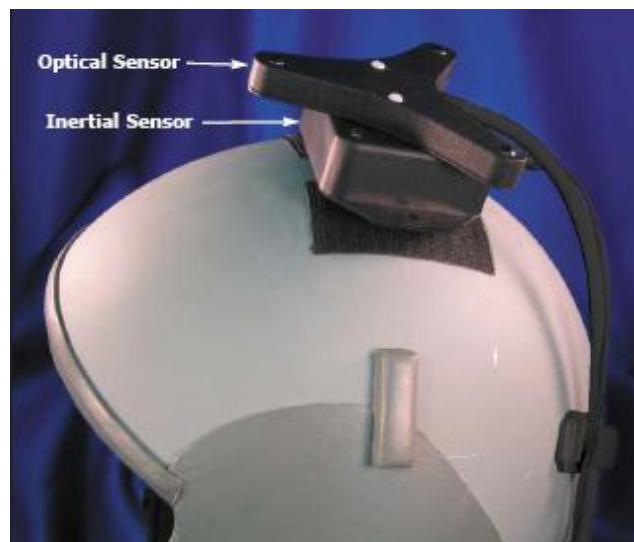


Figure 2: Ascension Technology Hy-BIRD [7]

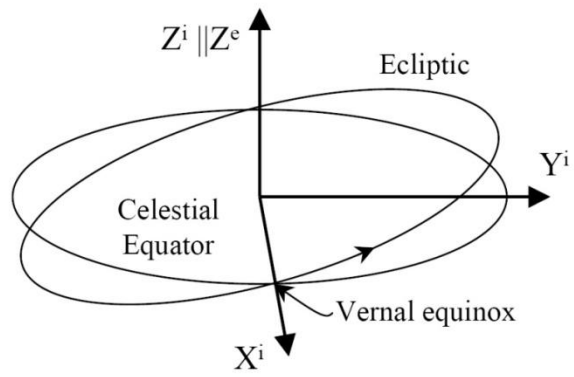


Figure 3: The Inertial Frame [19]

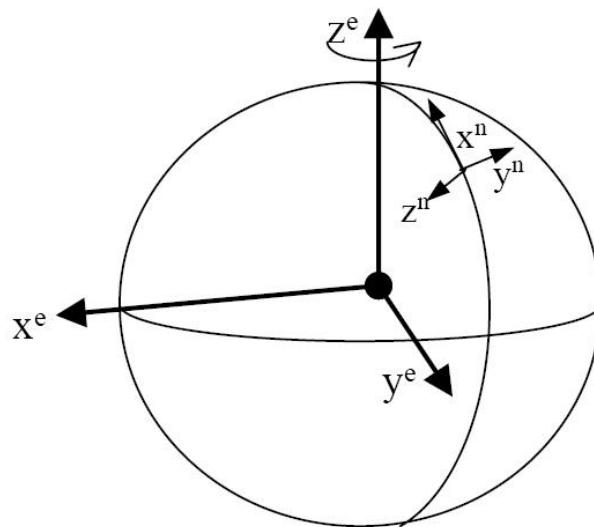


Figure 4: Earth and Navigational Frames [15]

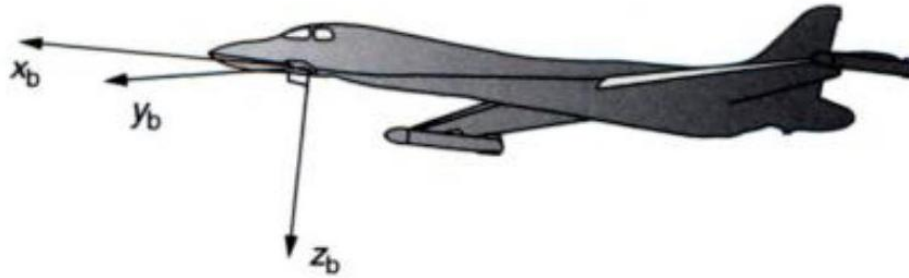


Figure 5: Aircraft Body Frame [17]

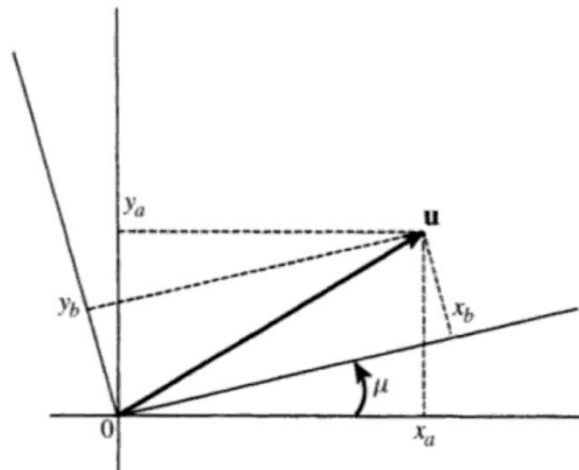


Figure 6: Plane Coordinate Rotation [19]

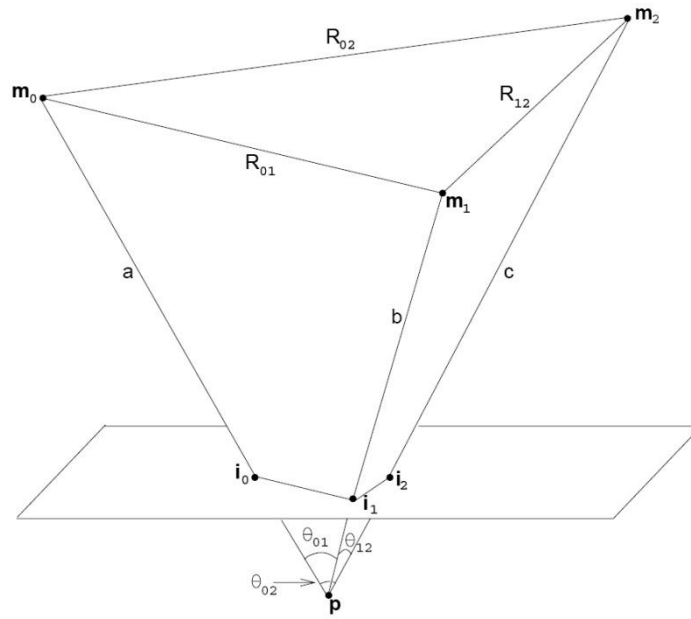


Figure 7: Perspective Projection [18]

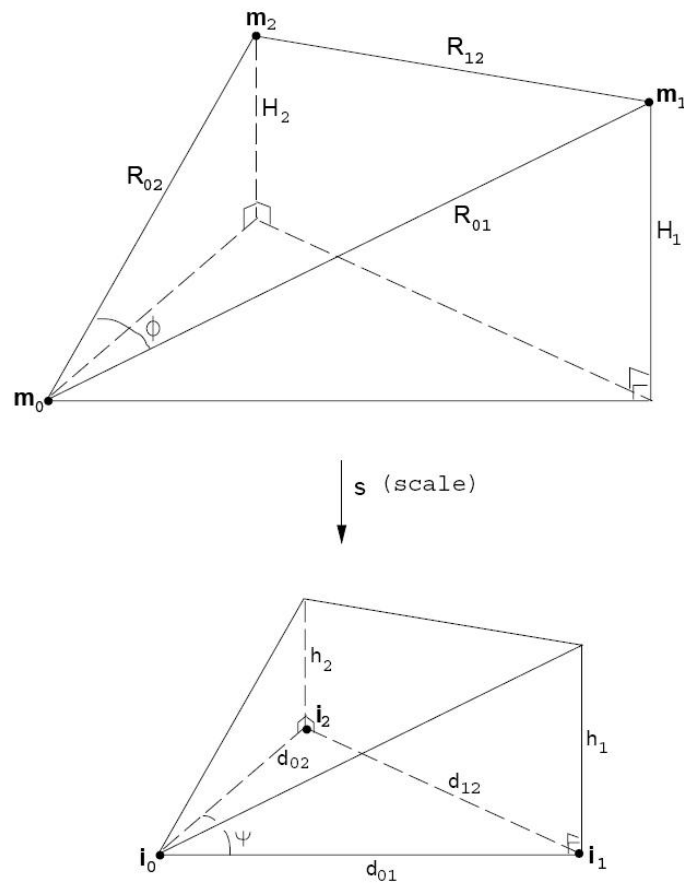


Figure 8: Weak-Perspective Projection [18]

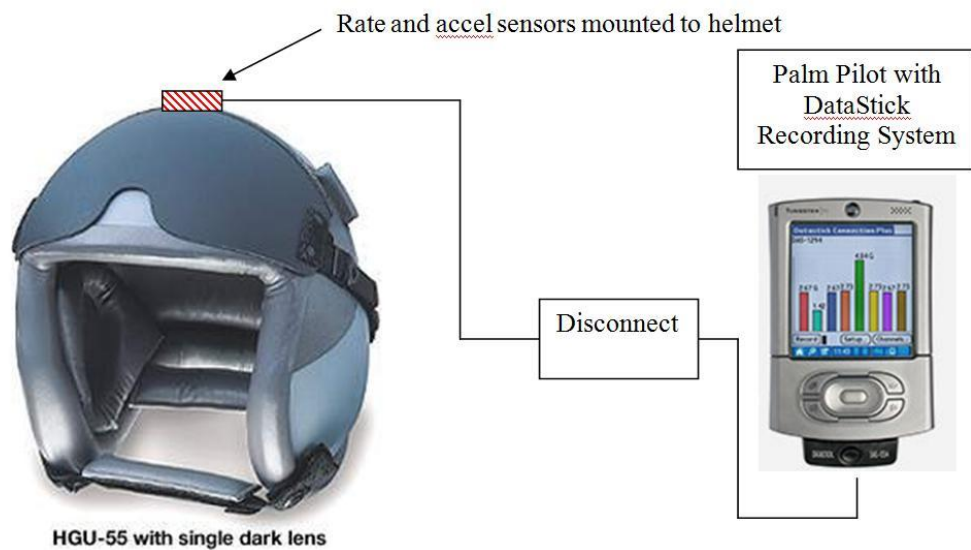


Figure 9: Portable Helmet Instrumentation System Initial Design [13]



Figure 10: UTSI Aviation Systems Customized HGU-55 Helmet

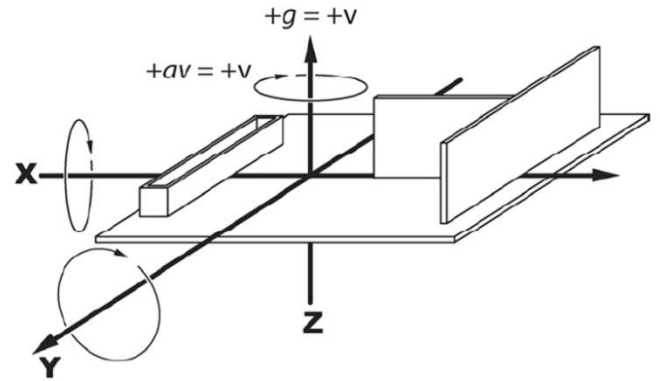
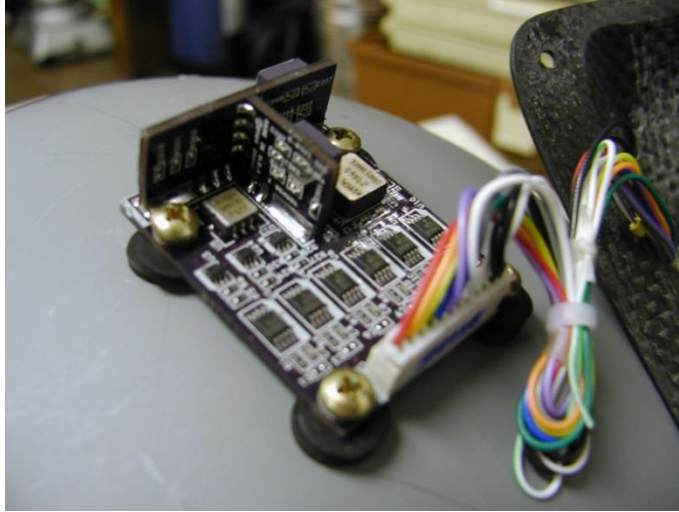


Figure 11: ONavi Gyrocube 3F 6DOF Sensor Module Mounted on Helmet [30]

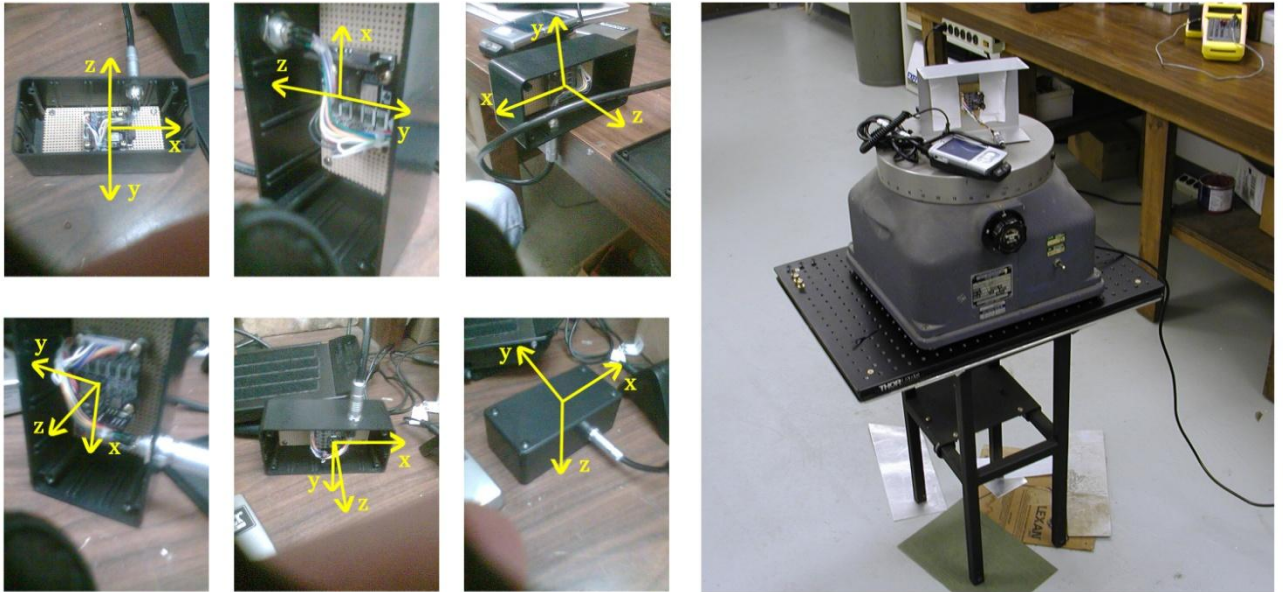


Figure 12: Rate Table Set-up for Six Position Calibration Method

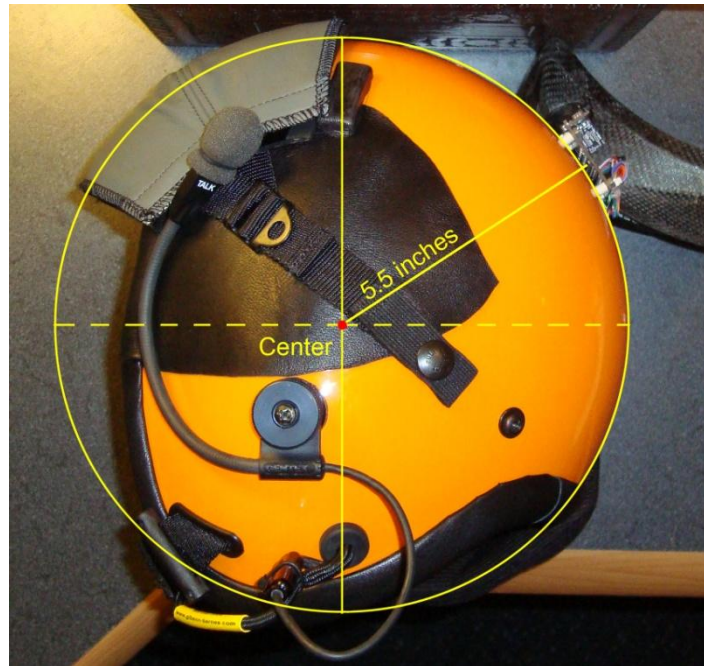


Figure 13: Determining Helmet Center

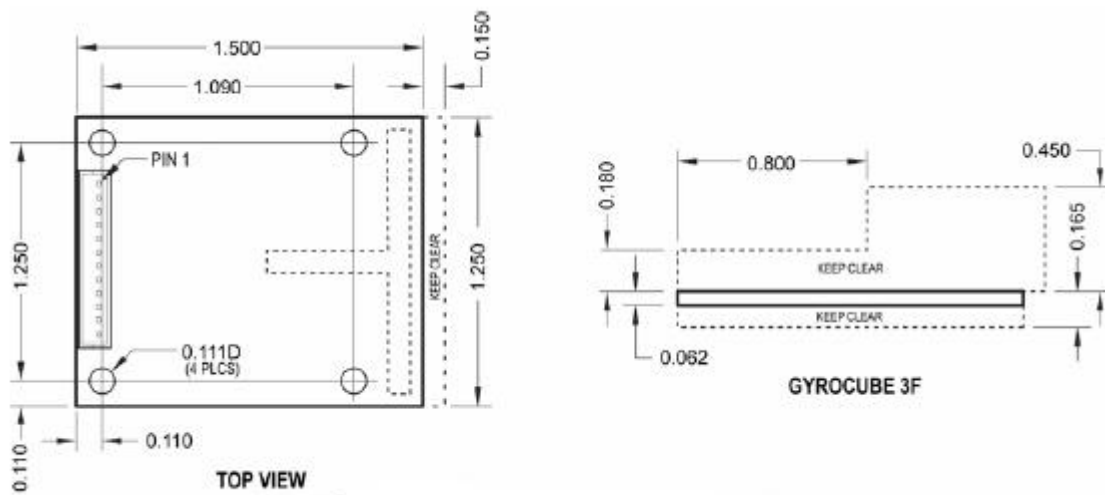


Figure 14: Gyrocube Dimensions [30]

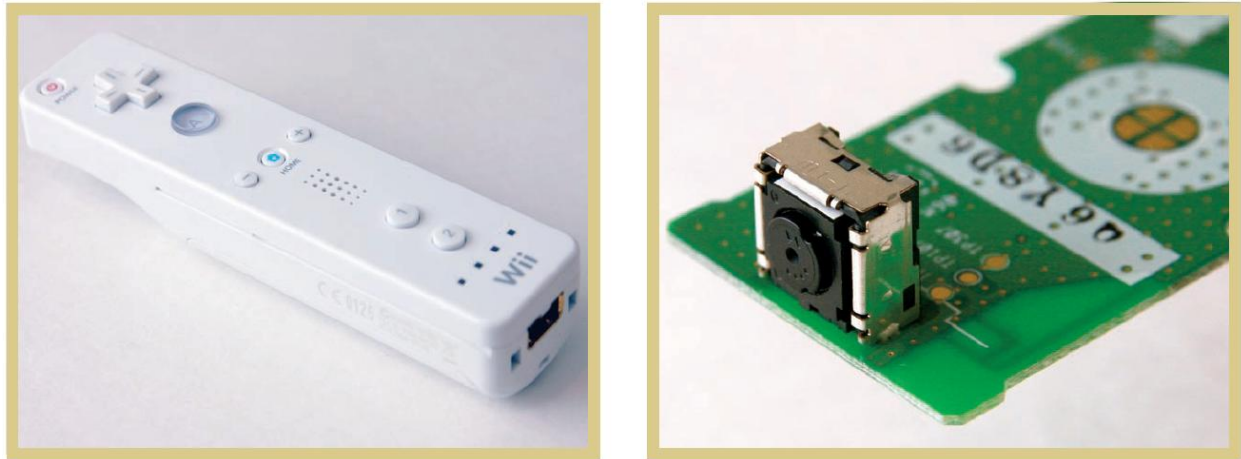


Figure 15: Nintendo Wii-Remote controller and PixArt IR Camera chip [25]

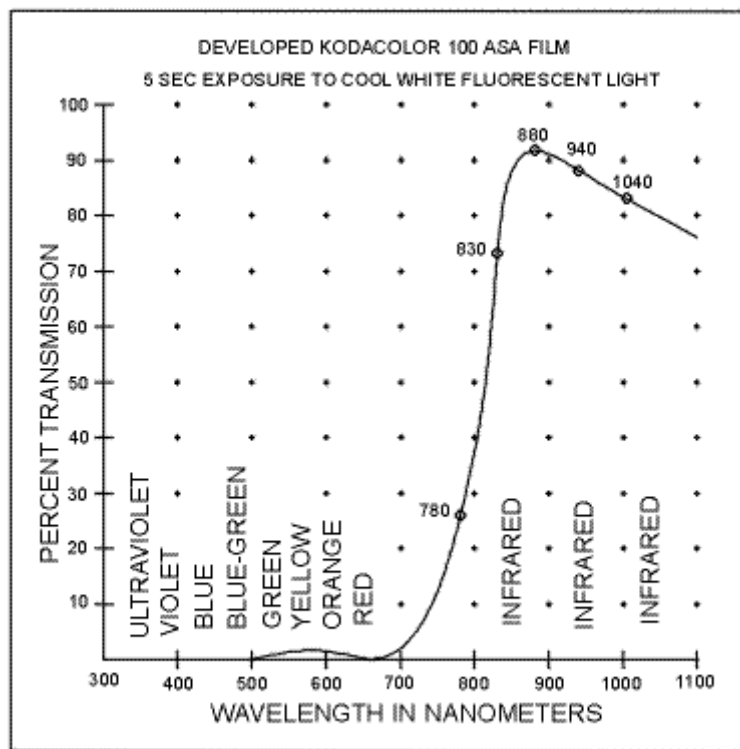


Figure 16: Visible Light Blocking Filter Transmission Characteristics [31]



Figure 17: Initial Installation of Infrared LEDs for use in Rear Seat of Aircraft

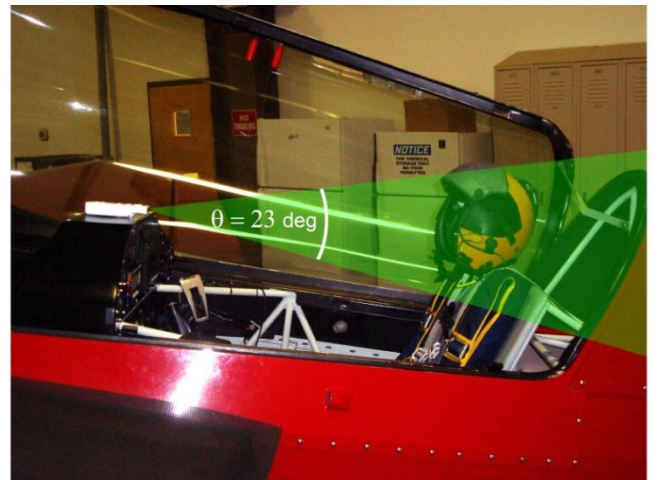


Figure 18: Proposed Locations of Camera in Relation to Forward and Rear Seats of Aircraft



Figure 19: Final Installation of IR-LEDs for use in Aircraft Forward Seat



Figure 20: Upgraded Helmet with Custom Made IR-LED Mount



Figure 21: USB to Serial Hub connection for AHRS in Right-Side, Forward Cockpit

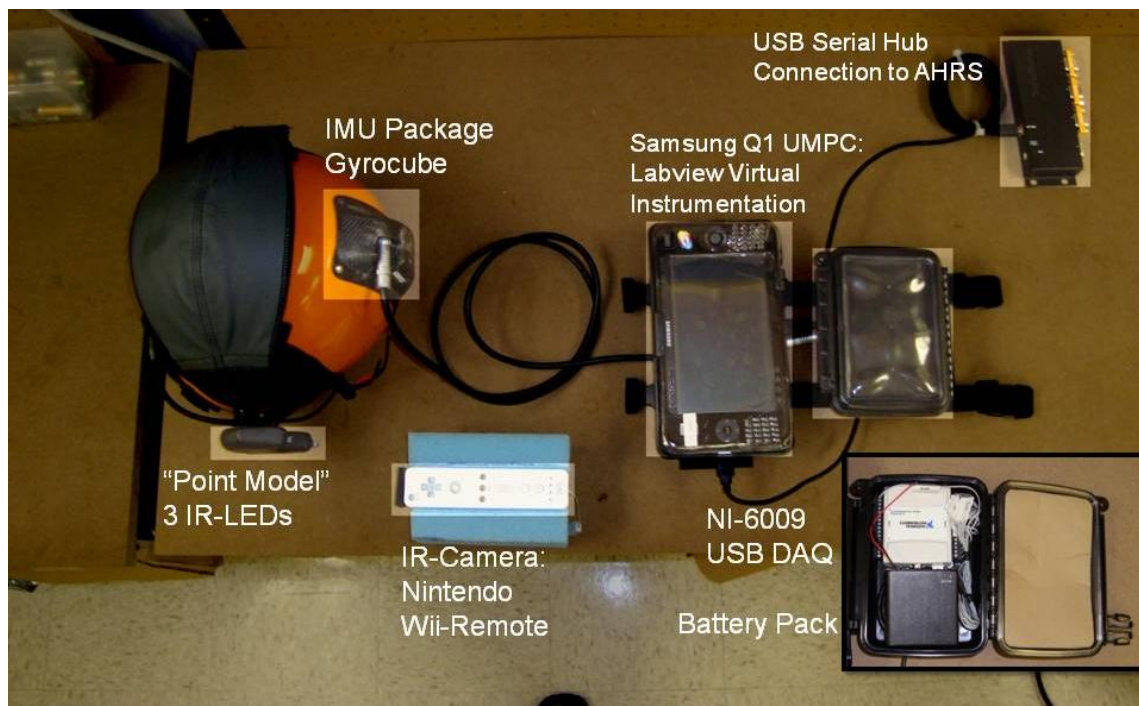


Figure 22: Complete Portable Helmet Instrumentation System



Figure 23: Demonstration of Leg-Mount System

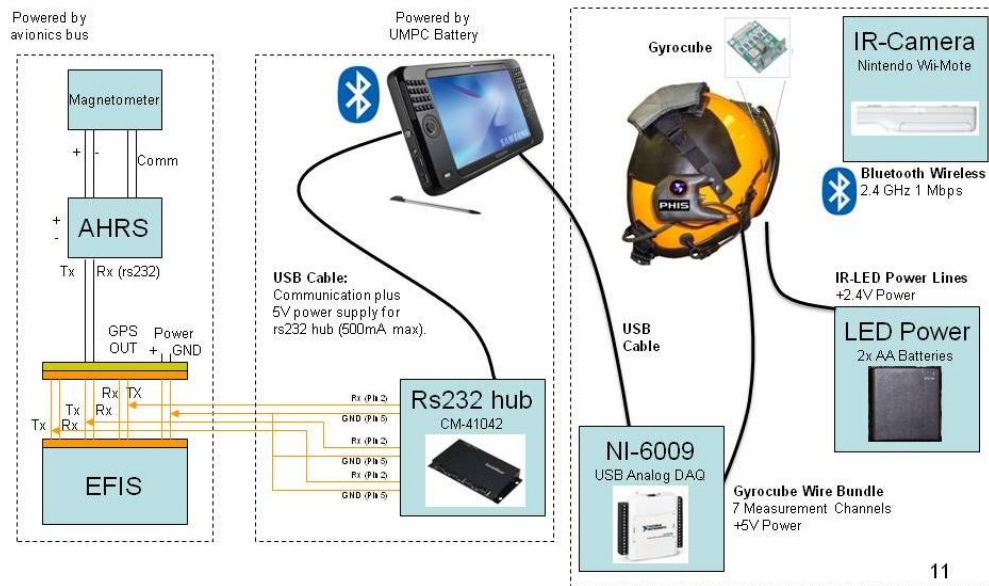


Figure 24: Portable Helmet Instrumentation System Interfaces

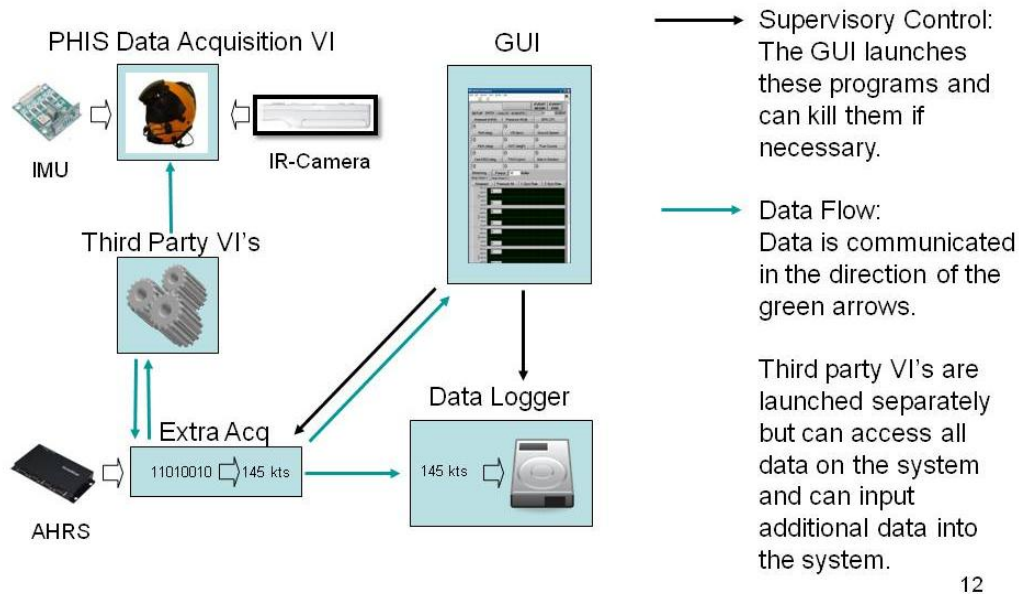


Figure 25: Portable Helmet Instrumentation System Software Structure



Figure 26: Calibrating IR Camera for Roll Angles



Figure 27: Complete PHIS in front seat of Extra 300 Aircraft



Figure 28: Extension Bracket Installation

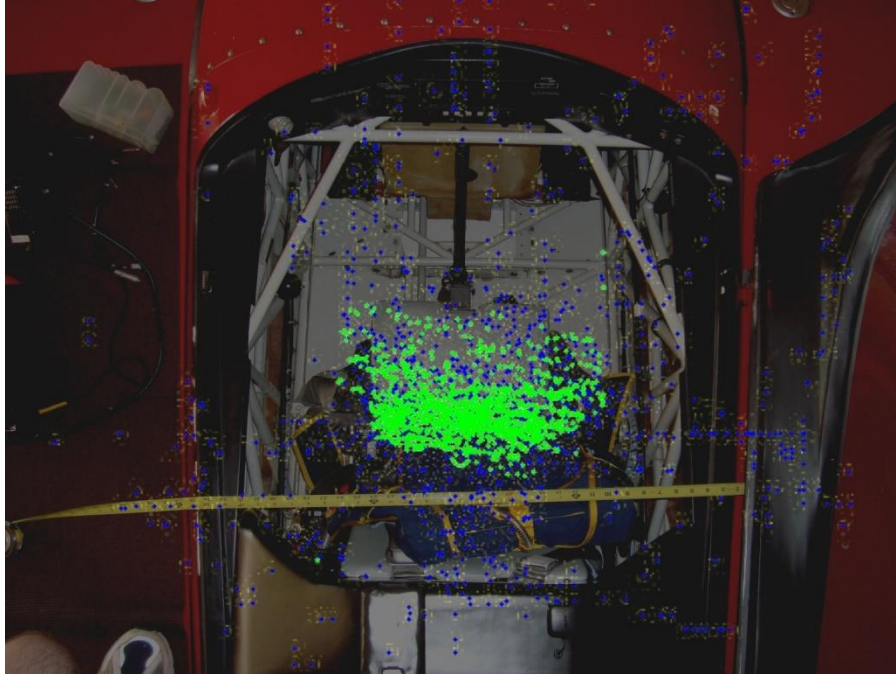


Figure 29: Top-Down View of Helmet Position in Extra 300

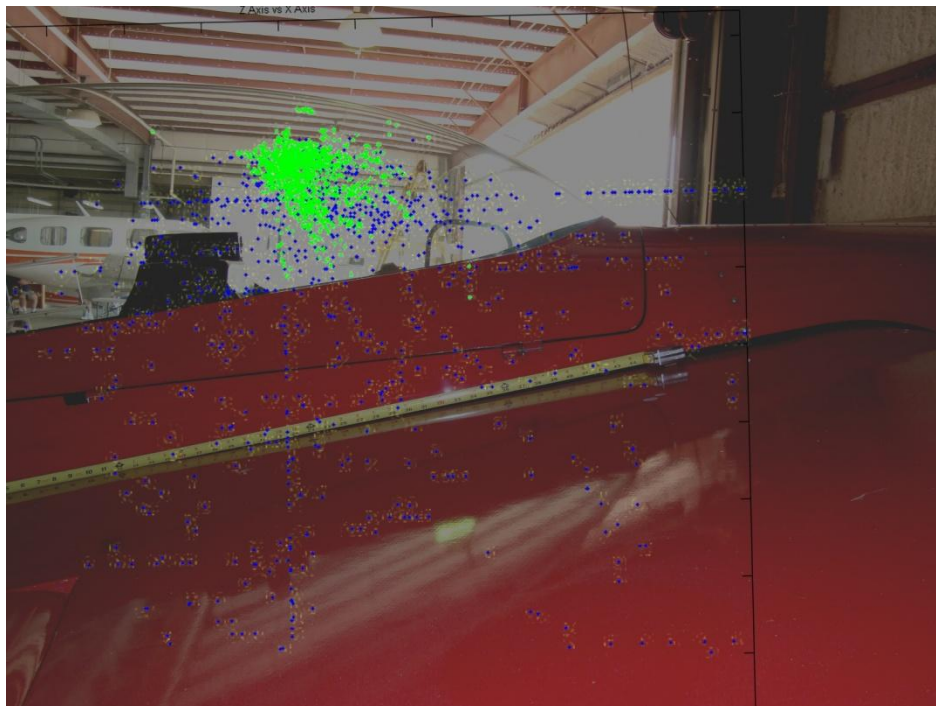


Figure 30: Right-side View of Helmet Position in Extra 300



Figure 31: Frontwards View of Helmet Position in Extra 300

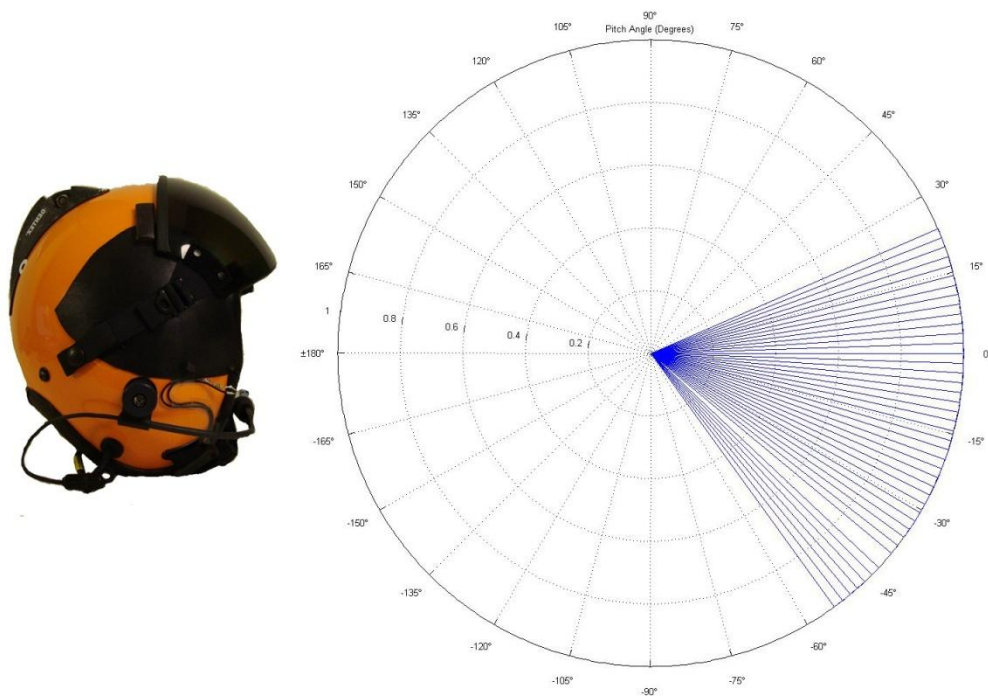


Figure 32: Functional Pitch Angle Range for LED-Based Tracking

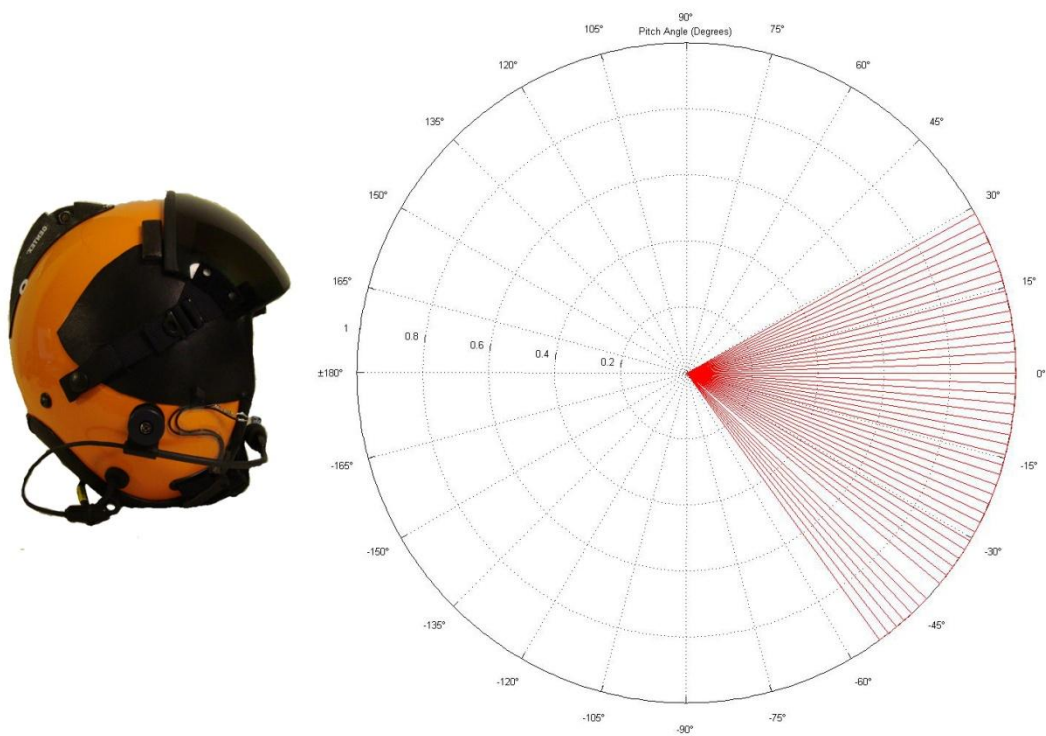


Figure 33: Functional Pitch Angle Range for Inertial and LED-Based Tracking

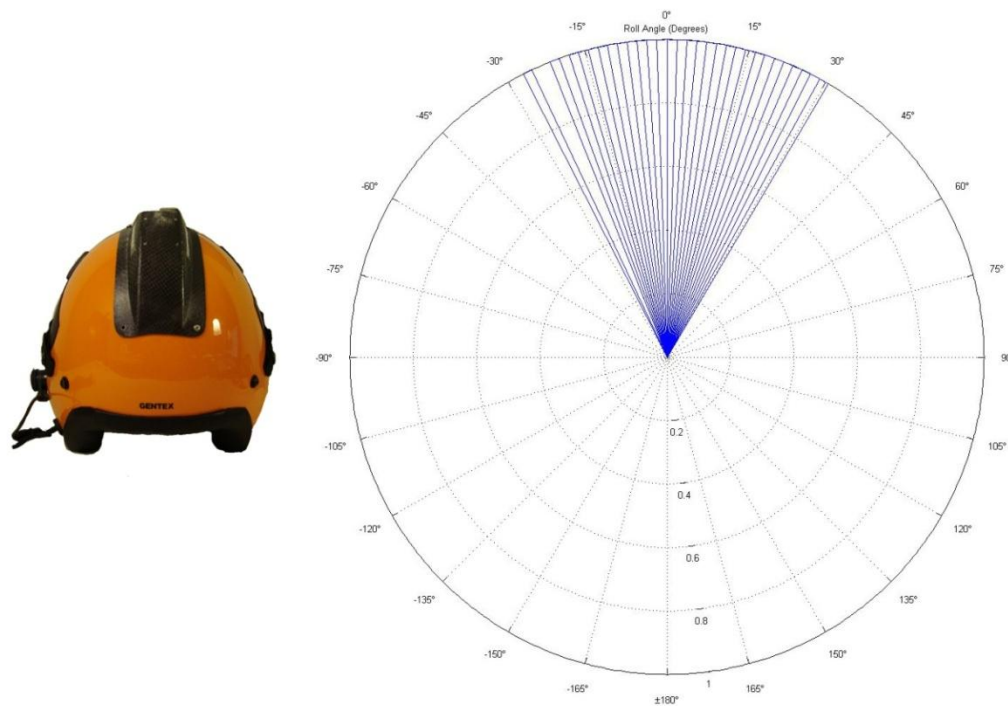


Figure 34: Functional Roll Angle Range for LED-Based Tracking

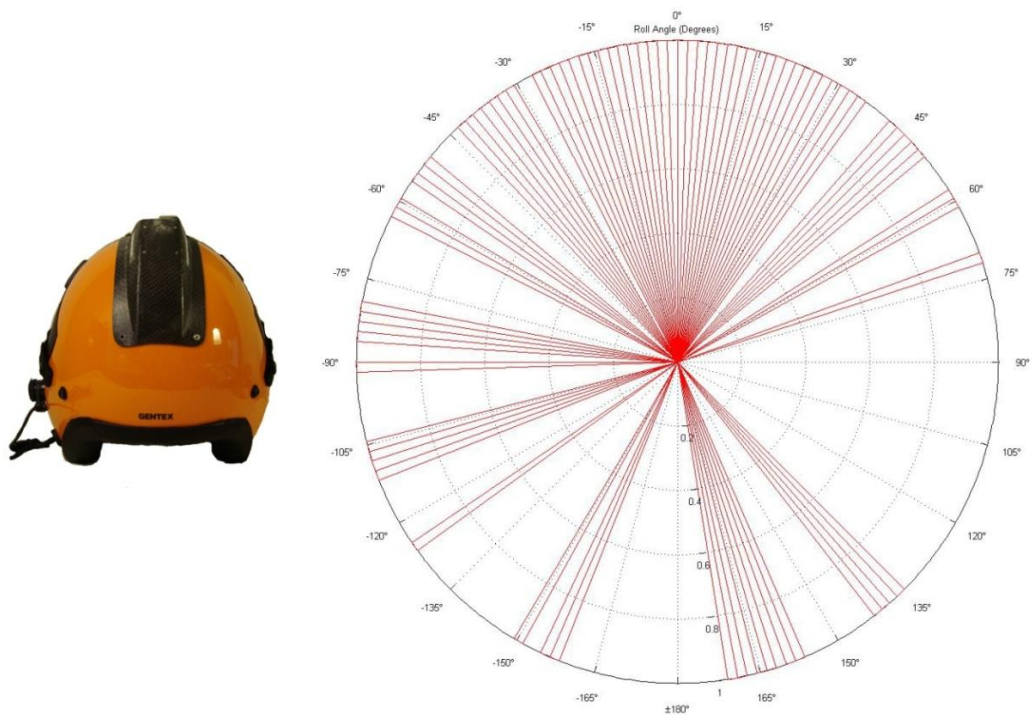


Figure 35: Functional Roll Angle Range for Inertial and LED-Based Tracking

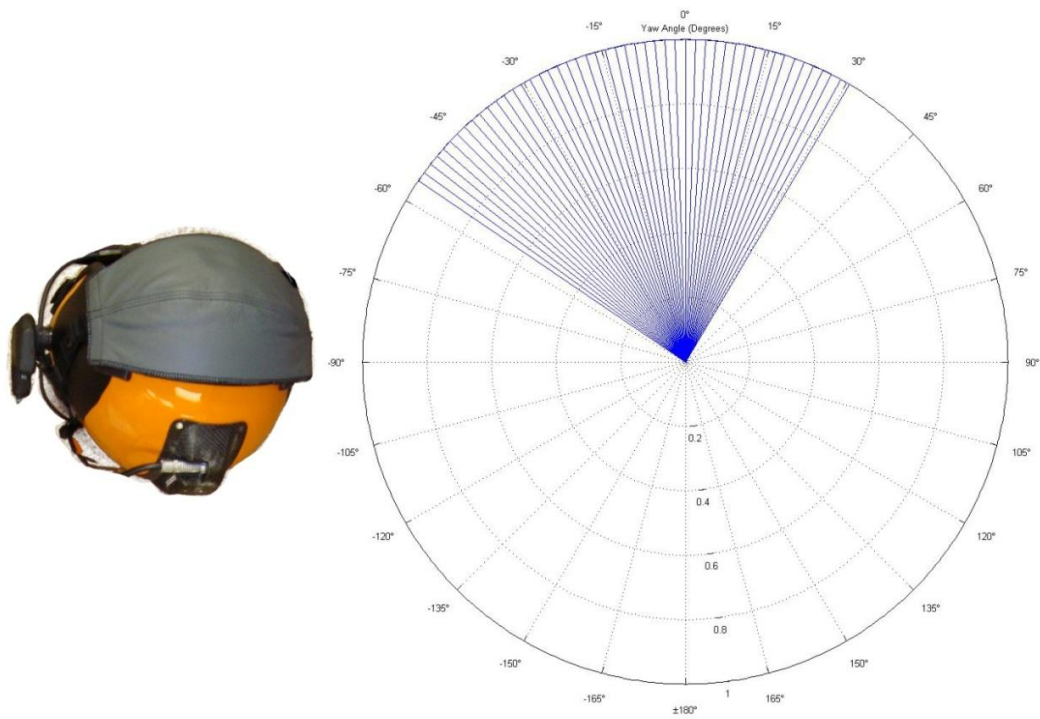


Figure 36: Functional Yaw Angle Range for LED-Based Tracking

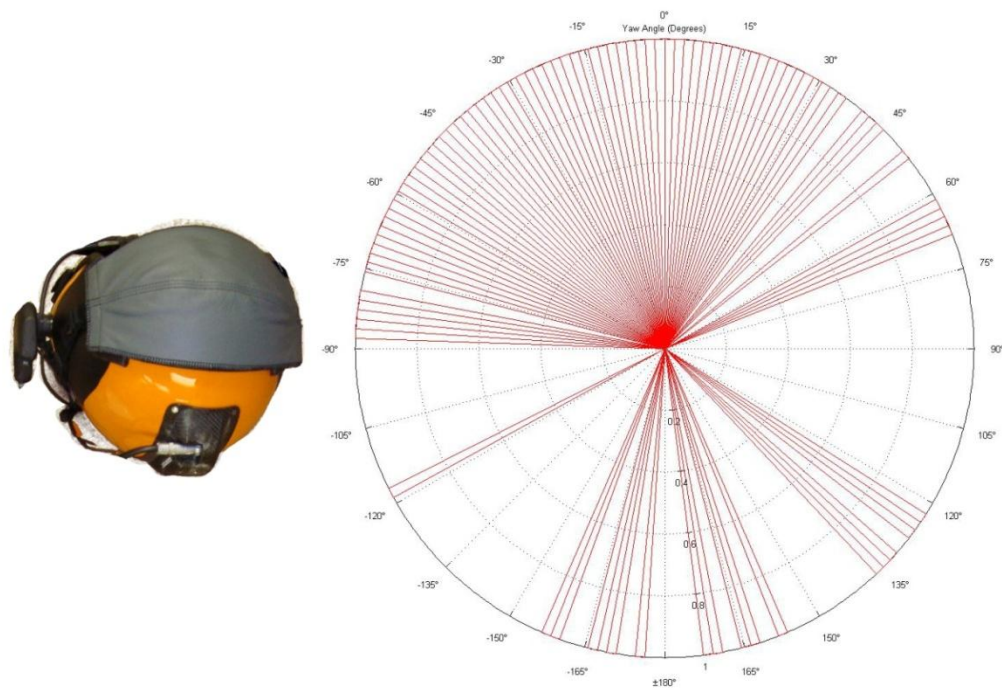


Figure 37: Functional Yaw Angle Range for Inertial and LED-Based Tracking



Figure 38: UTSI Aviation Systems' Extra 300 Aircraft

Longitudinal Acceleration vs Time for Level Flight

Portable Helmet Instrumentation System Test Flight 1
 Extra 300 N714X FTE: Michael Kamp 10:32:06 AM - 10:32:26 AM Date 06/30/2009
 TP: Stephen Corda Standard Day Conditions

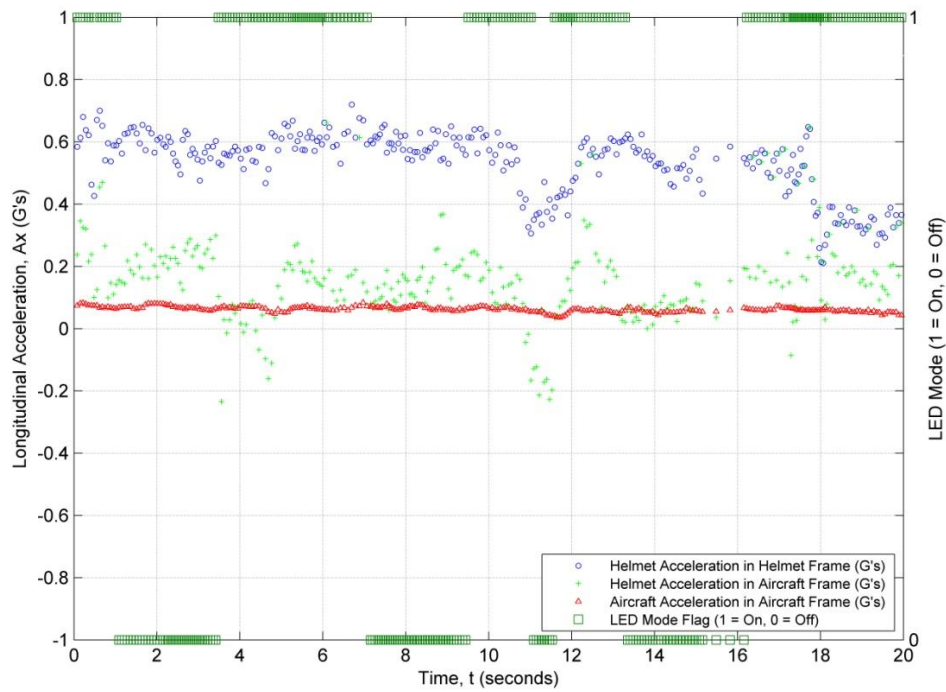


Figure 39: Longitudinal Acceleration versus Time for Level Flight

Roll Rate vs Time for Level Flight

Portable Helmet Instrumentation System Test Flight 1
 Extra 300 N714X FTE: Michael Kamp 10:32:06 AM - 10:32:26 AM Date 06/30/2009
 TP: Stephen Corda Standard Day Conditions

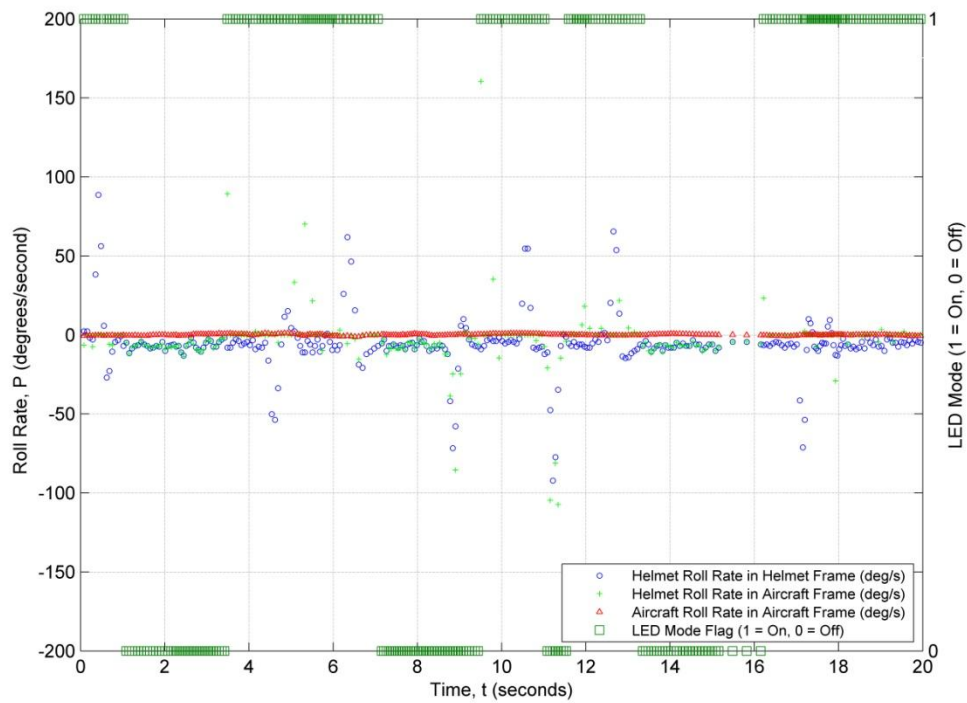


Figure 40: Roll Rate versus Time for Level Flight

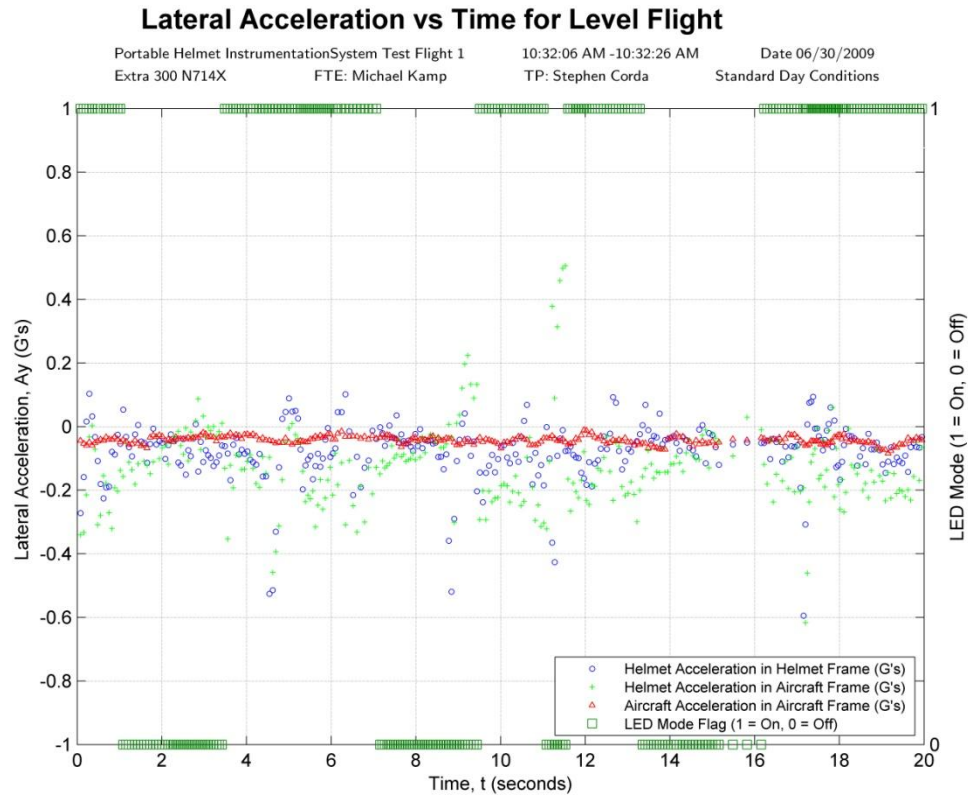


Figure 41: Lateral acceleration versus time for level flight

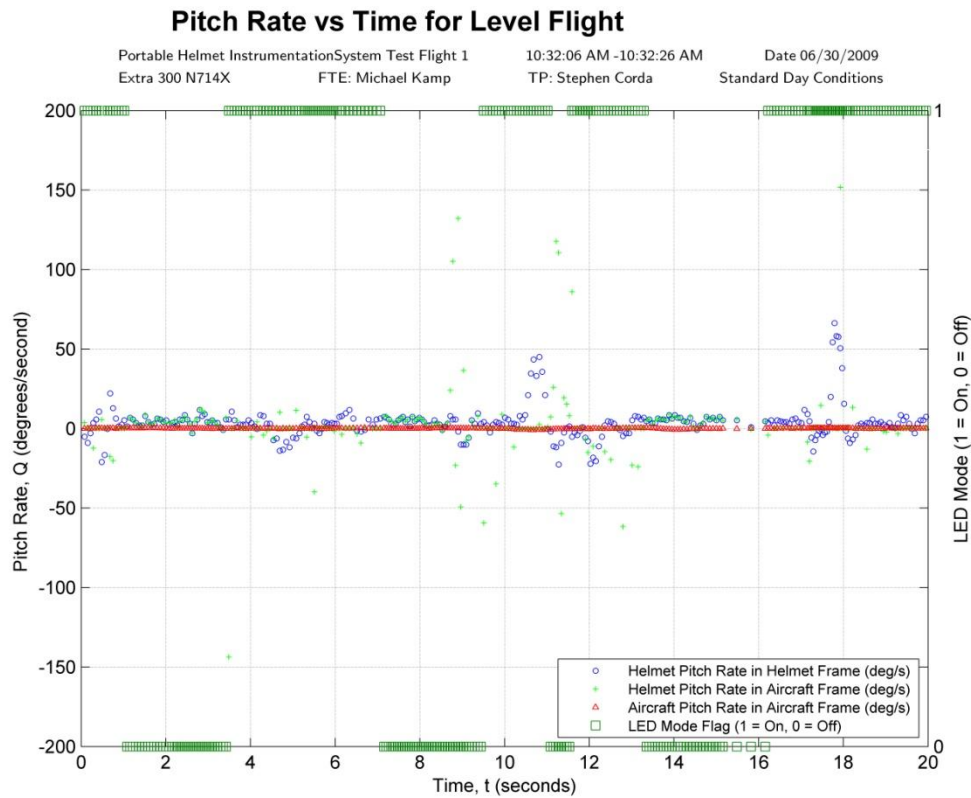


Figure 42: Pitch Rate versus Time for Level Flight

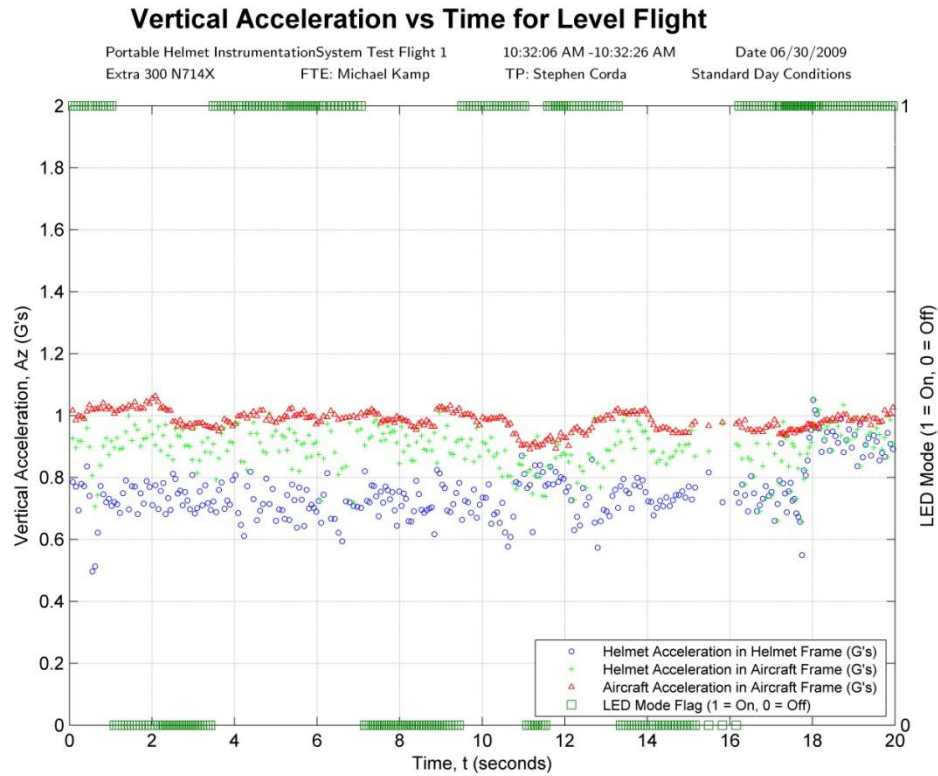


Figure 43: Vertical Acceleration versus Time for Level Flight

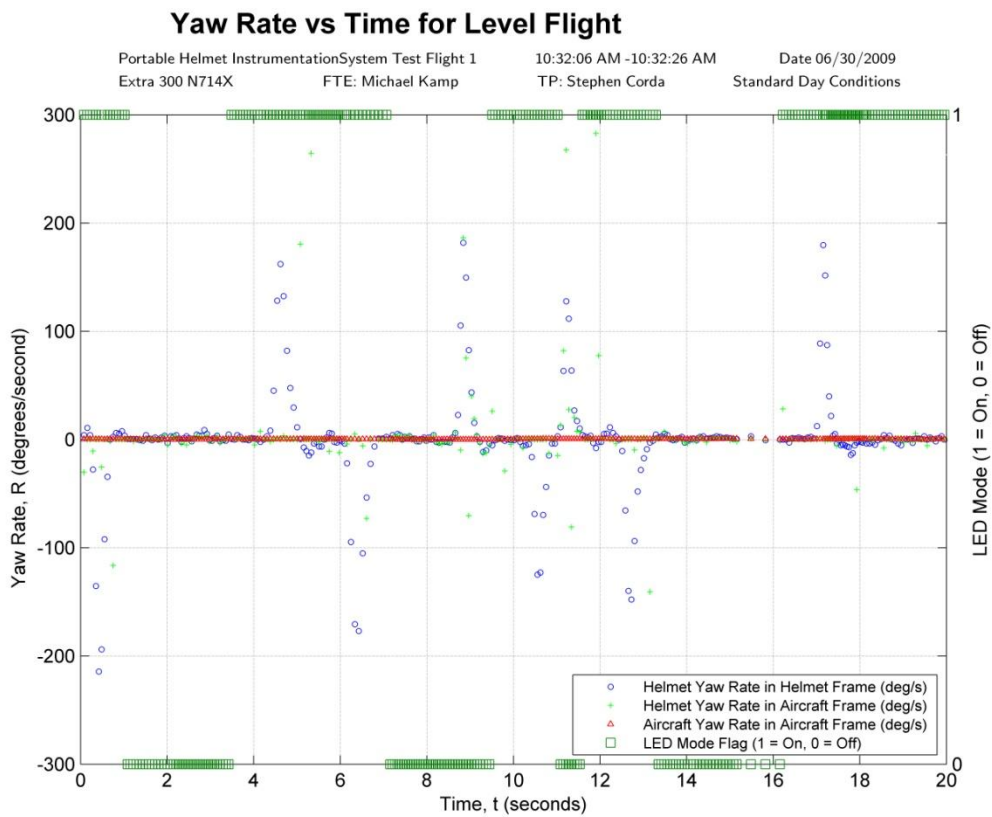


Figure 44: Yaw Rate versus Time for Level Flight

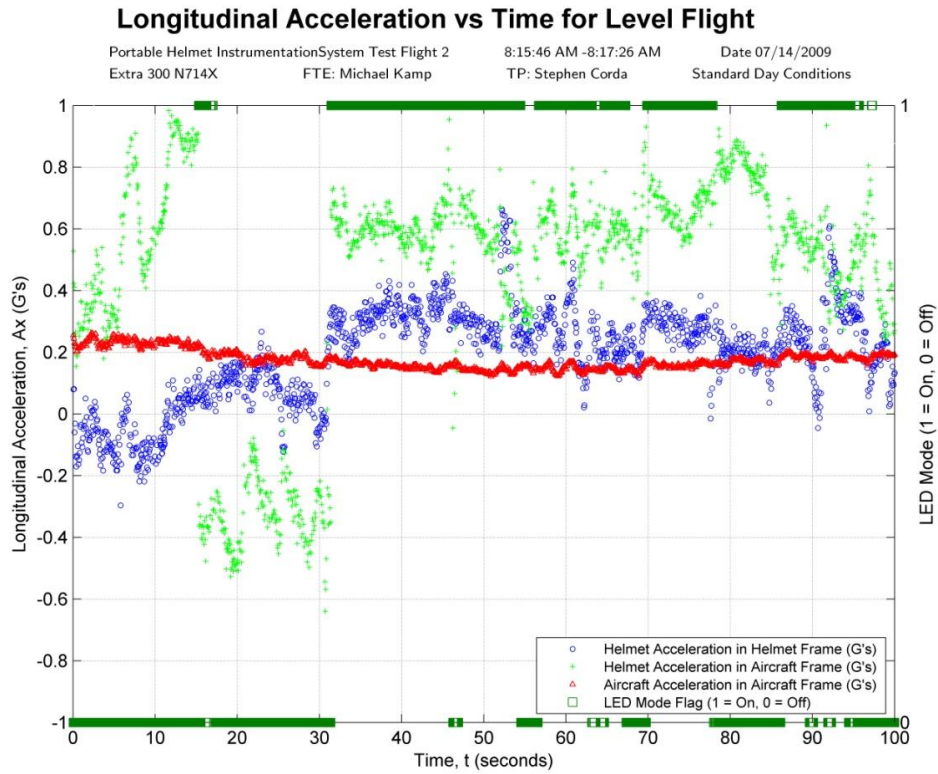


Figure 45: Longitudinal Acceleration versus Time for Level Flight



Figure 46: Roll Rate versus Time for Level Flight

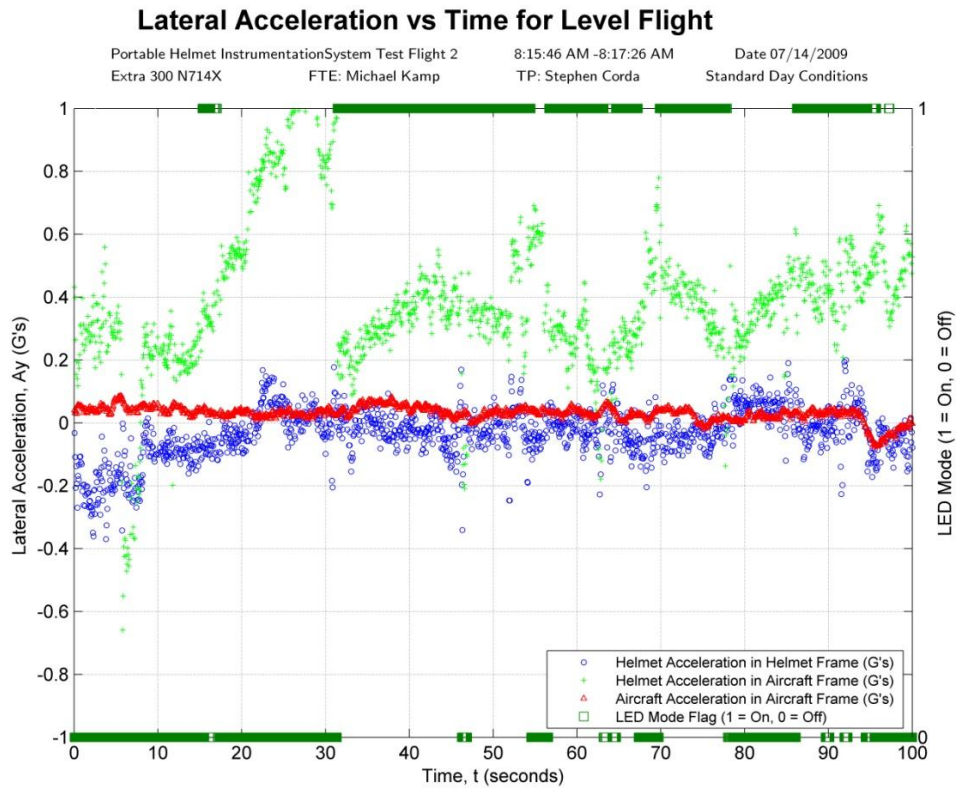


Figure 47: Lateral Acceleration versus Time for Level Flight

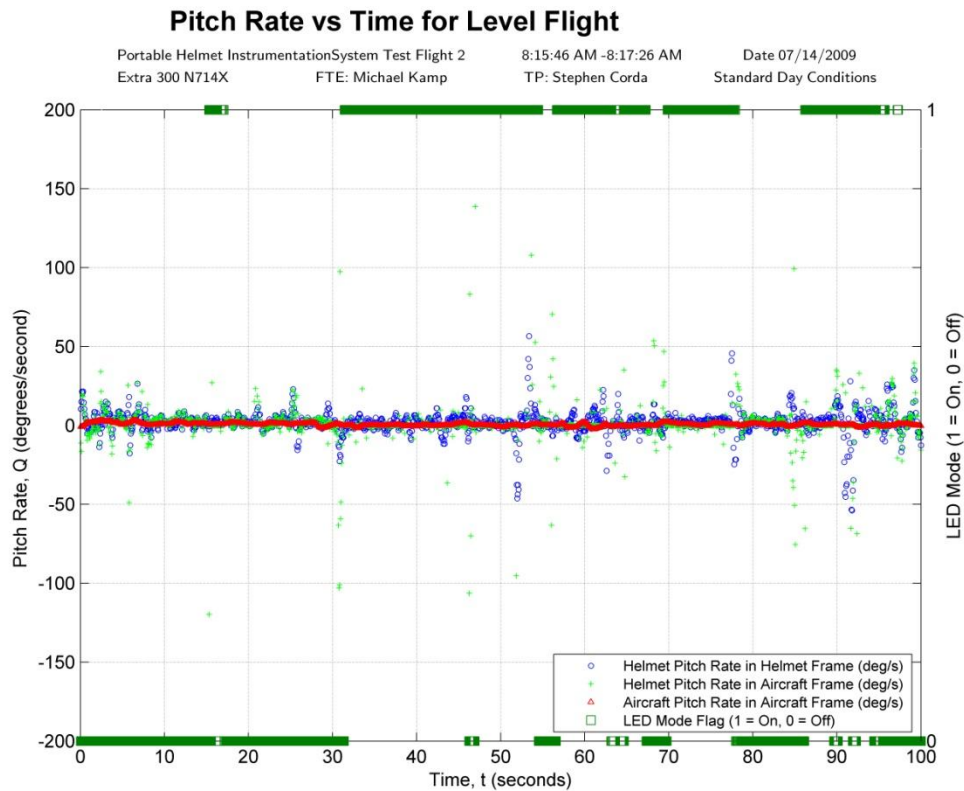


Figure 48: Pitch Rate versus Time for Level Flight

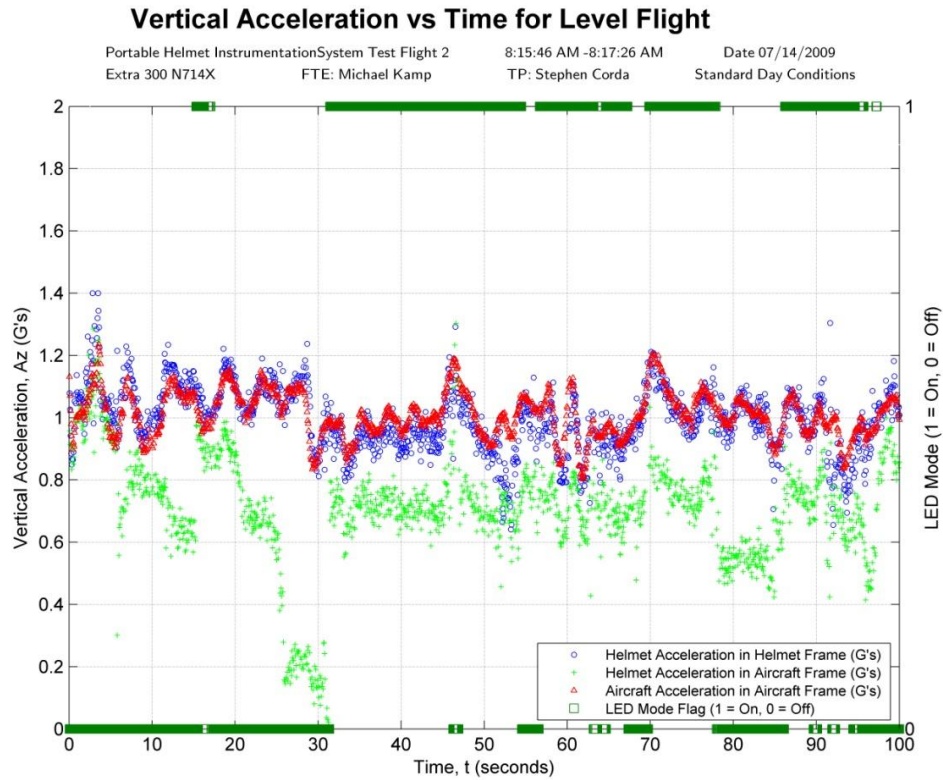


Figure 49: Vertical Acceleration versus Time for Level Flight

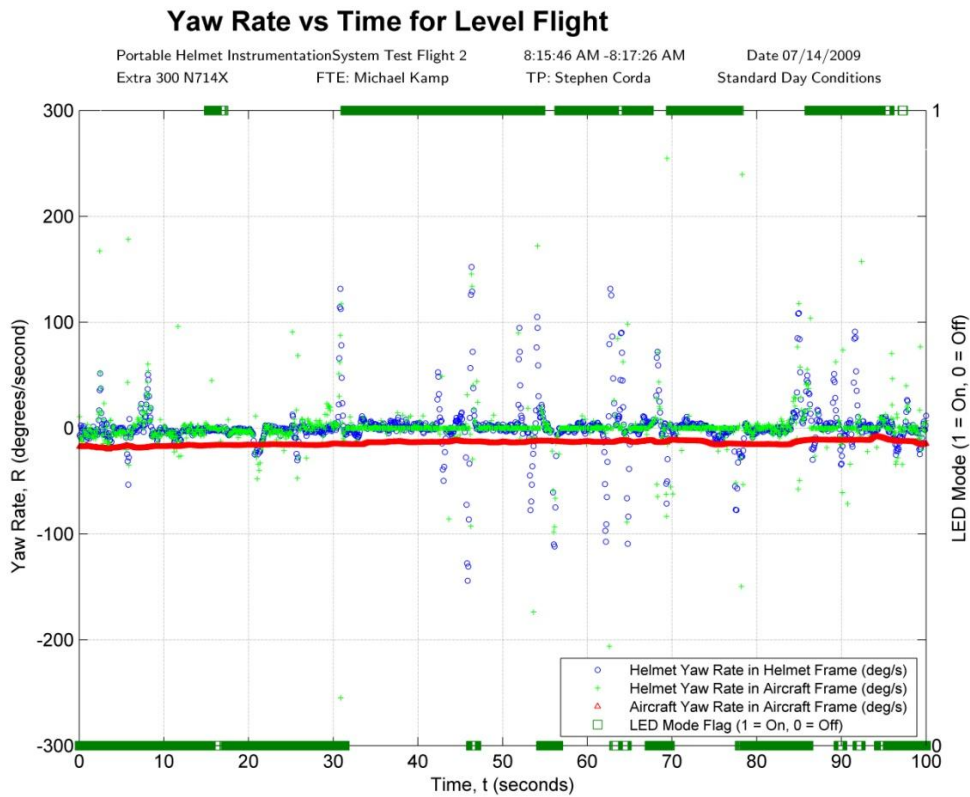


Figure 50: Yaw Rate versus Time for Level Flight

Longitudinal Acceleration vs Time for G-Warmup Maneuver

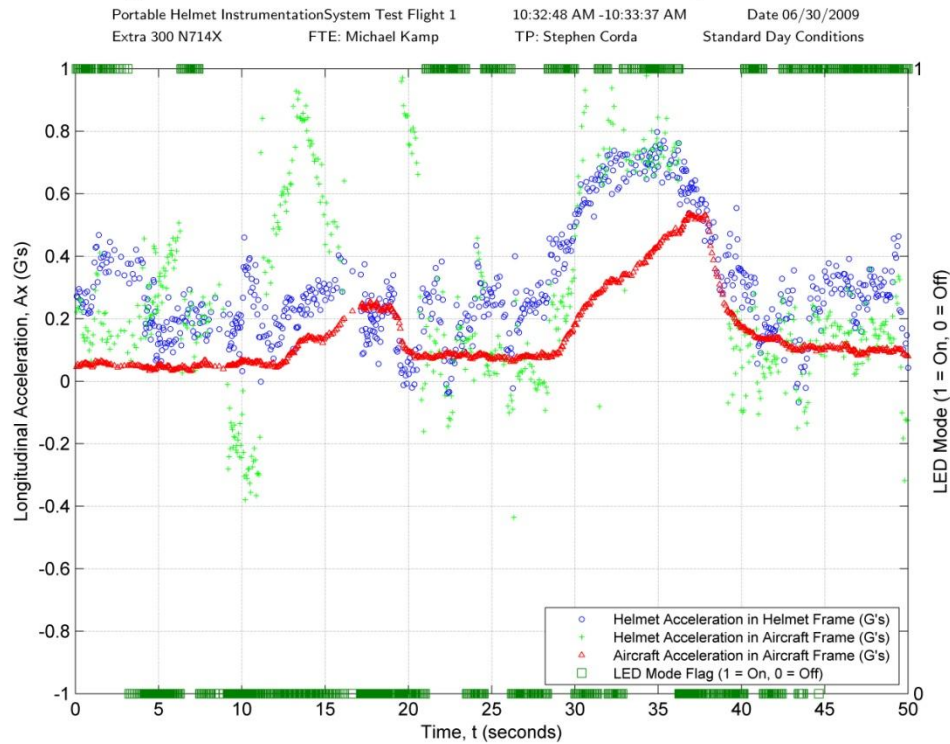


Figure 51: Longitudinal Acceleration versus Time for G-warmup Maneuver

Longitudinal Acceleration vs Time for Barrel Roll Maneuver

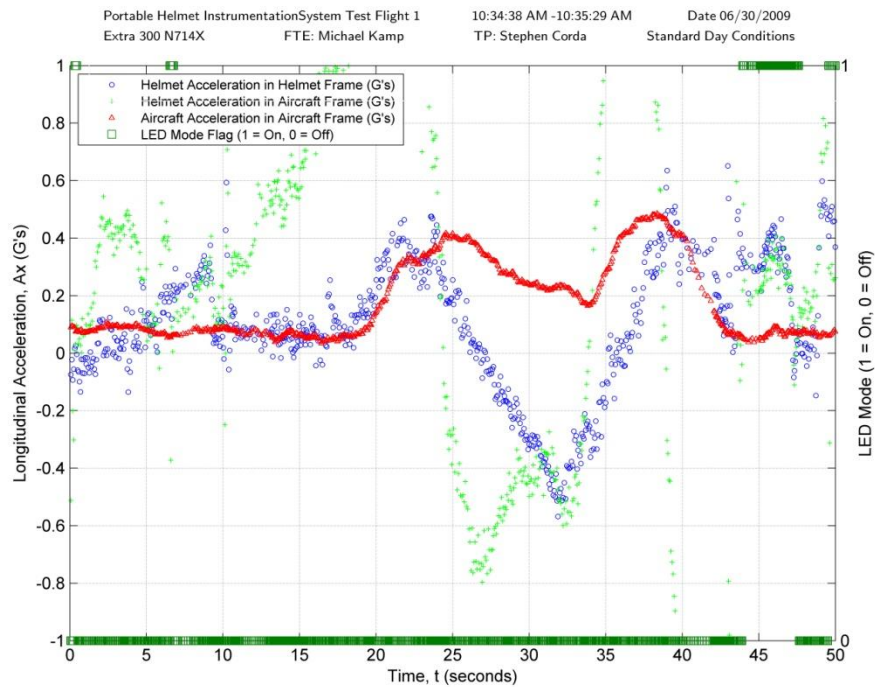


Figure 52: Longitudinal Acceleration versus Time for Barrel Roll Maneuver

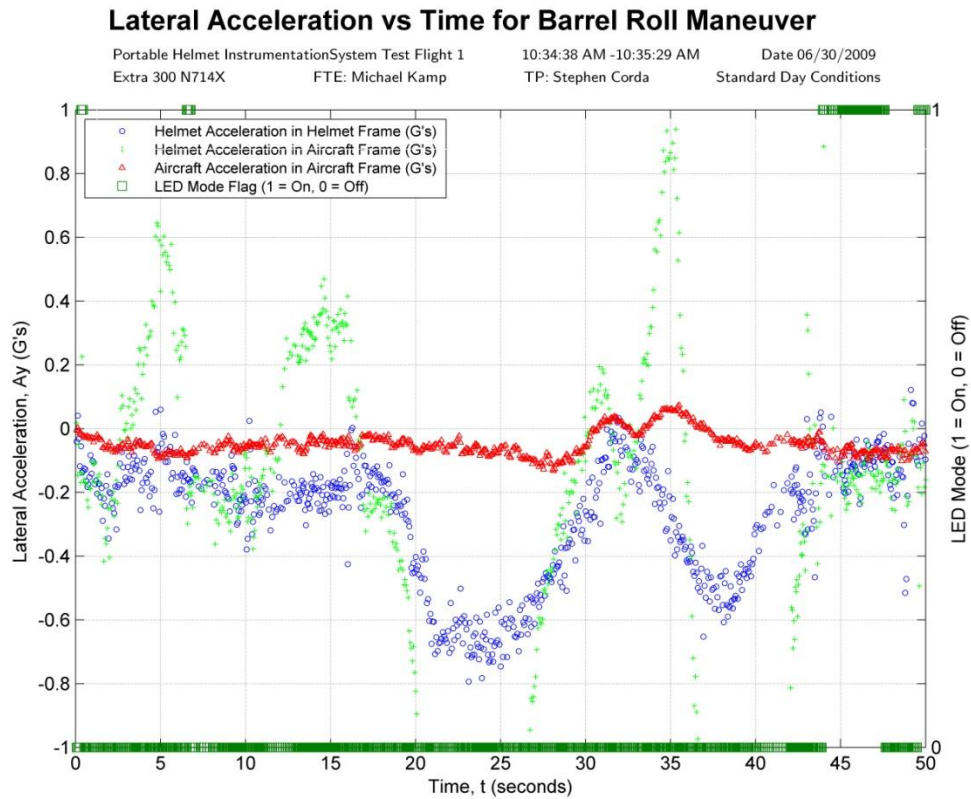


Figure 53: Lateral Acceleration versus Time for Barrel Roll Maneuver

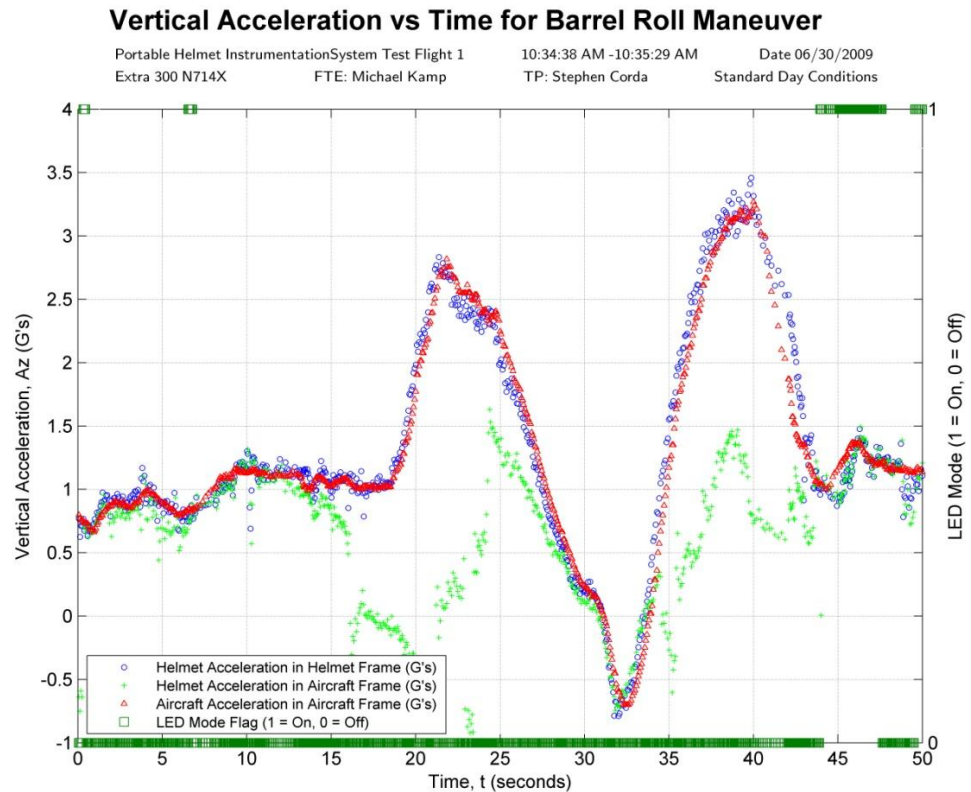


Figure 54: Vertical Acceleration versus Time for Barrel Roll Maneuver

Lateral Acceleration vs Time for Left Knife-Edge Maneuver

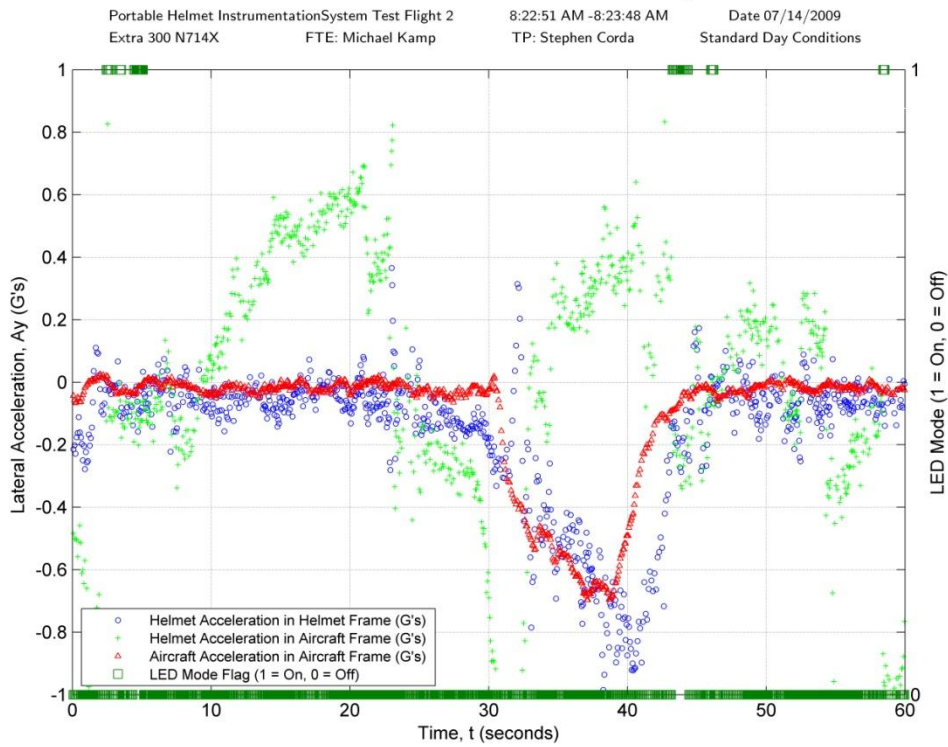


Figure 55: Lateral Acceleration versus Time for Left Knife-Edge Maneuver

Vertical Acceleration vs Time for Left Knife-Edge Maneuver

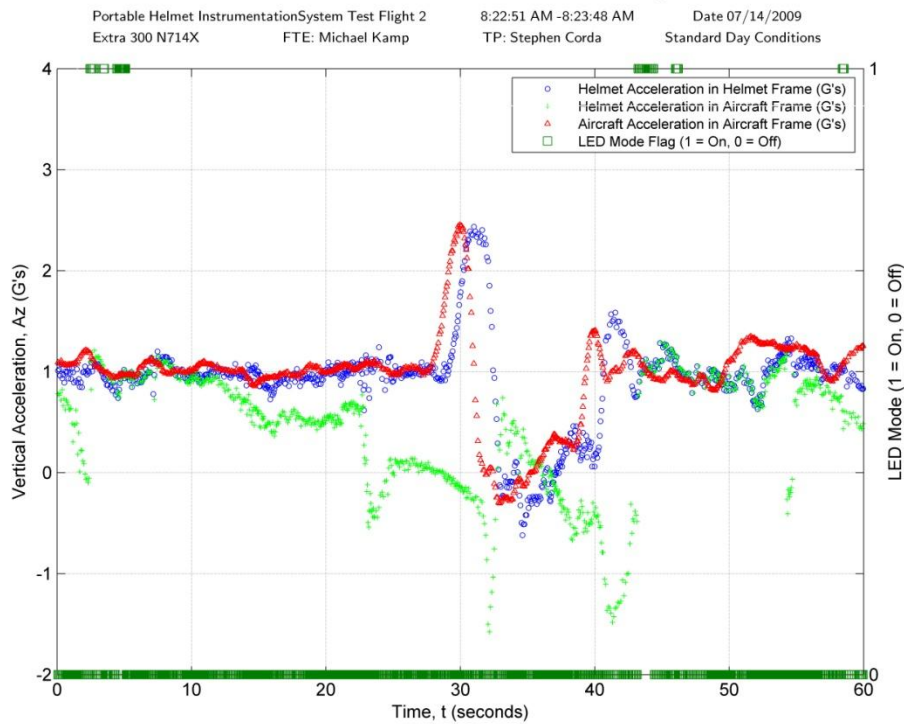


Figure 56: Vertical Acceleration versus Time for Left Knife-Edge Maneuver

Lateral Acceleration vs Time for Right Knife-Edge Maneuver

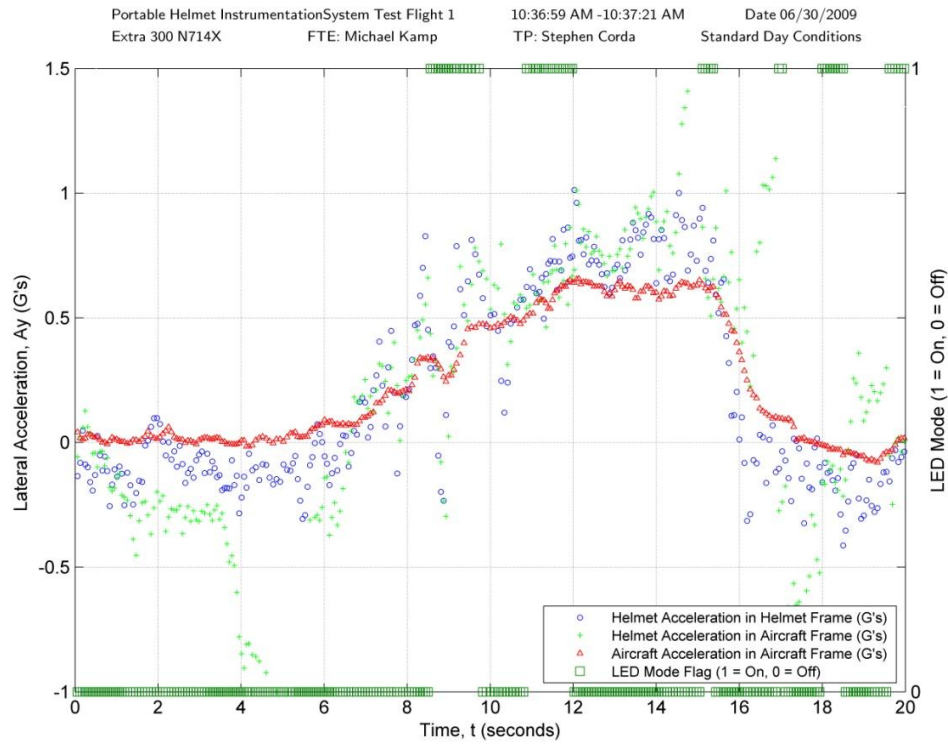


Figure 57: Lateral Acceleration versus Time for Right Knife-Edge Maneuver

Vertical Acceleration vs Time for Right Knife-Edge Maneuver

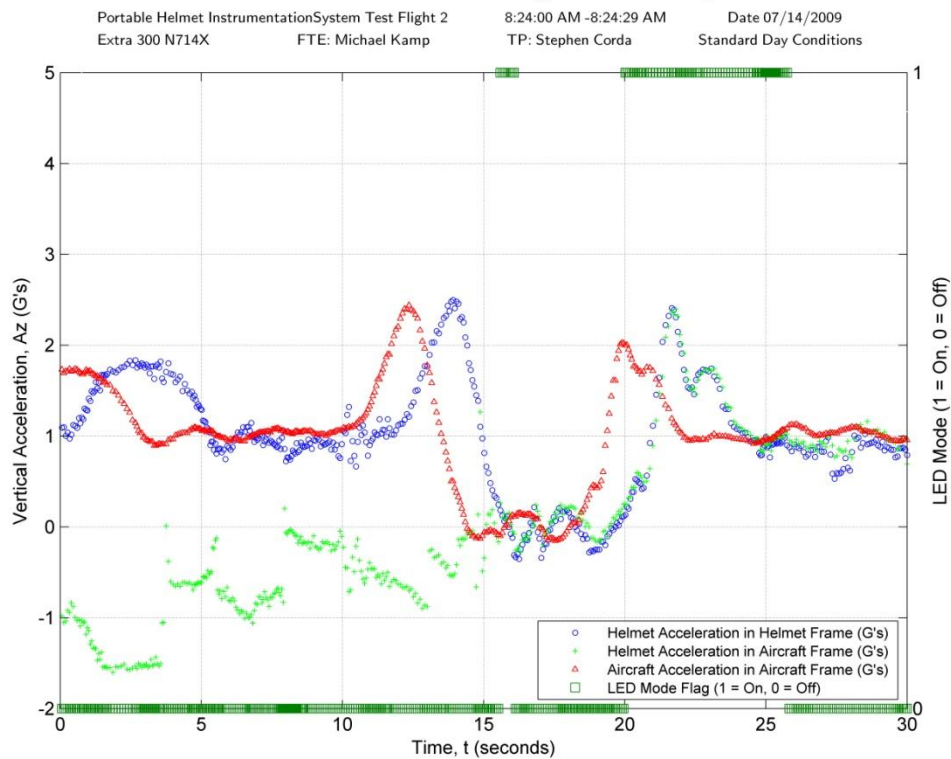


Figure 58: Vertical Acceleration versus Time for Right Knife-Edge Maneuver

Vertical Acceleration vs Time for Inverted Flight

Portable Helmet Instrumentation System Test Flight 1
Extra 300 N714X FTE: Michael Kamp TP: Stephen Corda Date 06/30/2009
Standard Day Conditions

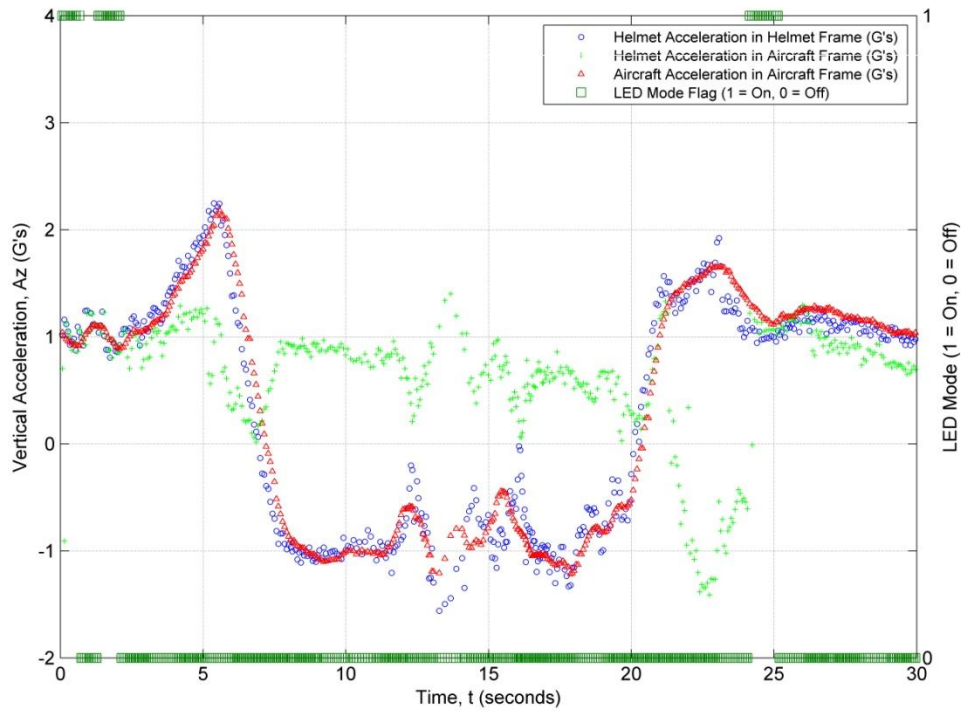


Figure 59: Vertical Acceleration versus Time for Inverted Flight

Vertical Acceleration vs Time for Bridge Maneuver

Portable Helmet Instrumentation System Test Flight 1
Extra 300 N714X FTE: Michael Kamp TP: Stephen Corda Date 06/30/2009
Standard Day Conditions

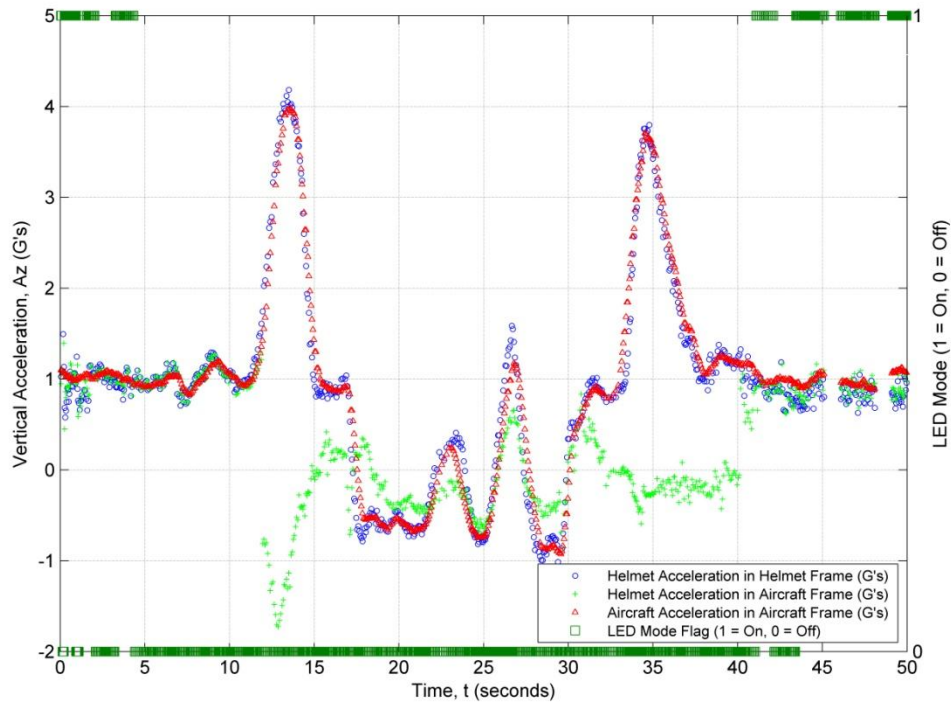


Figure 60: Vertical Acceleration versus Time for Bridge Maneuver

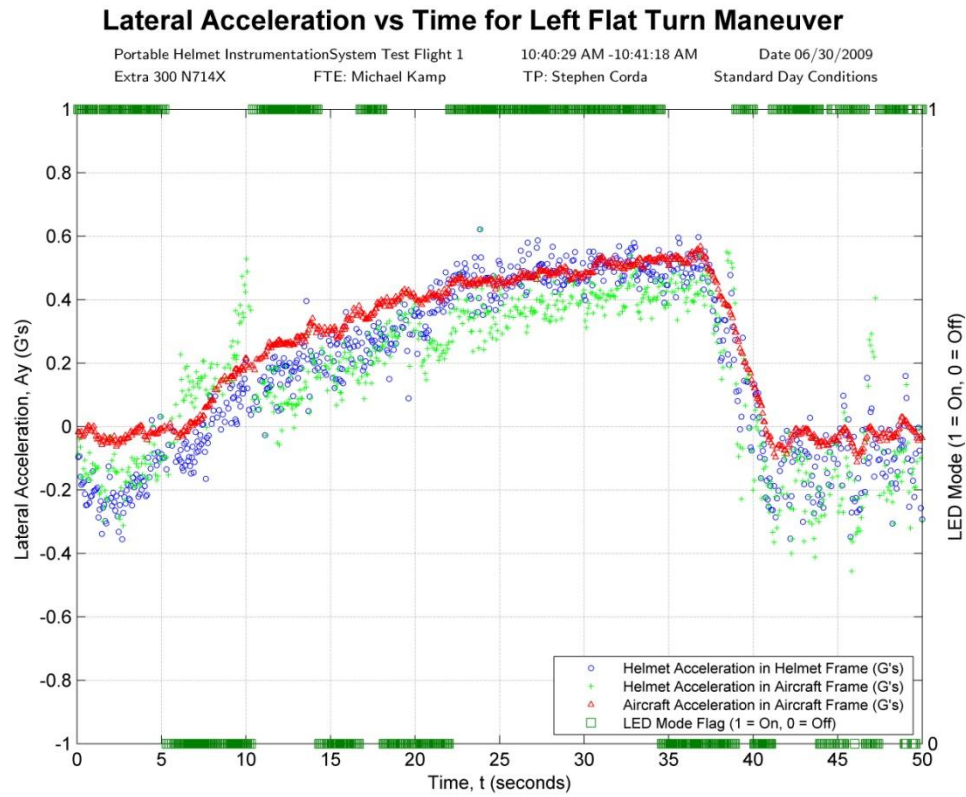


Figure 61: Lateral Acceleration versus Time for Left Flat Turn Maneuver

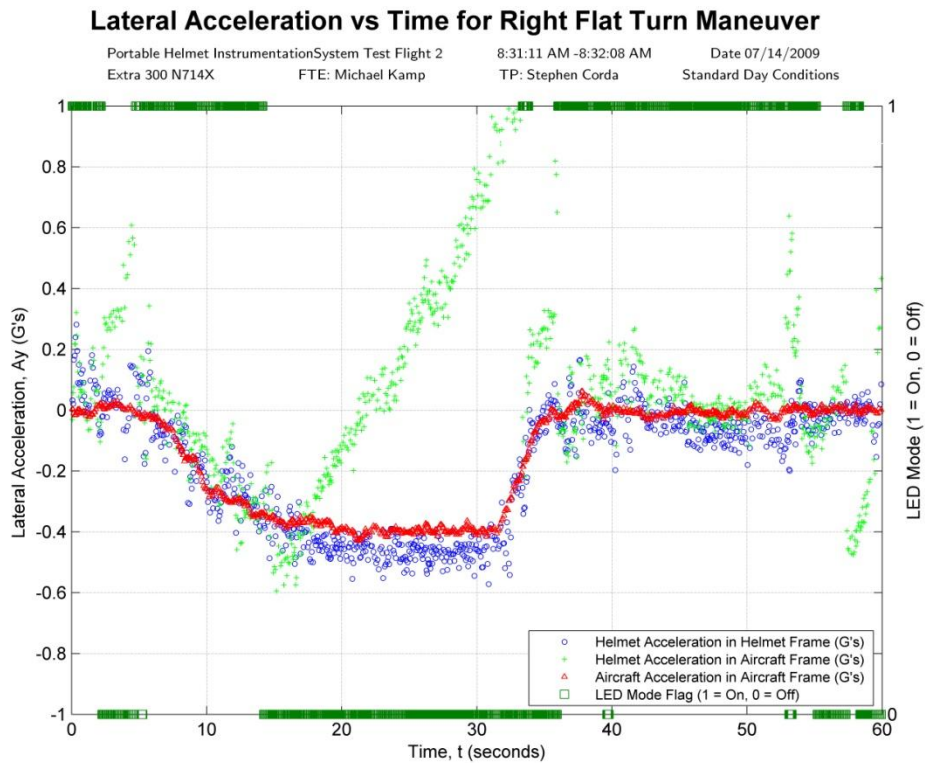


Figure 62: Lateral Acceleration versus Time for Right Flat Turn Maneuver

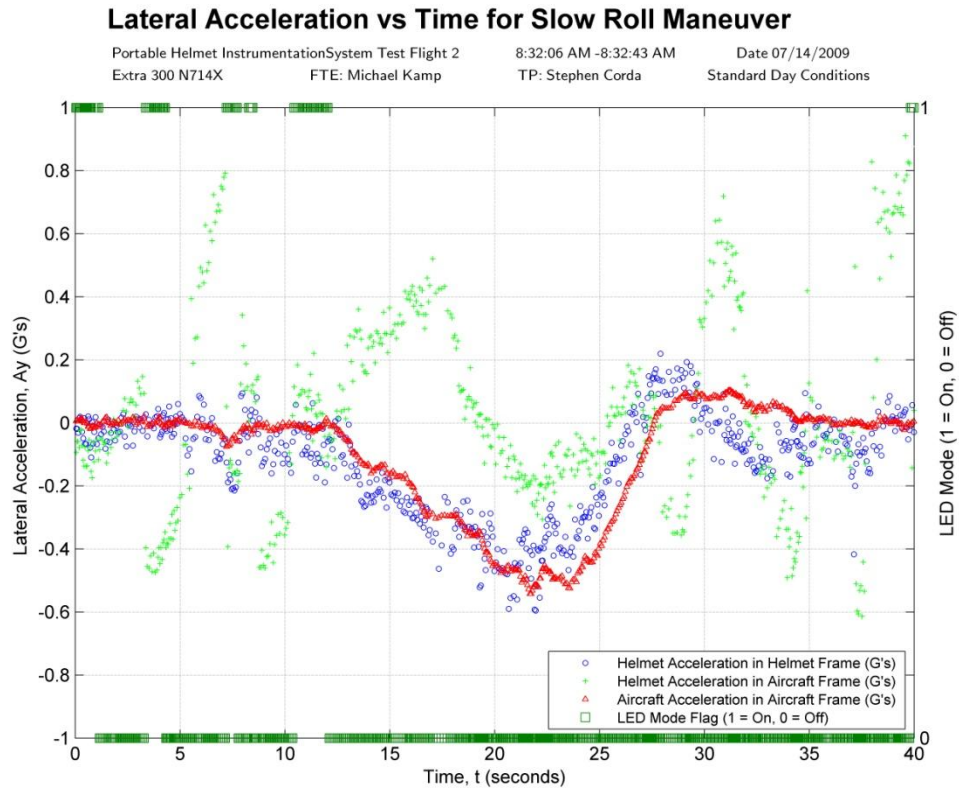


Figure 63: Lateral Acceleration versus Time for Slow Roll Maneuver

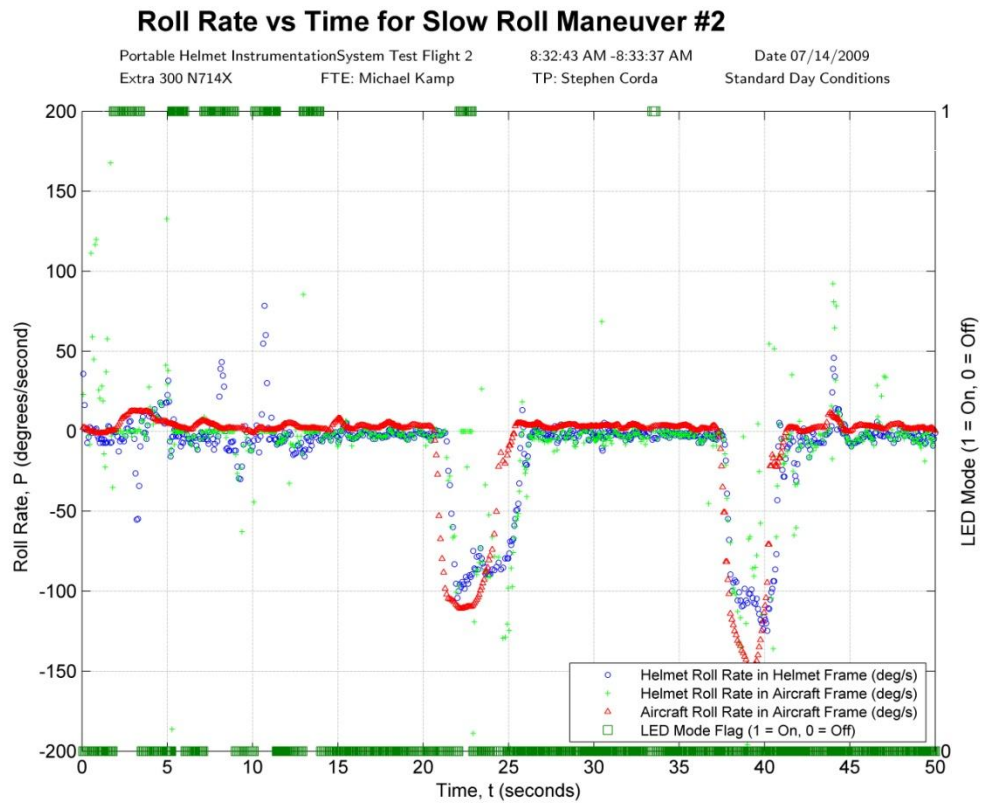


Figure 64: Roll Rate versus Time for Slow Roll Maneuver

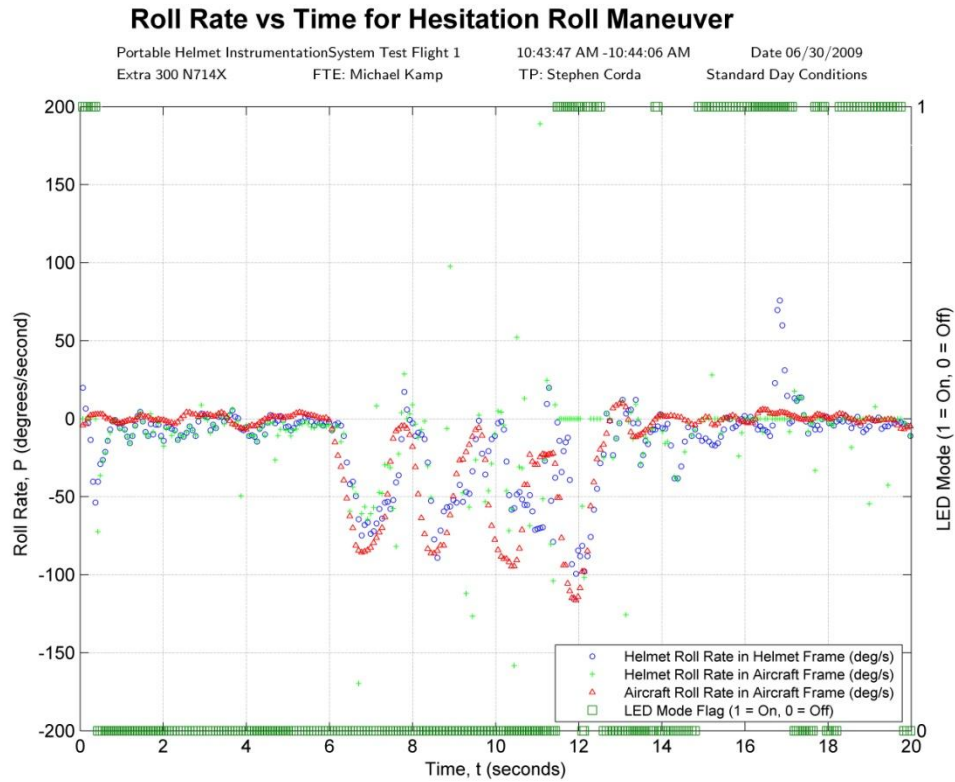


Figure 65: Roll Rate versus Time for Hesitation Roll Maneuver

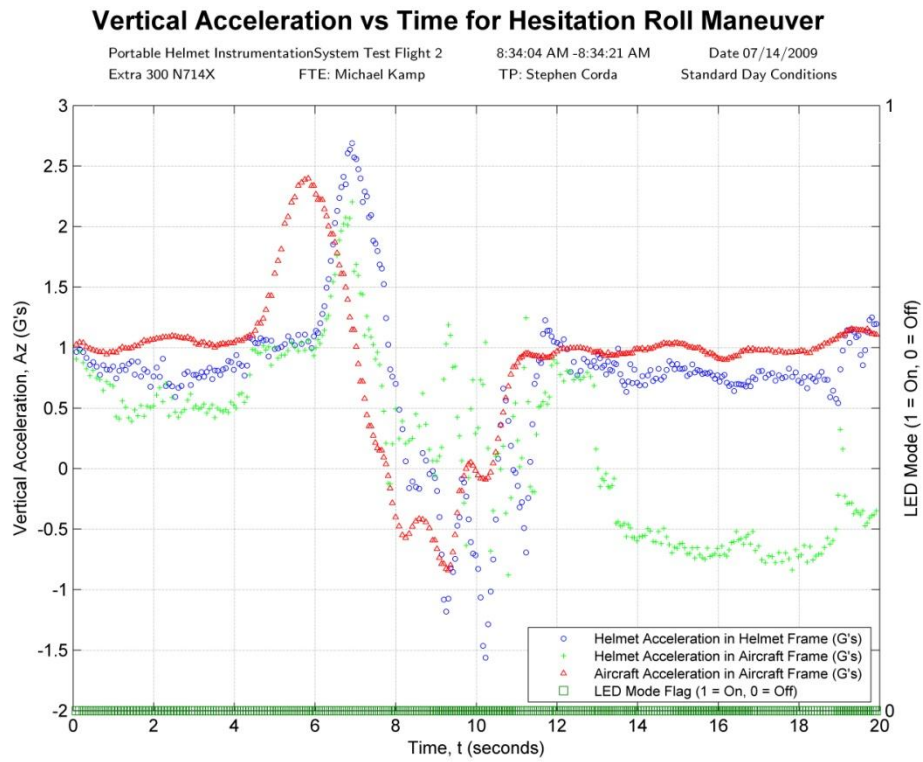


Figure 66: Acceleration versus Time for Hesitation Roll Maneuver

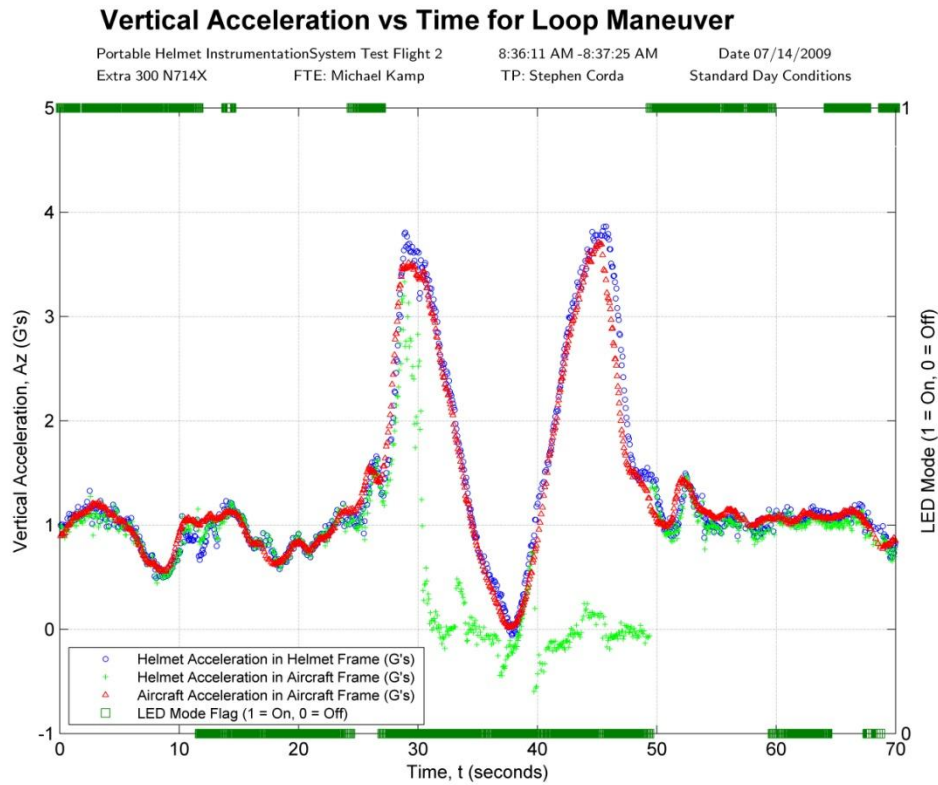


Figure 67: Vertical Acceleration versus Time for Loop Maneuver

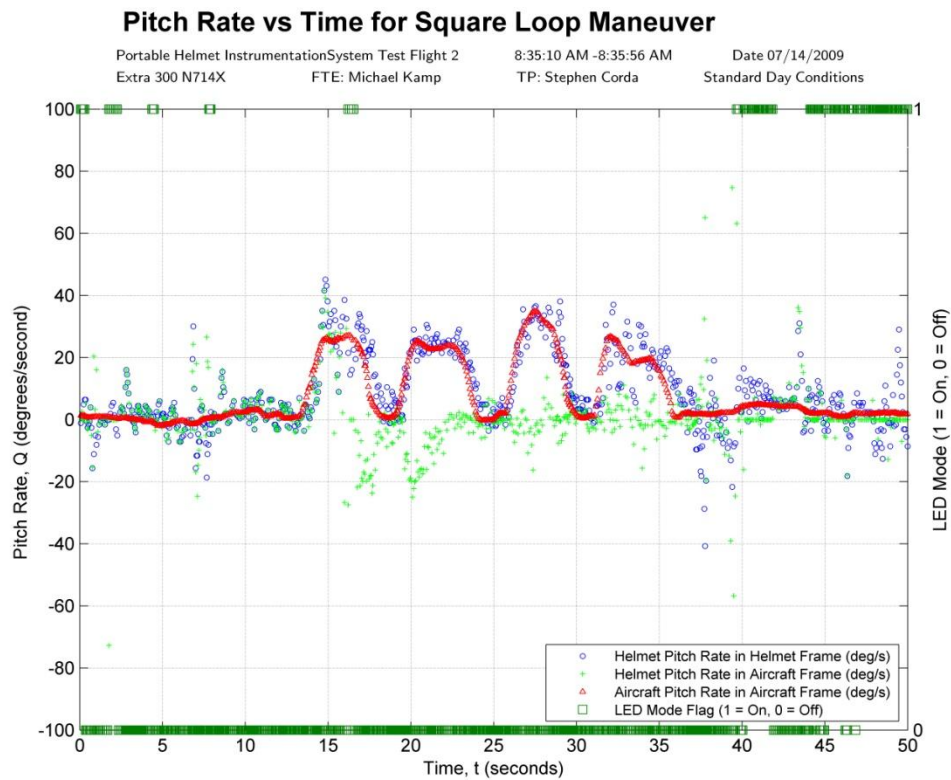


Figure 68: Pitch Rate versus Time for Square Loop Maneuver

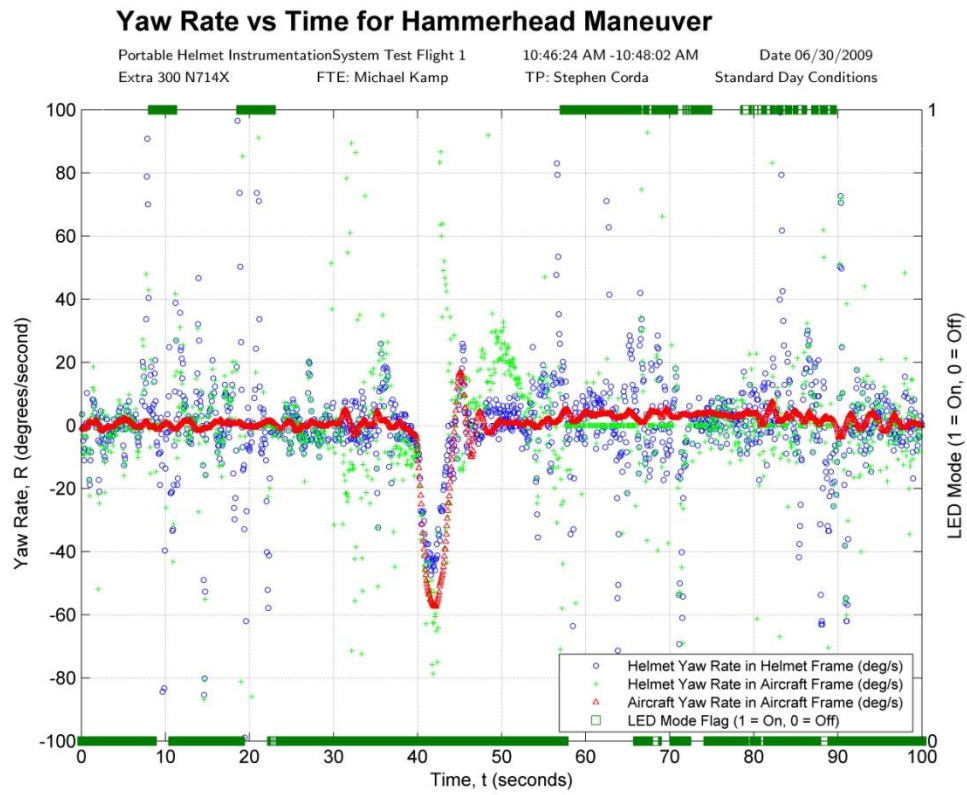


Figure 69: Yaw Rate versus Time for Hammerhead Maneuver

Appendix B

Table B - 1: List of Recorded Parameters

Helmet DAS		GRT Logger	
Time(sec)	1	datatype	53
Direct_Gyro_Wx_(deg/s)	2	Roll (deg)	54
Direct_Gyro_Wy_(deg/s)	3	Pitch (deg)	55
Direct_Gyro_Wz_(deg/s)	4	Yaw/HDG (deg, N=0)	56
Direct_Accel_X_(Gs)	5	Pressure Alt (ft)	57
Direct_Accel_Y_(Gs)	6	VSI (fpm)	58
Direct_Accel_Z_(Gs)	7	Airspeed (KIAS)	59
Raw_Gyro_Wy_(volts)	8	Delta V (kt/s^2)	60
Raw_Gyro_Wx_(volts)	9	Accel Roll (deg)	61
Raw_Gyro_Wz_(volts)	10	G	62
Raw_Accel_Y_(volts)	11	Raw Mag Heading	63
Raw_Accel_X_(volts)	12	X Accel (G)	64
Raw_Accel_Z_(volts)	13	Y Accel (G)	65
RawTemp(volts)	14	Z Accel (G)	66
RawCamX_slot1	15	X Gyro Rate	67
RawCamX_slot2	16	Y Gyro Rate	68
RawCamX_slot3	17	Z Gyro Rate	69
RawCamX_slot4	18	GPS Flag (1=good)	70
RawCamY_slot1	19	GPS UTC (hhmmss)	71
RawCamY_slot2	20	Latitude (dd.dd, +N/-S)	72
RawCamY_slot3	21	Longitude (dd.dd, +E/-W)	73
RawCamY_slot4	22	Ground Speed (kts)	74
Helmet_Ax(Gs)	23	True Course (deg)	75
Helmet_Ay(Gs)	24	UTC Date (dmmmyy)	76
Helmet_Az(Gs)	25	Mag Var (deg, +E/-W)	77
RollAngleinAC(rad)	26	Sats in Solution	78
PitchAngleinAC(rad)	27	HDOP	79
YawAngleinAC(rad)	28	GPS Altitude (ft)	80
XPosinAC(cm)	29	TACH (rpm)	81
YPosinAC(cm)	30	Voltage (V)	82
ZPosinAC(cm)	31	Fuel FLOW (gal/hr)	83
RollRateinAC(rad/s)	32	OAT (degF)	84
PitchRateinAC(rad/s)	33	OIL Temp (degF)	85
YawRateinAC(rad/s)	34	Oil Pressure (psi)	86
XVelinAC(cm/s)	35	EIS Aux 1	87
YVelinAC(cm/s)	36	EIS Aux 2	88
ZVelinAC(cm/s)	37	EIS Aux 3	89
XAccelinAC(Gs)	38	EIS Aux 4	90
YAccelinAC(Gs)	39	EIS Aux 5	91
ZAccelinAC(Gs)	40	EIS Aux 6	92
RollAngleInertial(rad)	41	Engine Time	93
PitchAngleInertial(rad)	42	Fuel Quantity (gal)	94
YawAngleInertial(rad)	43	Flight Timer (Hrs)	95
RollRateInertial(rad/s)	44	Flight Timer (Min)	96
PitchRateInertial(rad/s)	45	Flight Timer (Sec)	97
YawRateInertial(rad/s)	46	Fuel Reserve (Min)	98
XAccelInertial(Gs)	47	Baroset (Hg)	99
YAccelInertial(Gs)	48	Event Counter	100
ZAccelInertial(Gs)	49		
LEDModeFlag	50		
PositionSet	51		
LOGGER_Time	52		

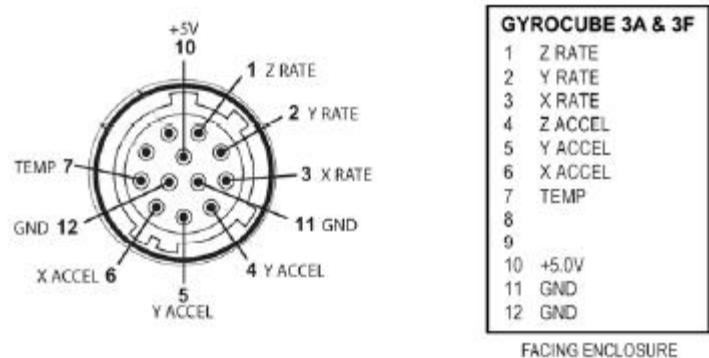


Figure B - 1: O-Navi Gyrocube 3F Pin-out Diagram

Six Position Method For Accelerometer and Gyroscope Calibration	
Scale Factor Accel X-Axis	Bias Accel X-Axis (m/s ²)
0	0
Scale Factor Accel Y-Axis	Bias Accel Y-Axis (m/s ²)
0	0
Scale Factor Accel Z-Axis	Bias Accel Z-Axis (m/s ²)
0	0
Rotational Speed	
0	
Scale Factor Rate X-Axis	Bias Rate X-Axis (deg/s)
0	0
Scale Factor Rate Y-Axis	Bias Rate Y-Axis (deg/s)
0	0
Scale Factor Rate Z-Axis	Bias Rate Z-Axis (deg/s)
0	0

Figure B - 2: Six Position Method for Accelerometer and Gyroscope Calibration Labview Front Panel

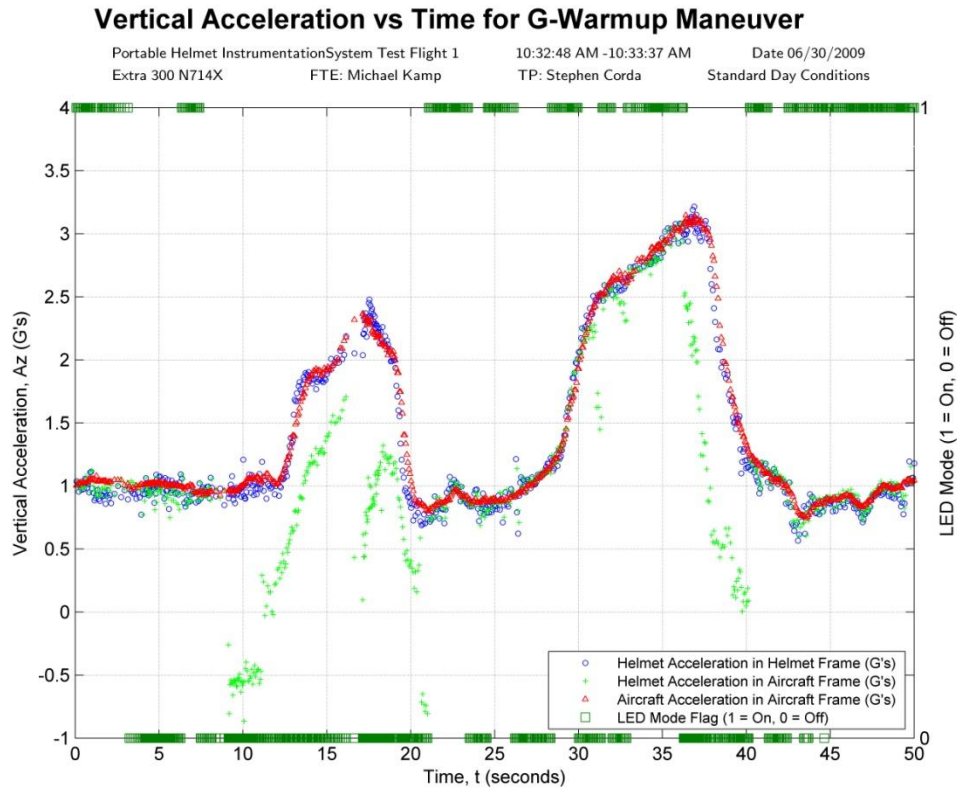


Figure B - 3: Vertical Acceleration versus Time for G-Warmup Maneuver

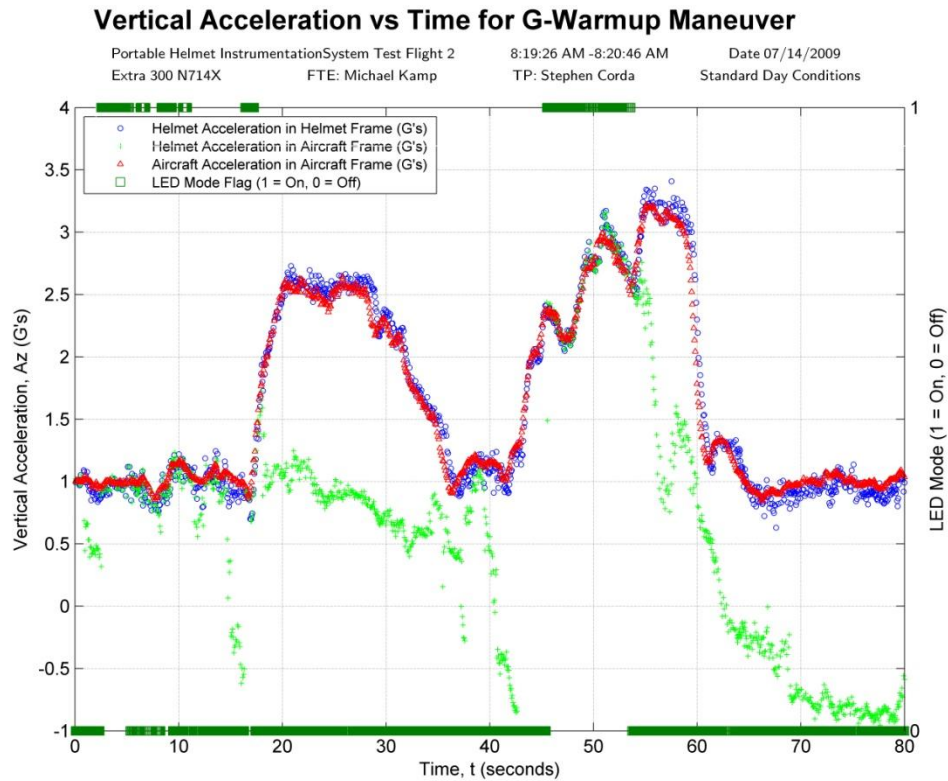


Figure B - 4: Vertical Acceleration versus Time for G-Warmup Maneuver

Longitudinal Acceleration vs Time for G-Warmup Maneuver

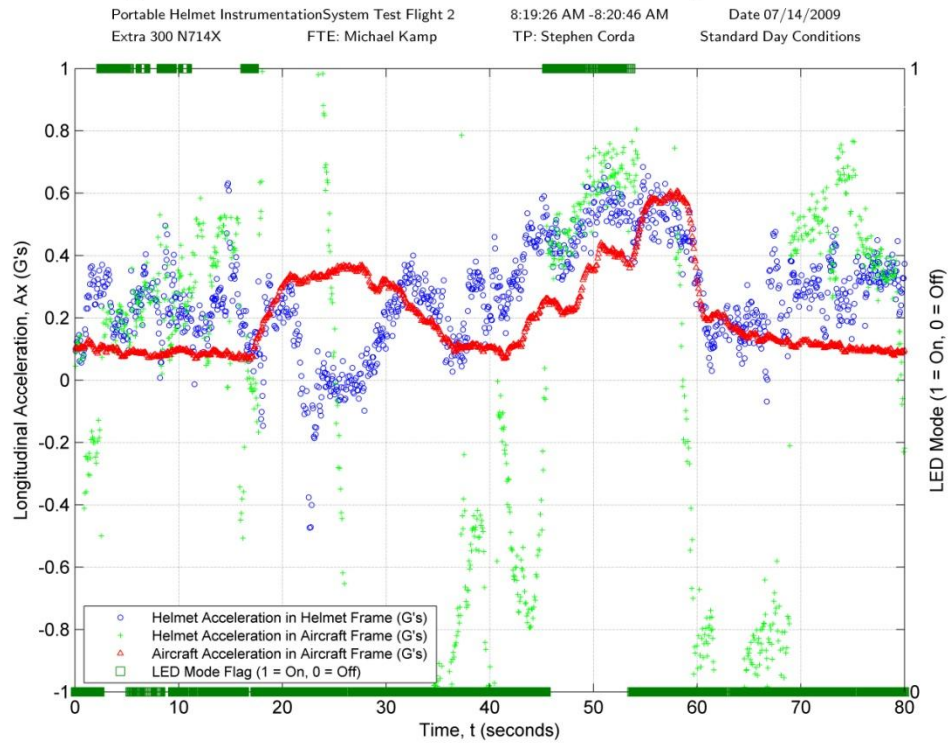


Figure B - 5: Longitudinal Acceleration versus Time for G-Warmup Maneuver

Longitudinal Acceleration vs Time for Barrel Roll Maneuver

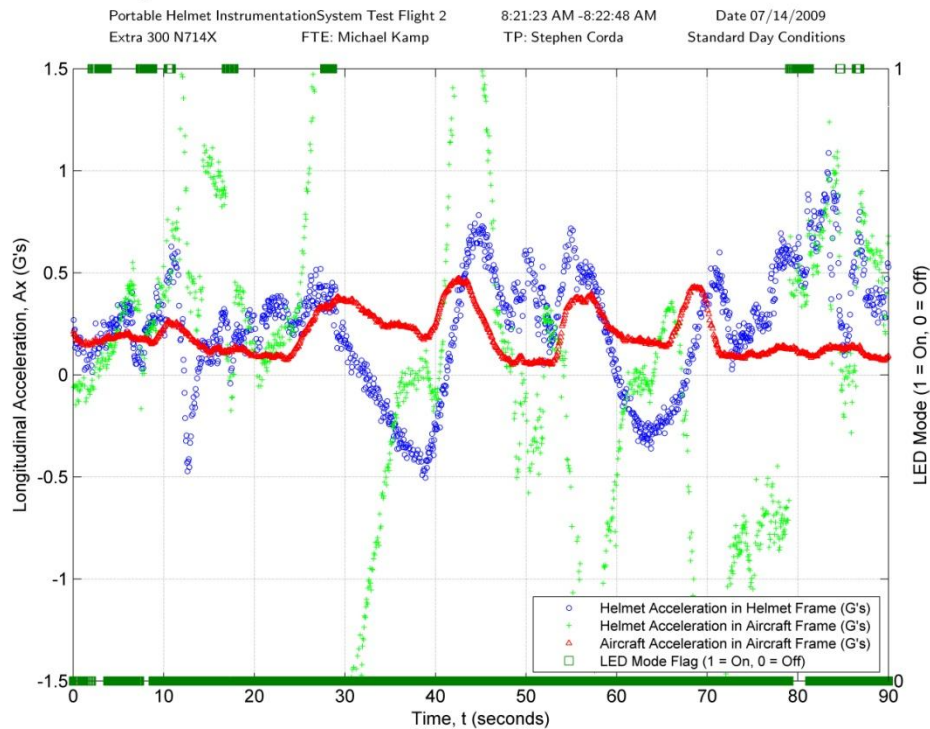


Figure B - 6: Longitudinal Acceleration versus Time for Barrel Roll Maneuver

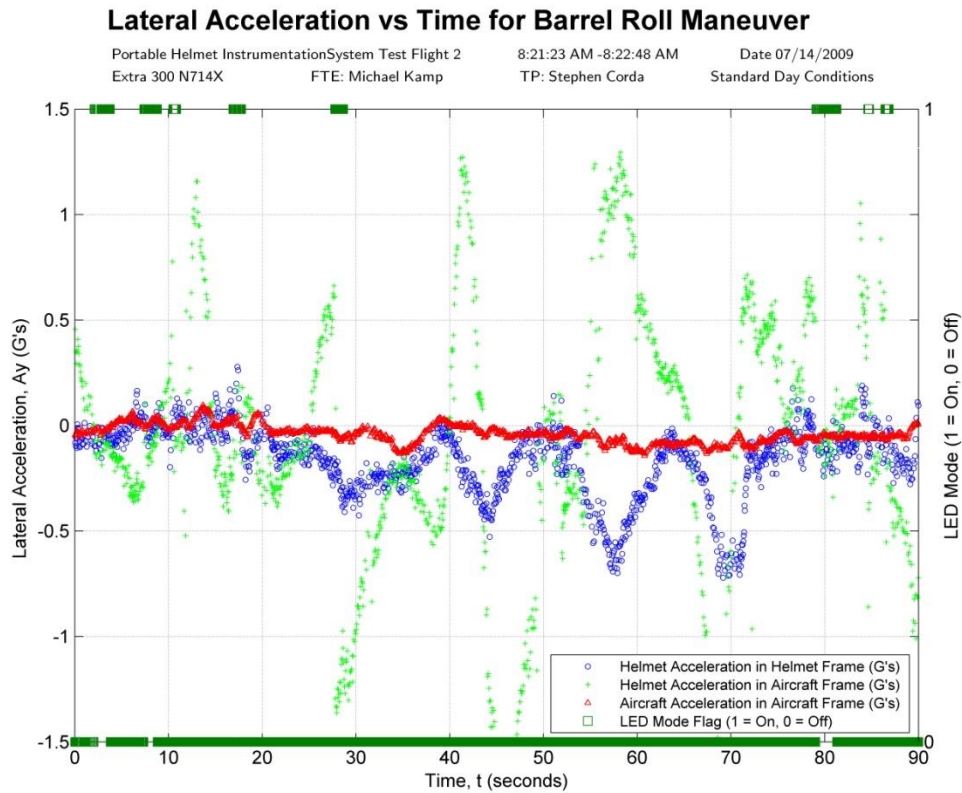


Figure B - 7: Lateral Acceleration versus Time for Barrel Roll Maneuver

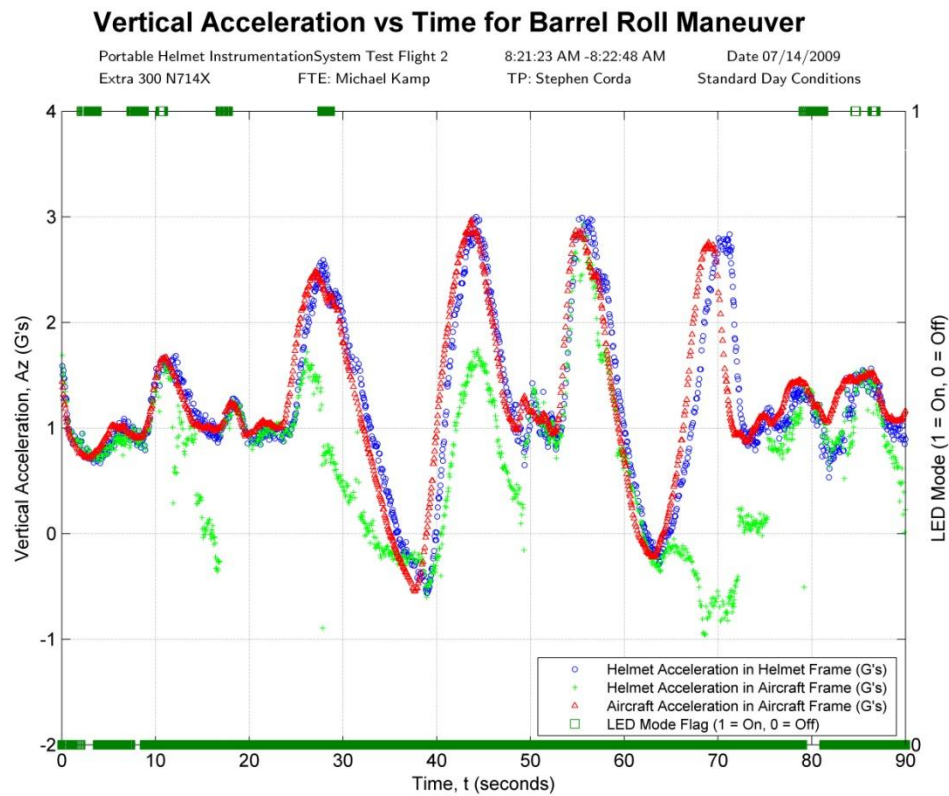


Figure B - 8: Vertical Acceleration versus Time for Barrel Roll Maneuver

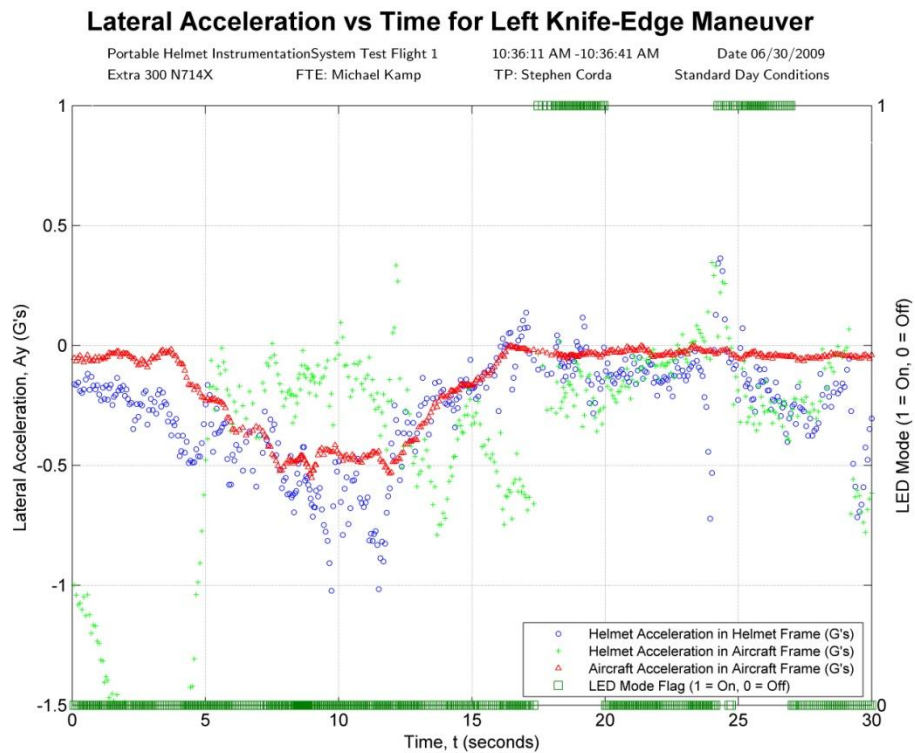


Figure B - 9: Lateral Acceleration versus Time for Left Knife-Edge Maneuver

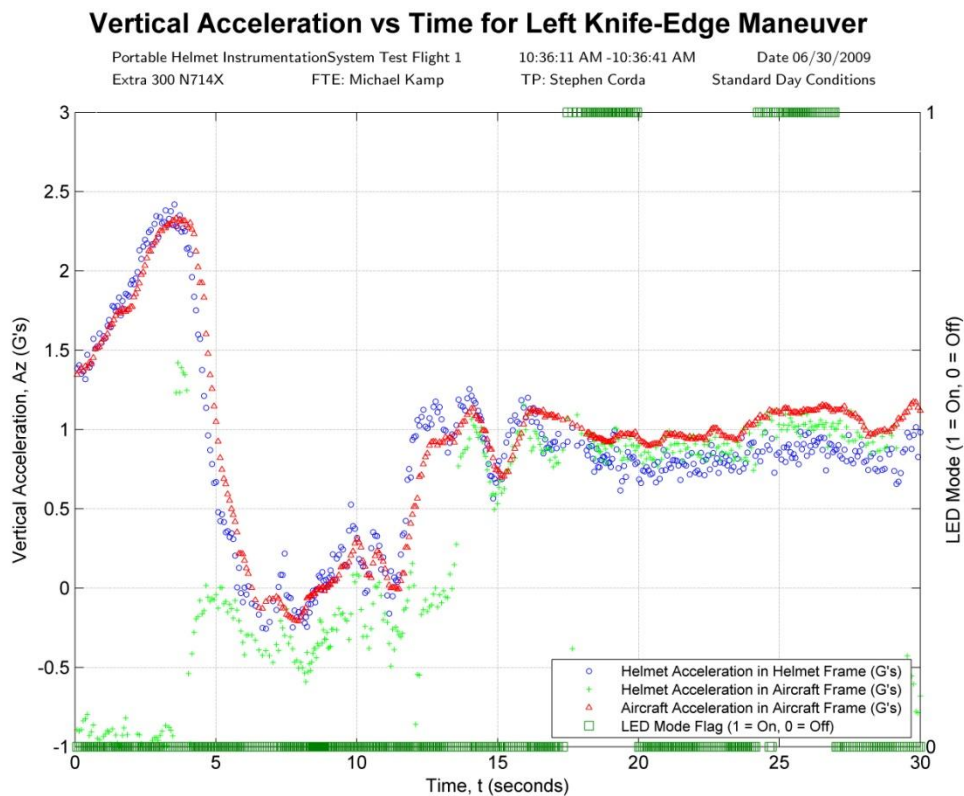


Figure B - 10: Vertical Acceleration versus Time for Left Knife-Edge Maneuver

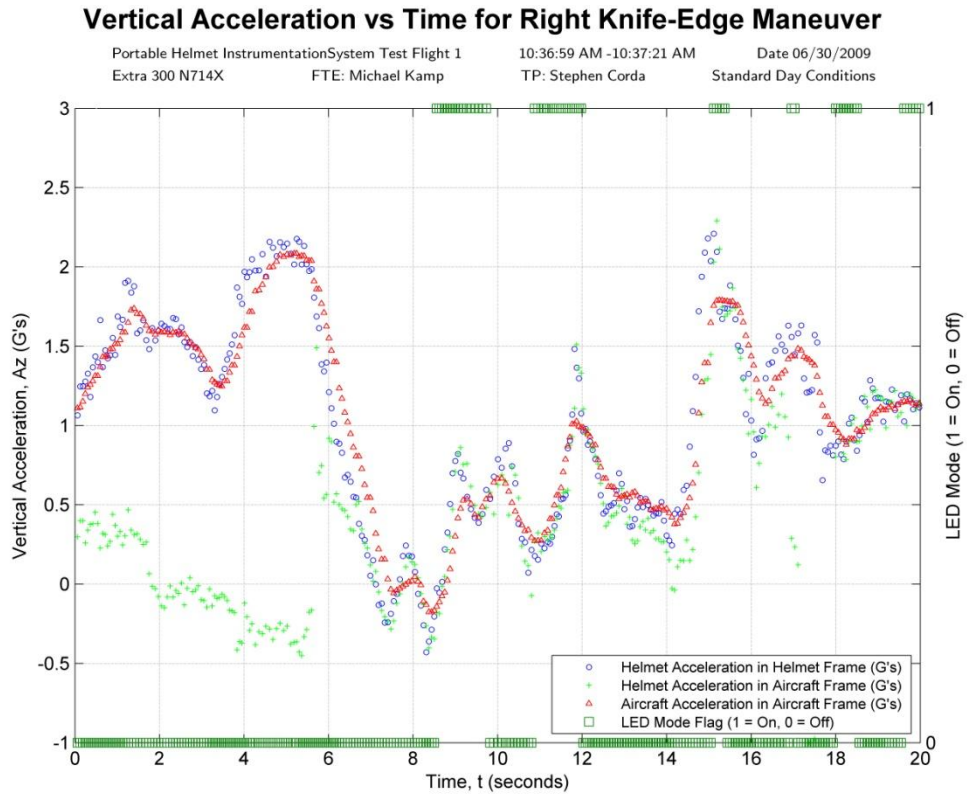


Figure B - 11: Vertical Acceleration versus Time for Right Knife-Edge Maneuver

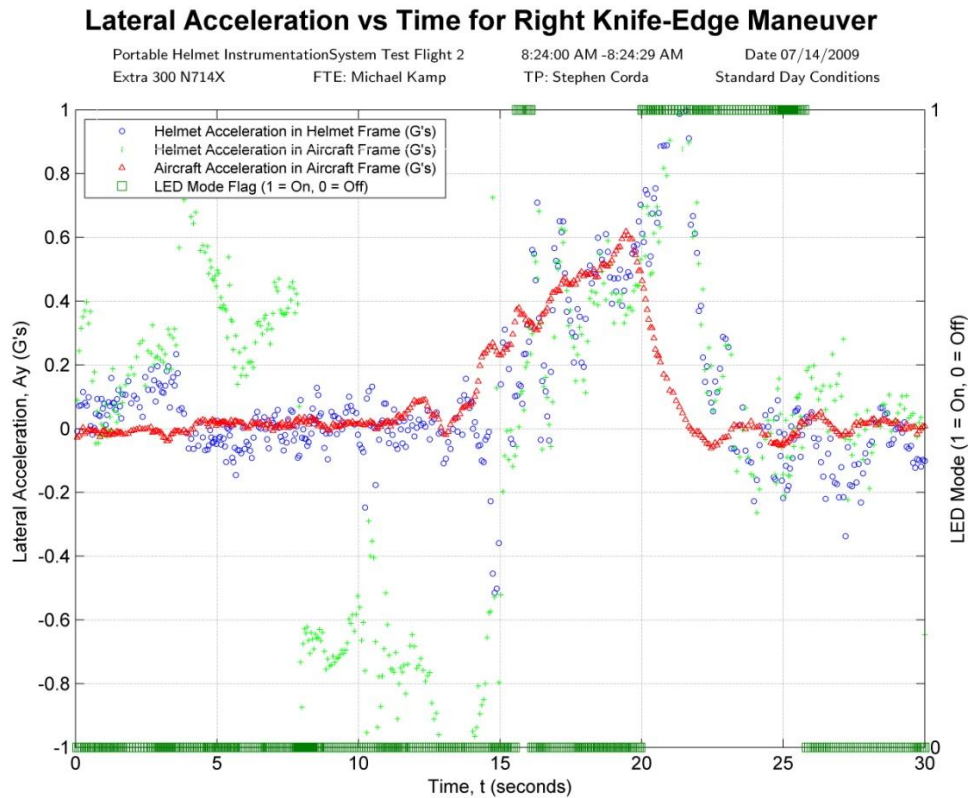


Figure B - 12: Lateral Acceleration versus Time for Right Knife-Edge Maneuver

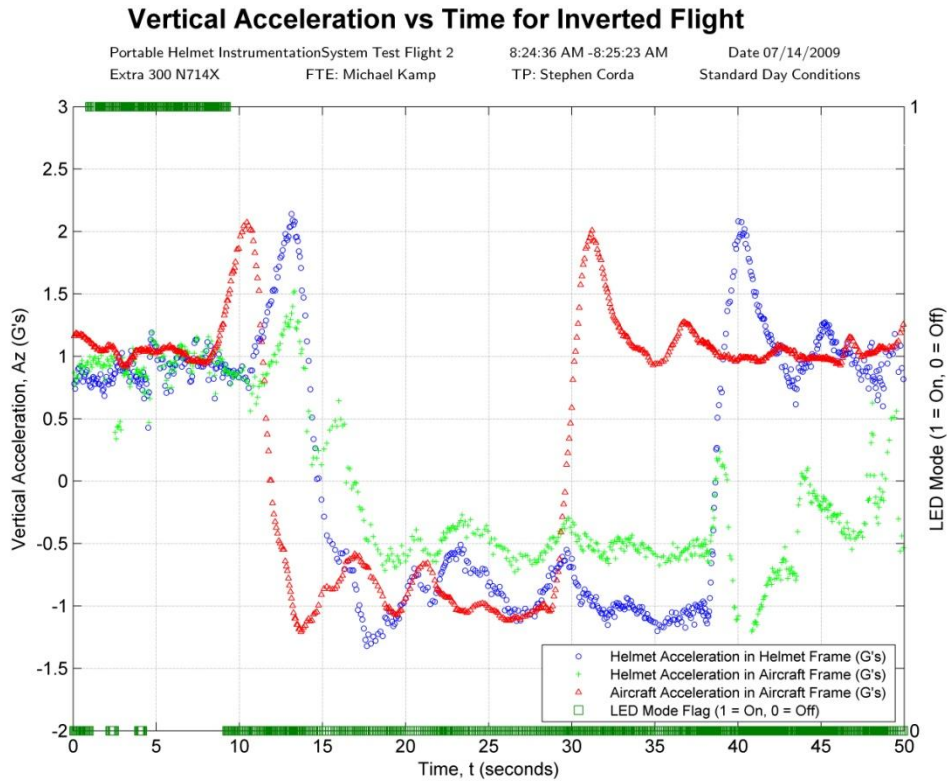


Figure B - 13: Vertical Acceleration versus Time for Inverted Flight

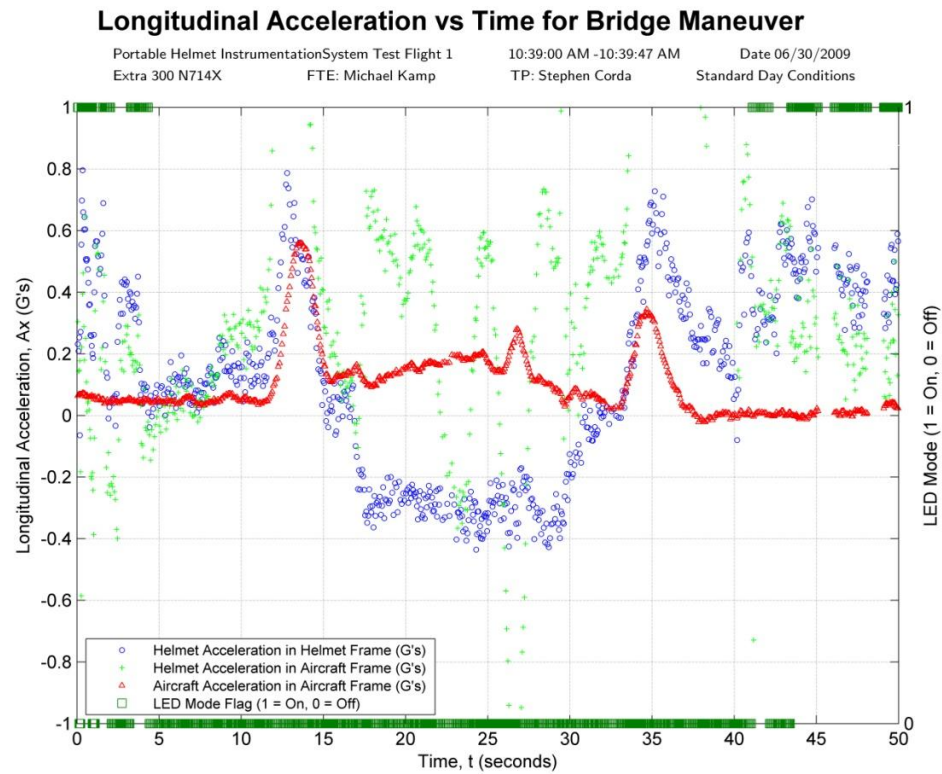


Figure B - 14: Longitudinal Acceleration versus Time for Bridge Maneuver

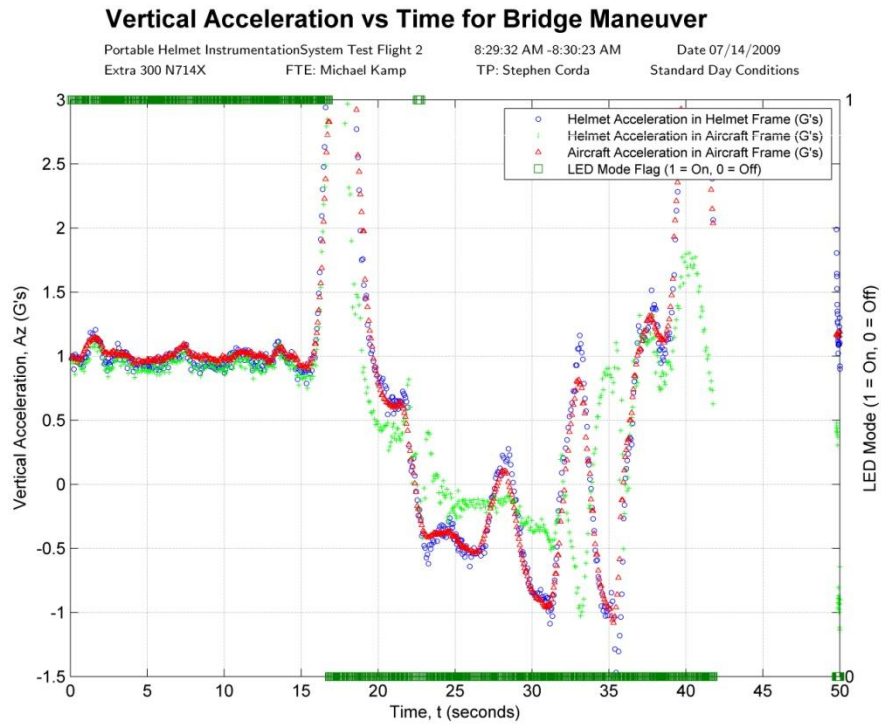


Figure B - 15: Vertical Acceleration versus Time for Bridge Maneuver

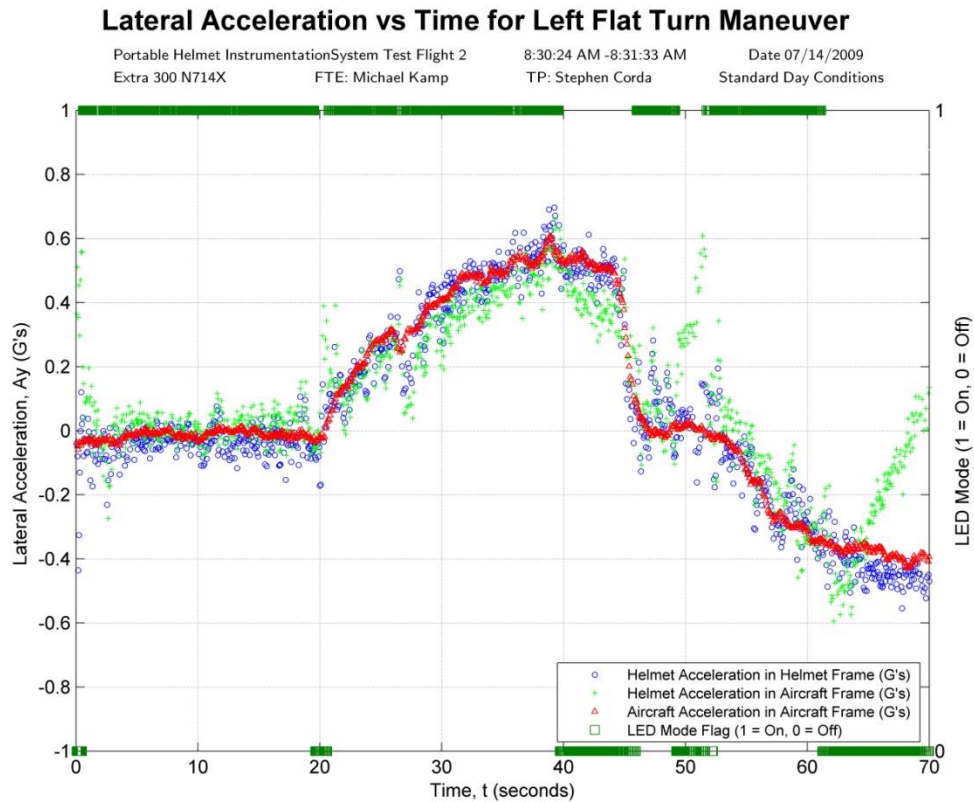


Figure B - 16: Lateral Acceleration versus Time for Left Flat Turn Maneuver

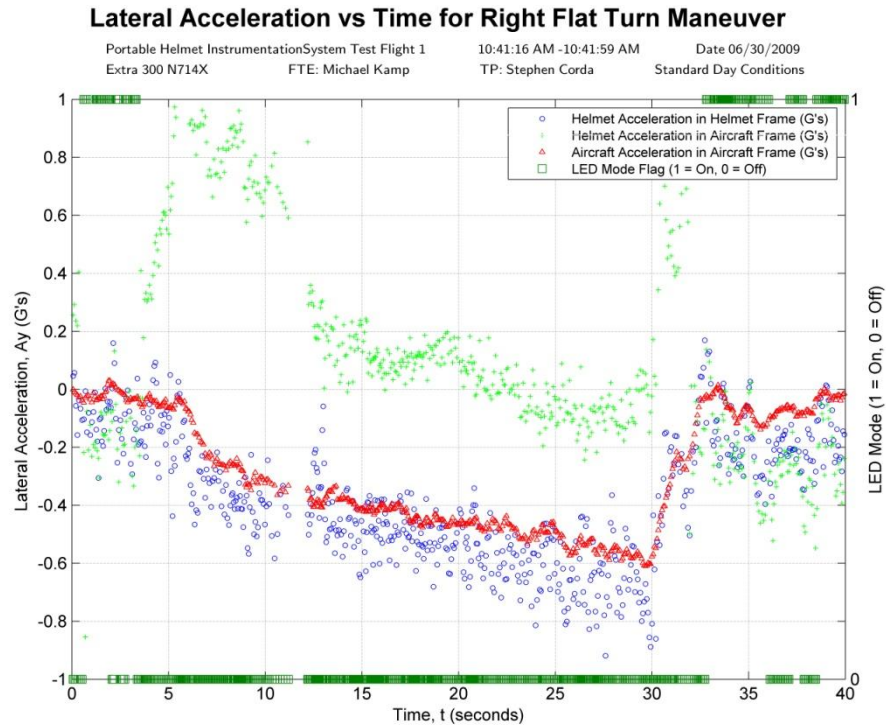


Figure B - 17: Lateral Acceleration versus Time for Right Flat Turn Maneuver

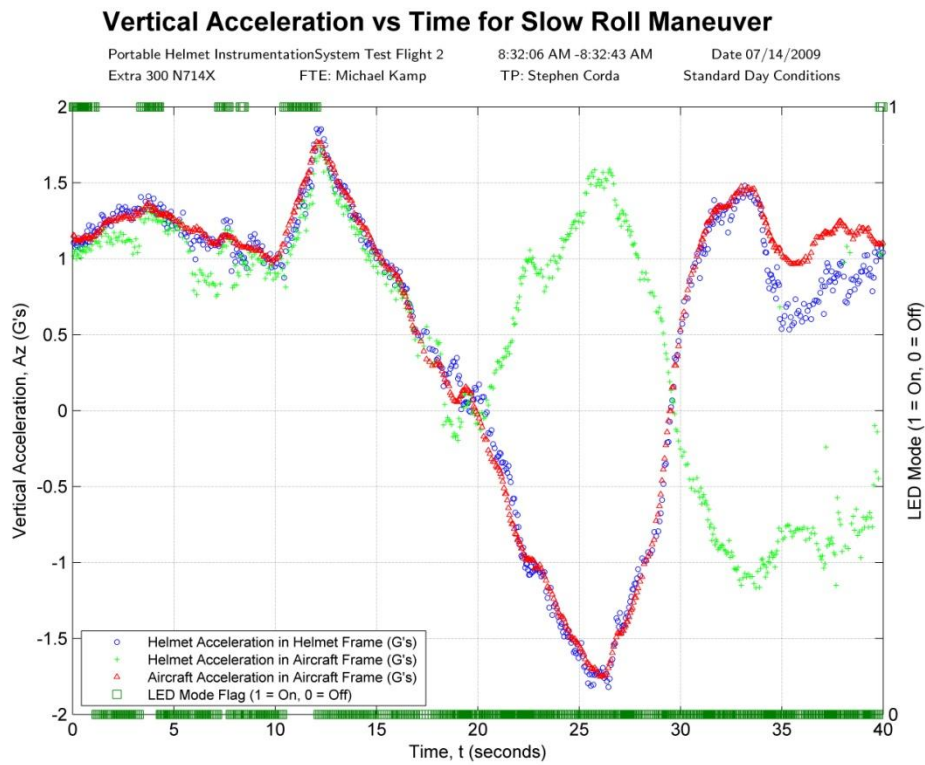


Figure B - 18: Vertical Acceleration versus Time for Slow Roll Maneuver

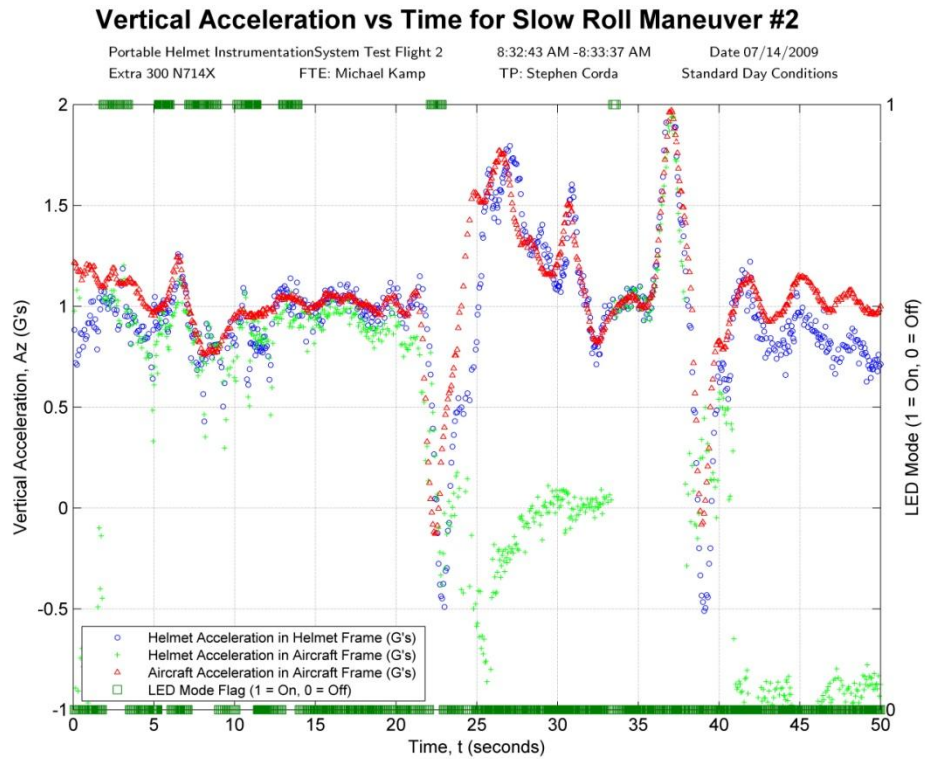


Figure B - 19: Vertical Acceleration versus Time for Slow Roll Maneuver

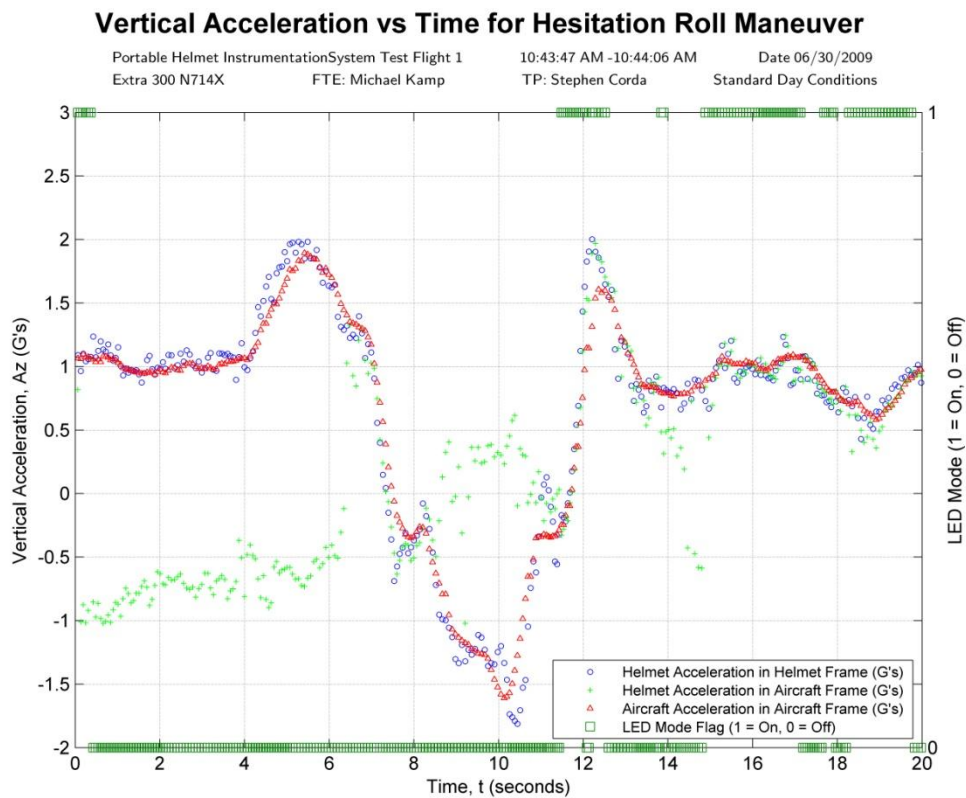


Figure B - 20: Vertical Acceleration versus Time for Hesitation Roll Maneuver

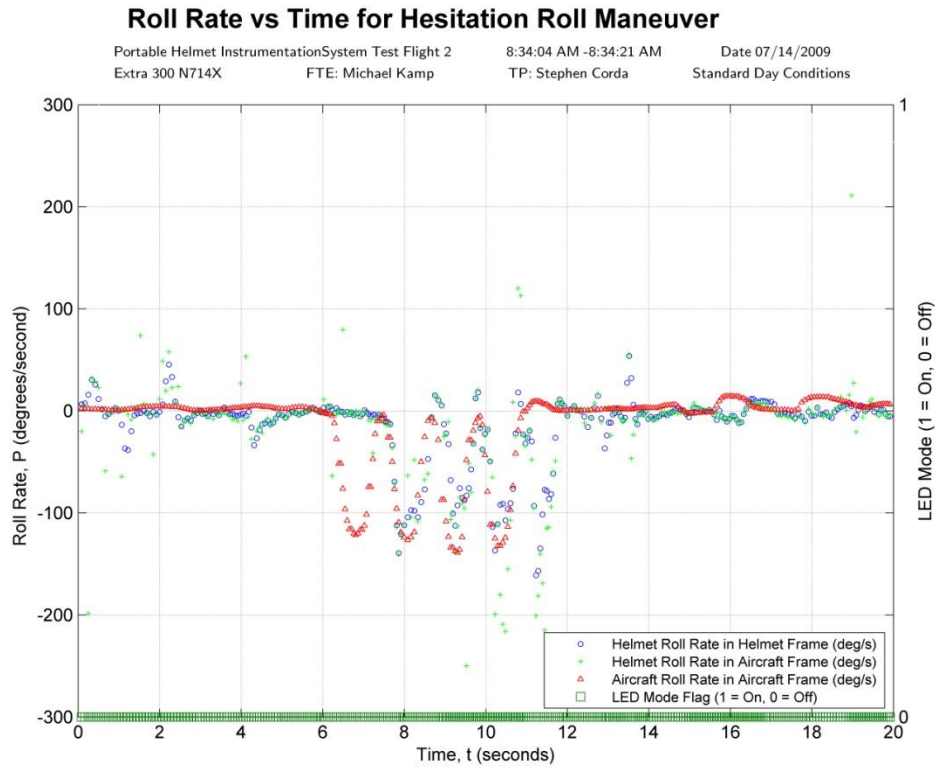


Figure B - 21: Roll Rate versus Time for Hesitation Roll Maneuver

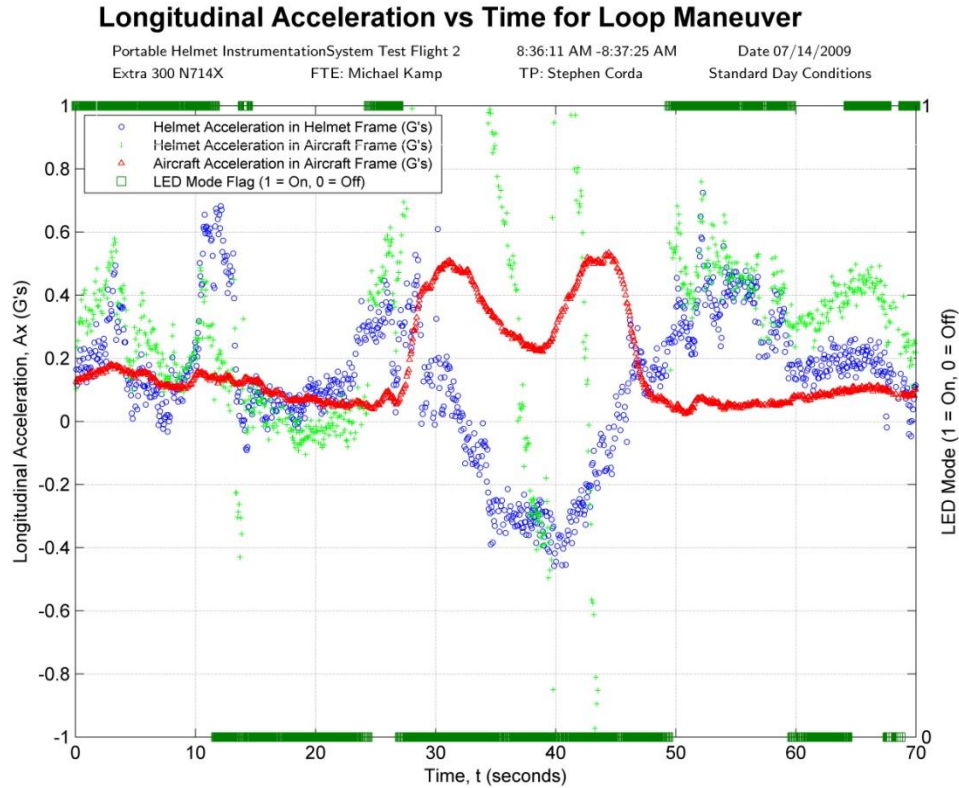


Figure B - 22: Longitudinal Acceleration versus Time for Loop Maneuver

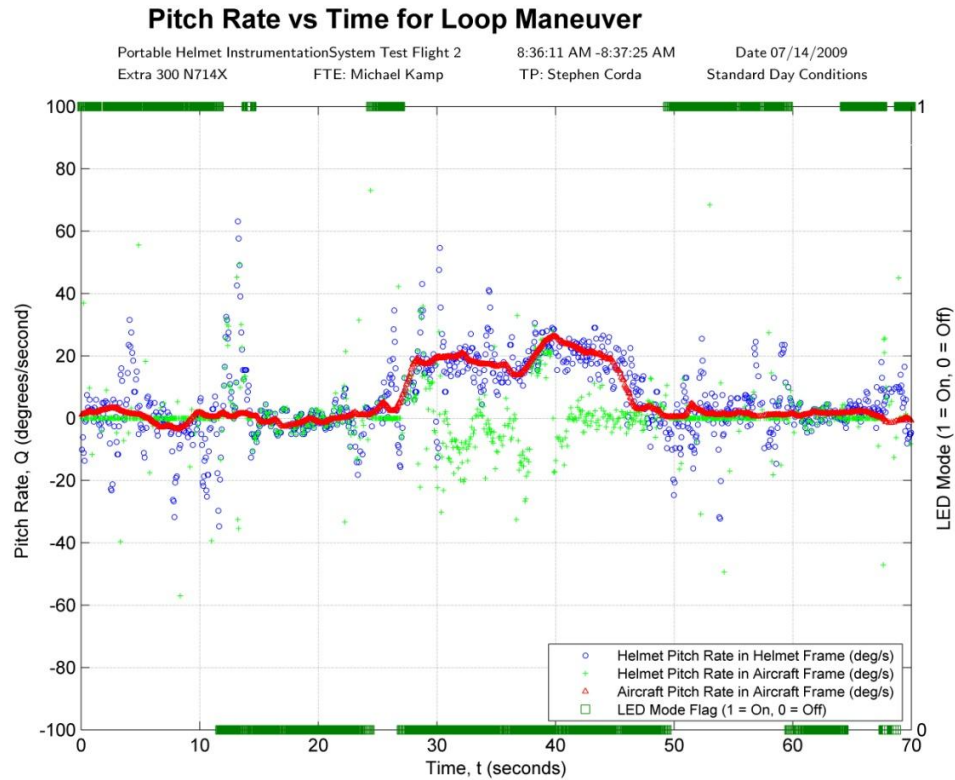


Figure B - 23: Pitch Rate versus Time for Loop Maneuver

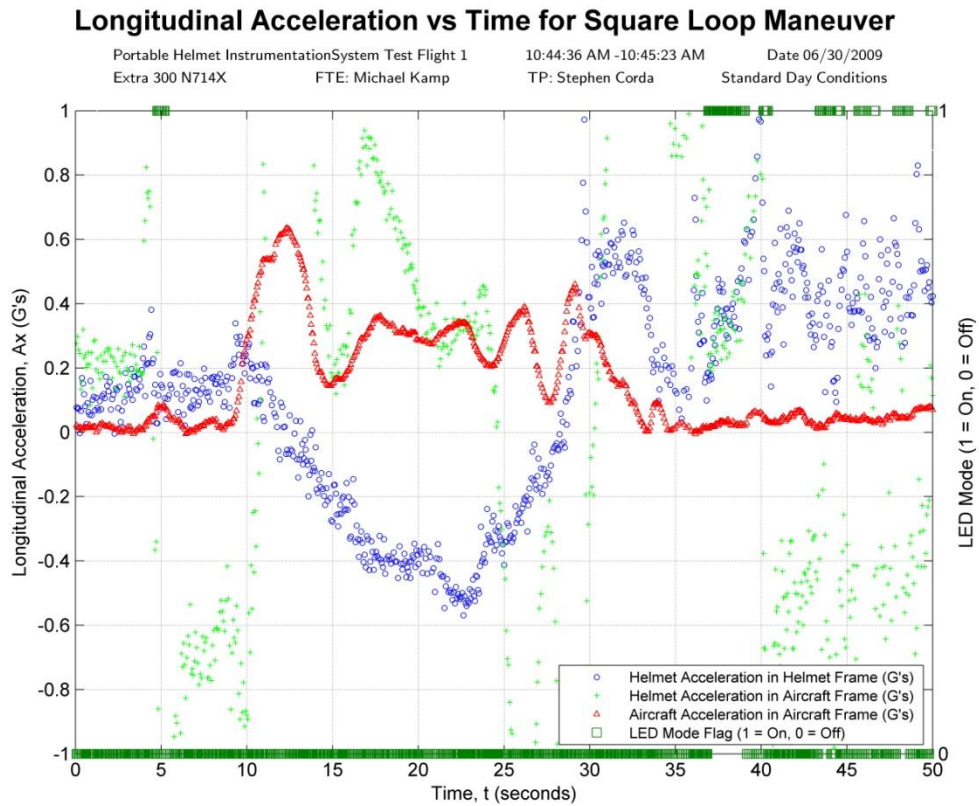


Figure B - 24: Longitudinal Acceleration versus Time for Square Loop Maneuver

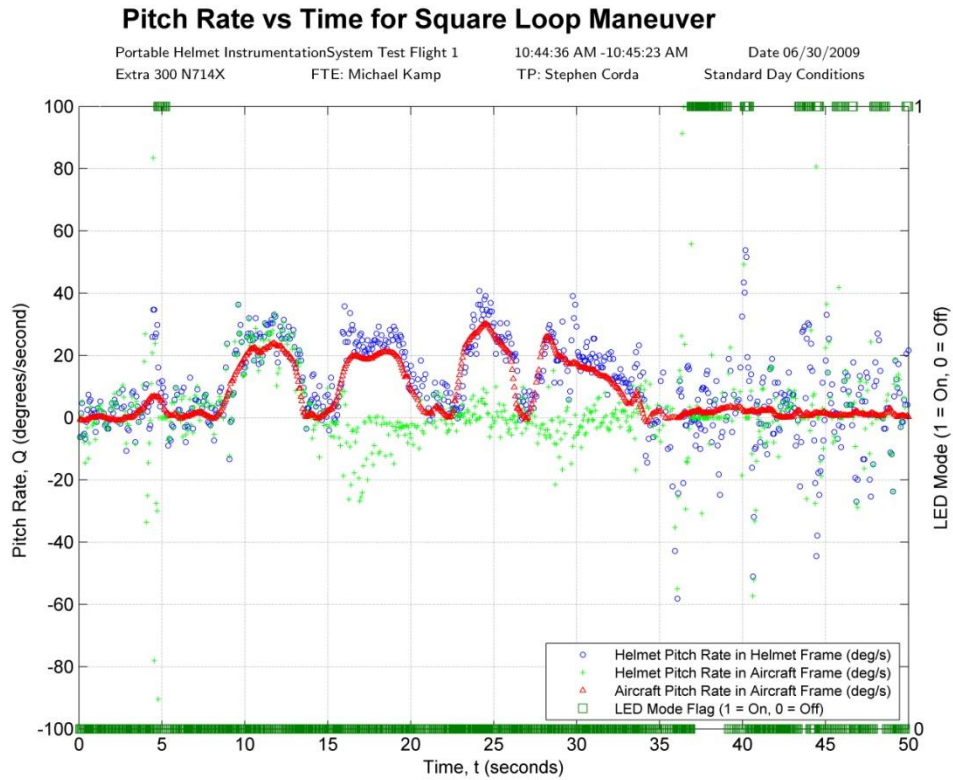


Figure B - 25: Pitch Rate versus Time for Square Loop Maneuver

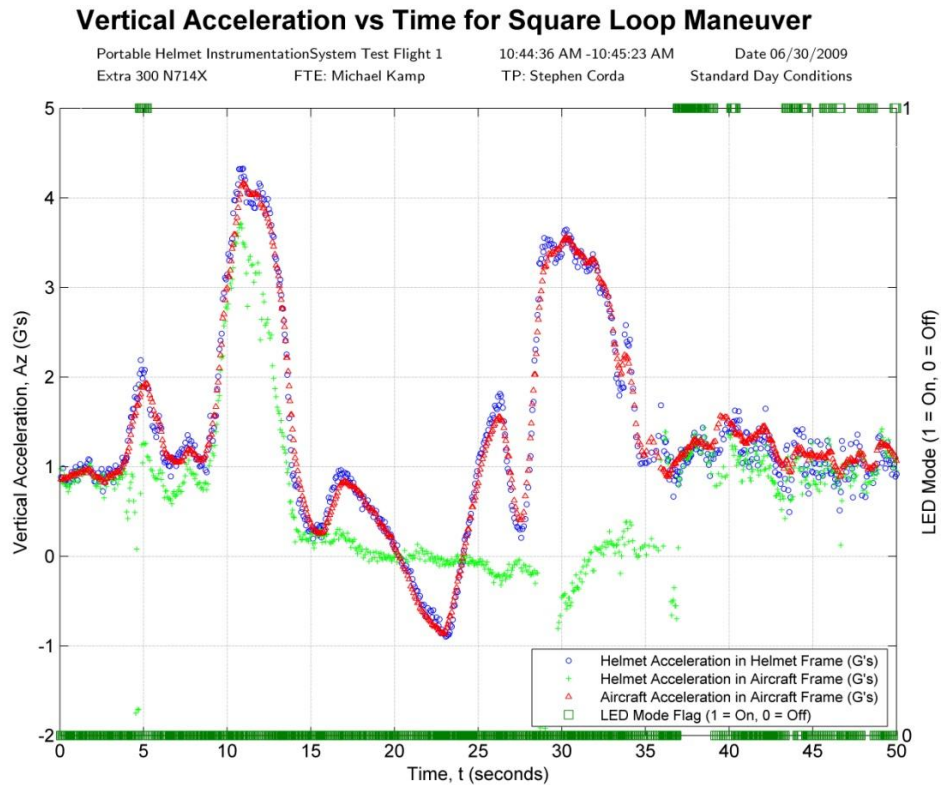


Figure B - 26: Vertical Acceleration versus Time for Square Loop Maneuver

Longitudinal Acceleration vs Time for Square Loop Maneuver

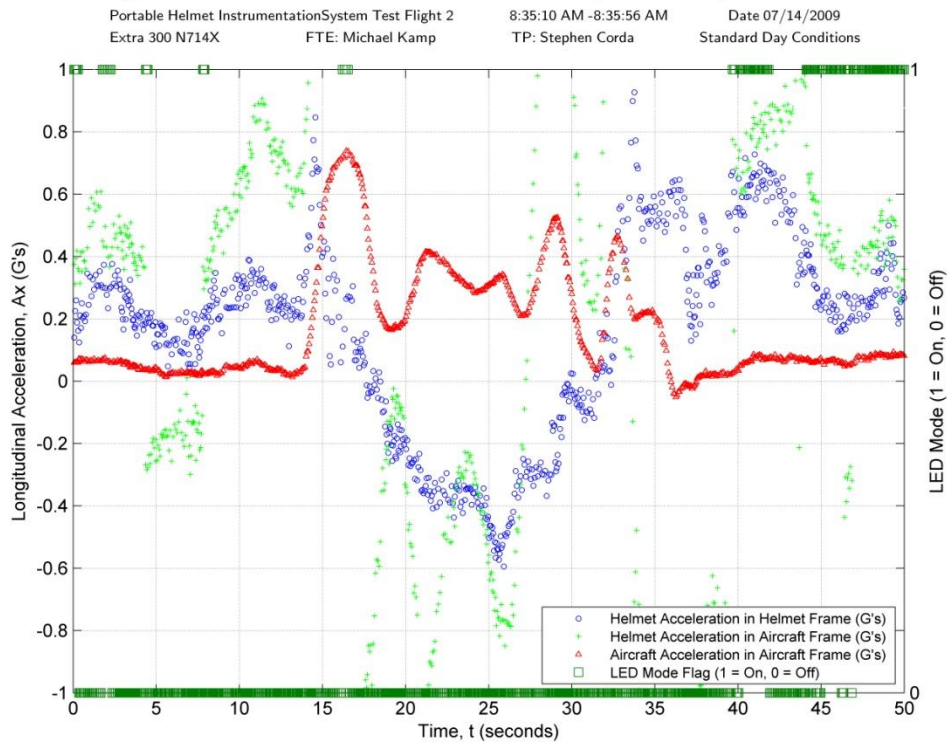


Figure B - 27: Longitudinal Acceleration versus Time for Square Loop Maneuver

Vertical Acceleration vs Time for Square Loop Maneuver

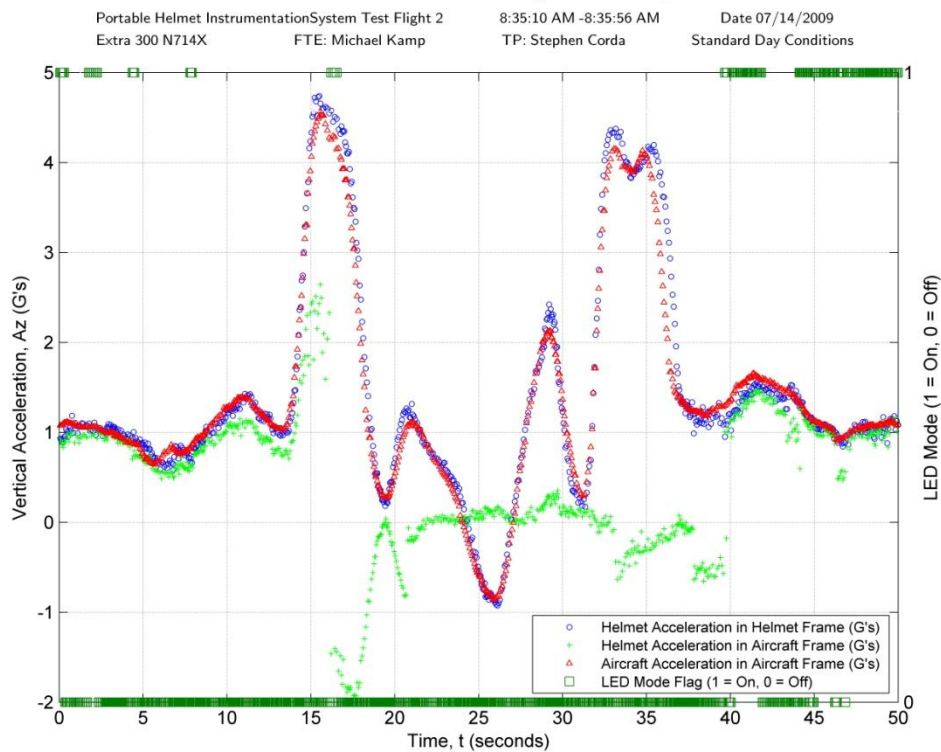


Figure B - 28: Vertical Acceleration versus Time for Square Loop Maneuver

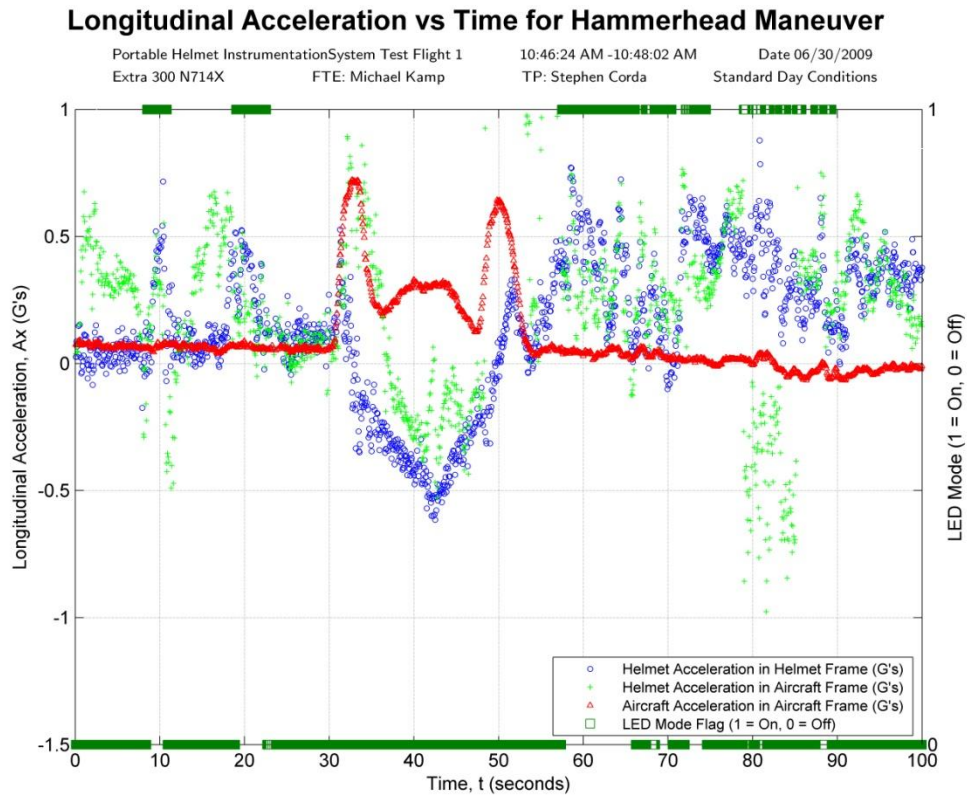


Figure B - 29: Longitudinal Acceleration versus Time for Hammerhead Maneuver

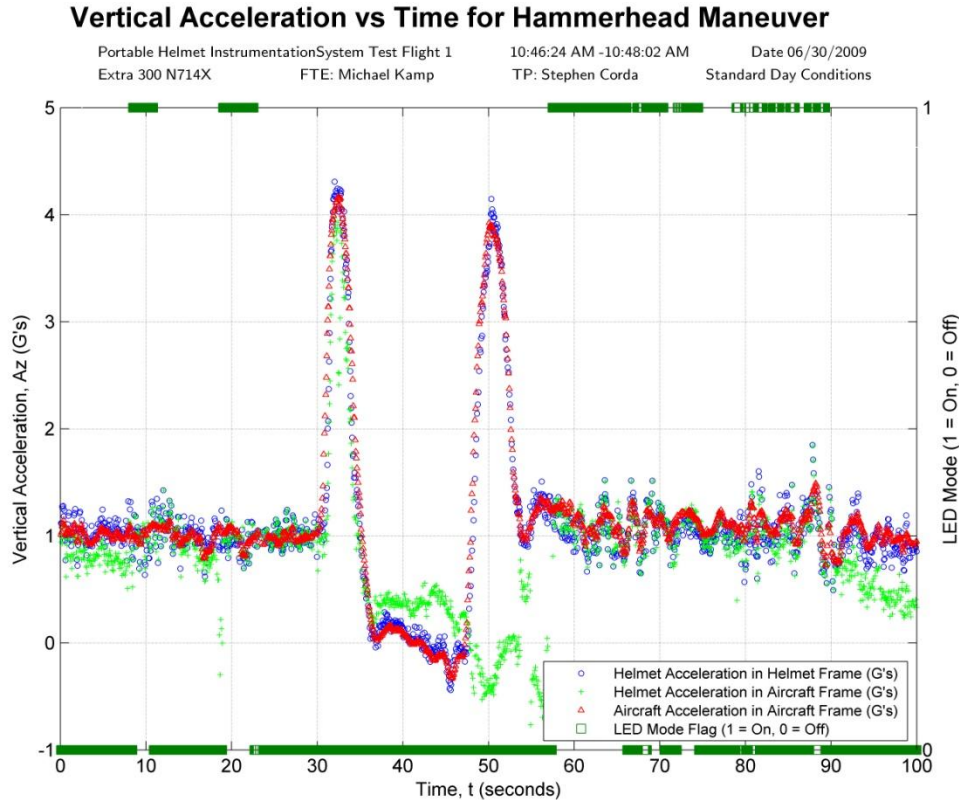


Figure B - 30: Vertical Acceleration versus Time for Hammerhead Maneuver

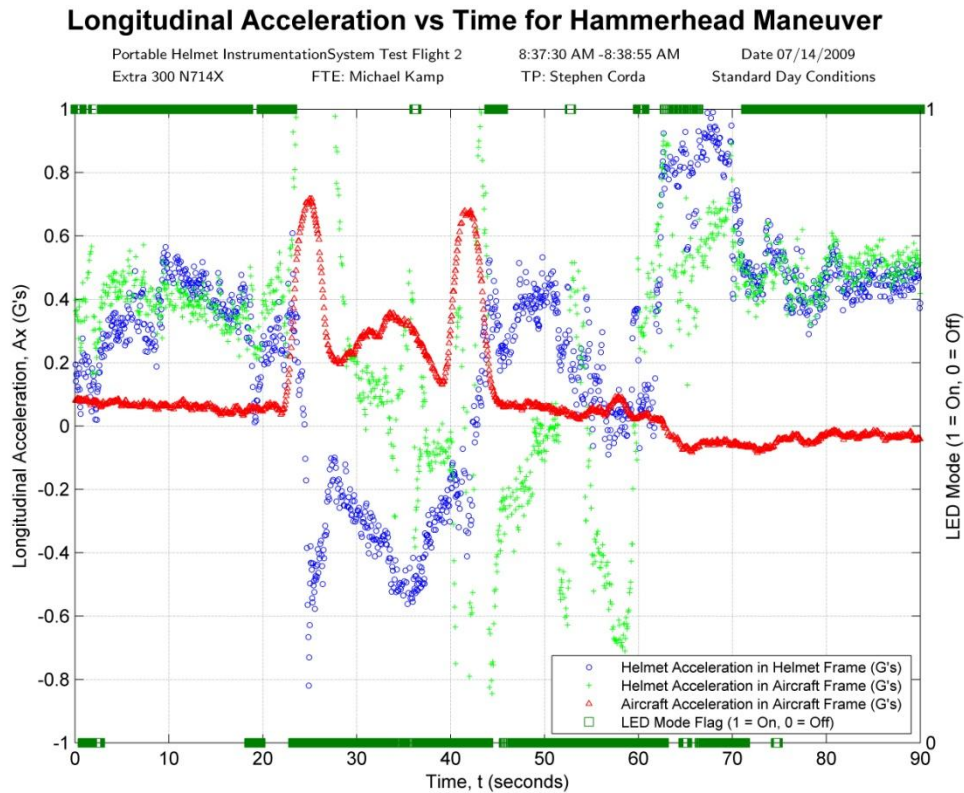


Figure B - 31: Longitudinal Acceleration versus Time for Hammerhead Maneuver

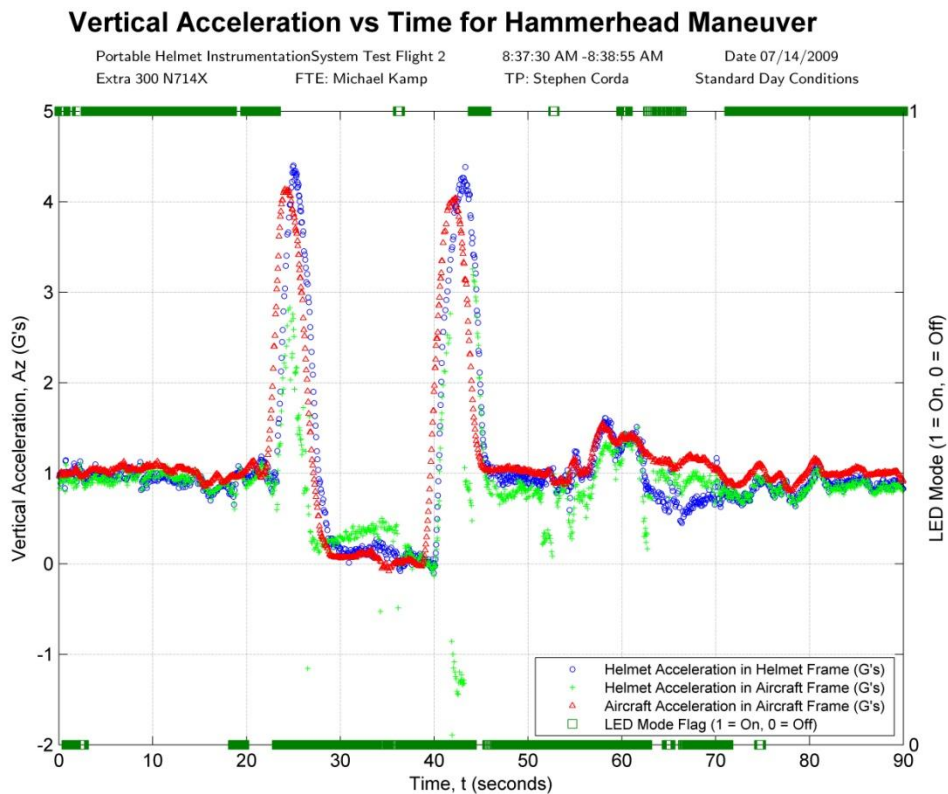


Figure B - 32: Vertical Acceleration versus Time for Hammerhead Maneuver

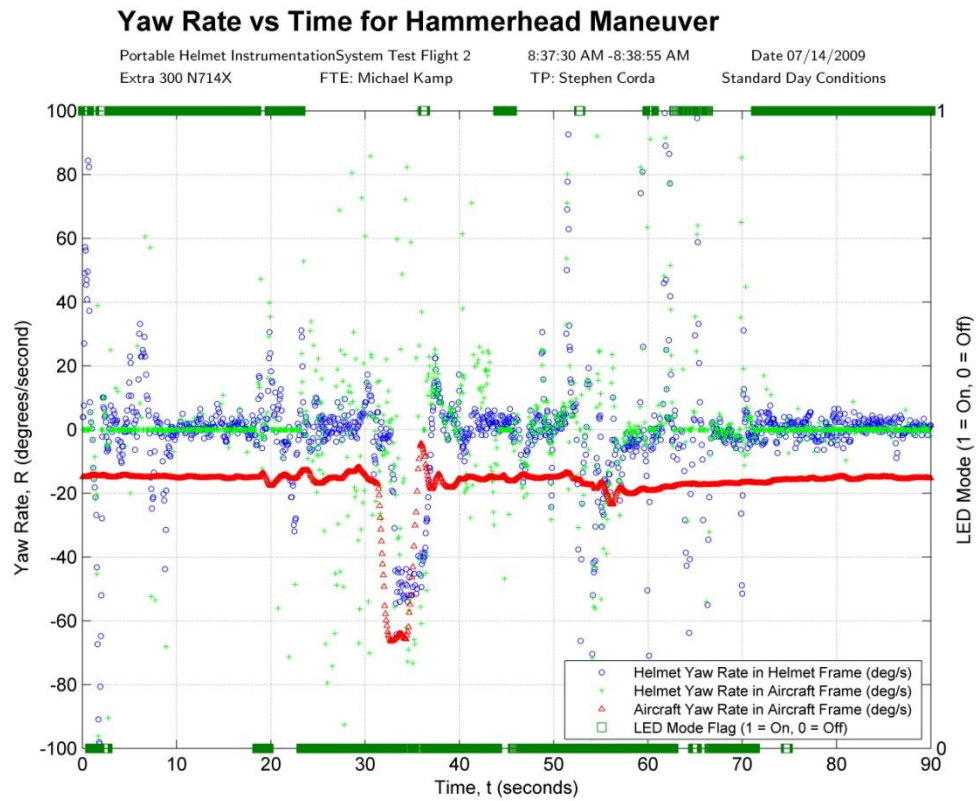


Figure B - 33: Yaw Rate versus Time for Hammerhead Maneuver

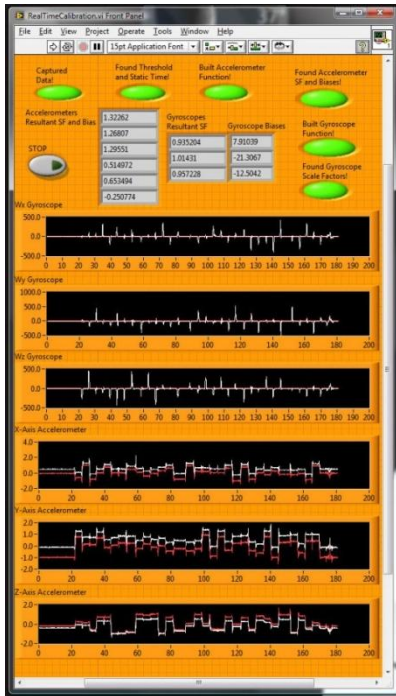


Figure B - 34: Multi-Position Calibration Method Main Panel

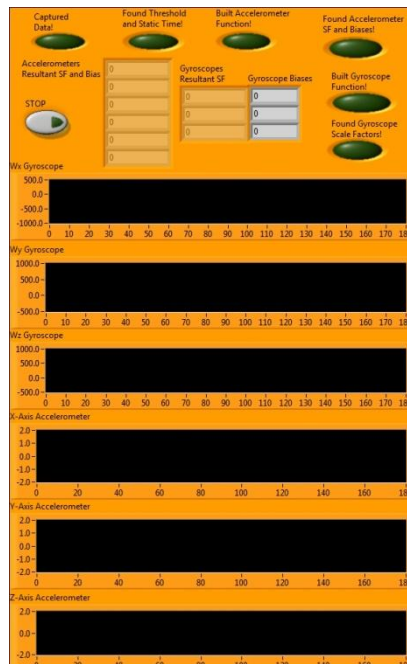


Figure B - 35: Multi-Position Calibration Method Labview Front Panel

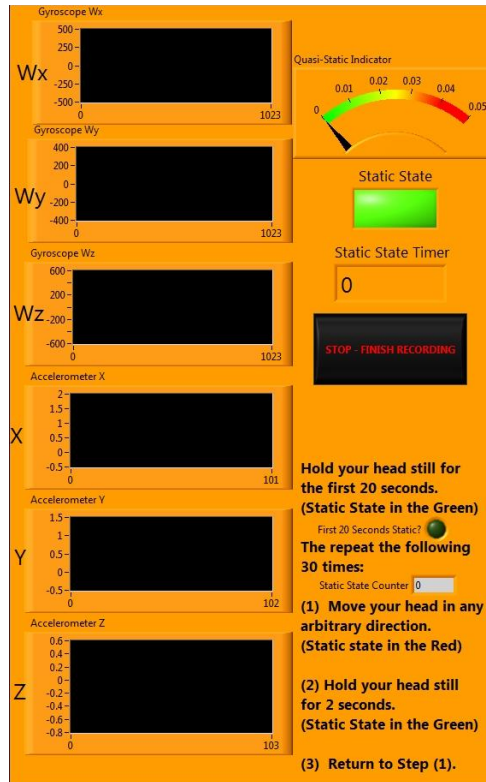


Figure B - 36: Multi-Position Calibration Data Collection Panel

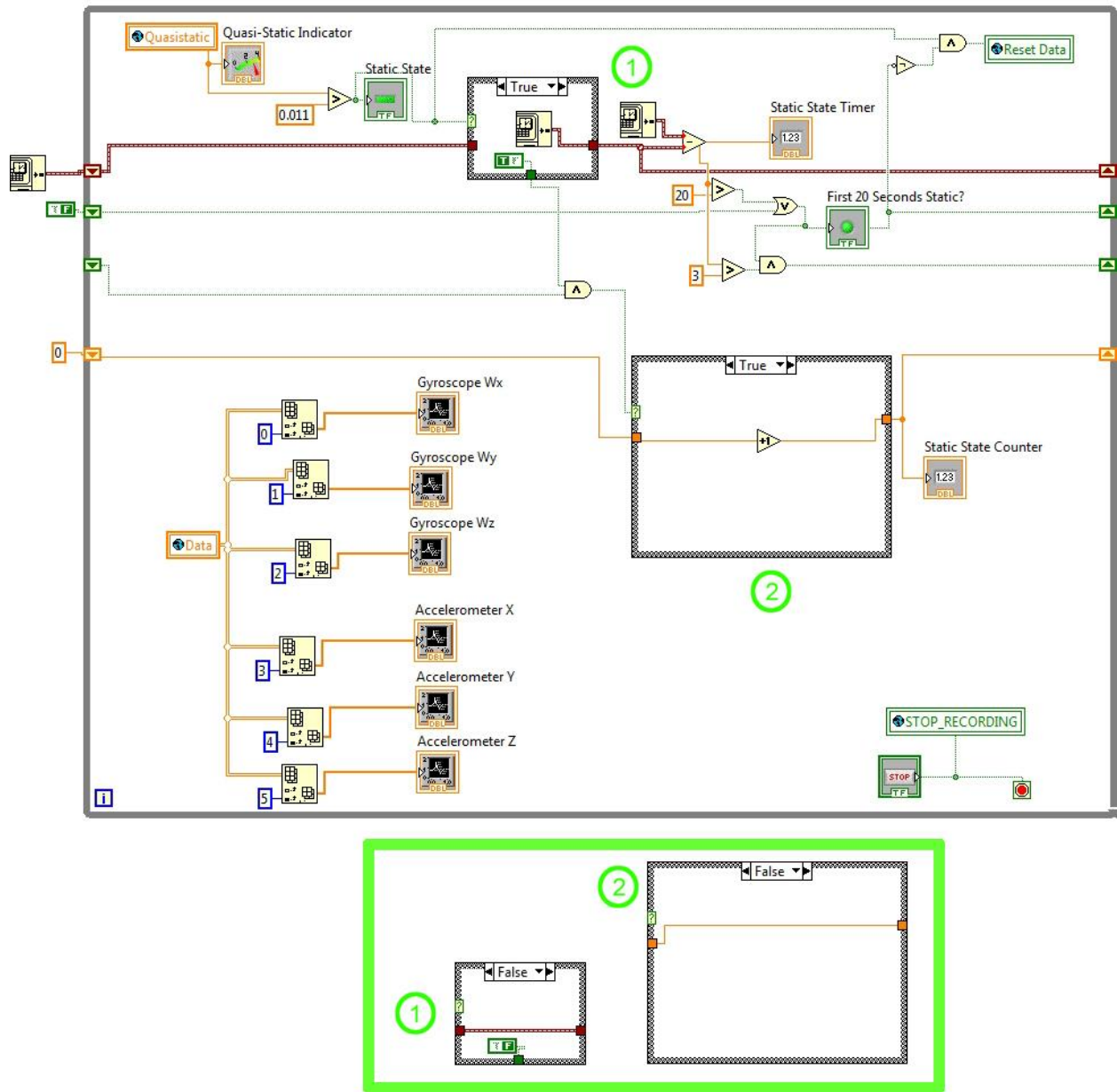


Figure B - 37: Multi-Position Calibration Data Collection Labview Diagram

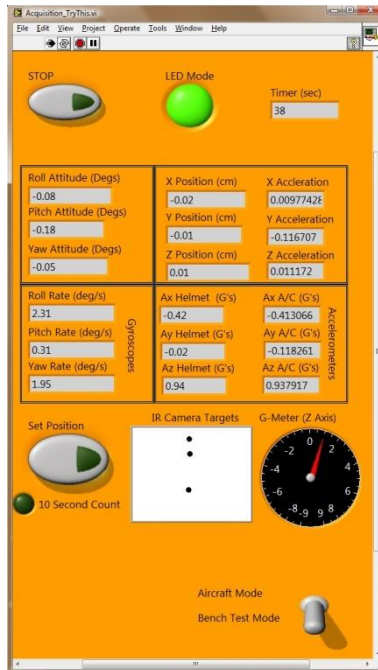


Figure B - 38: PHIS Data Acquisition in use

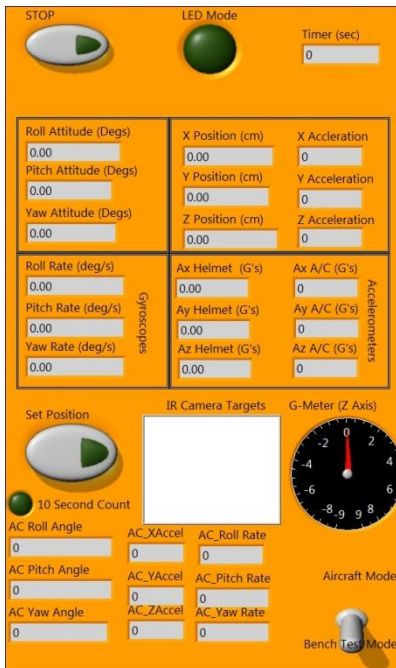
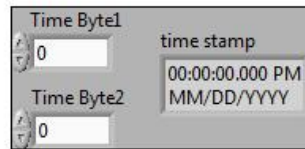


Figure B - 39: PHIS Data Acquisition Labview Front Panel



Takes a timestamp that was spit across the bytes of two singles and rejoins to make a timestamp.

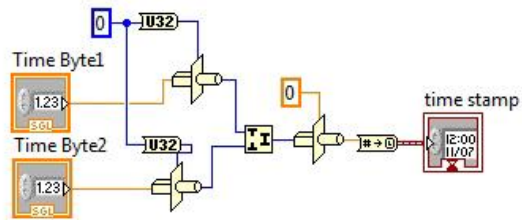


Figure B - 40: Two Singles to Time Stamp Labview Diagram

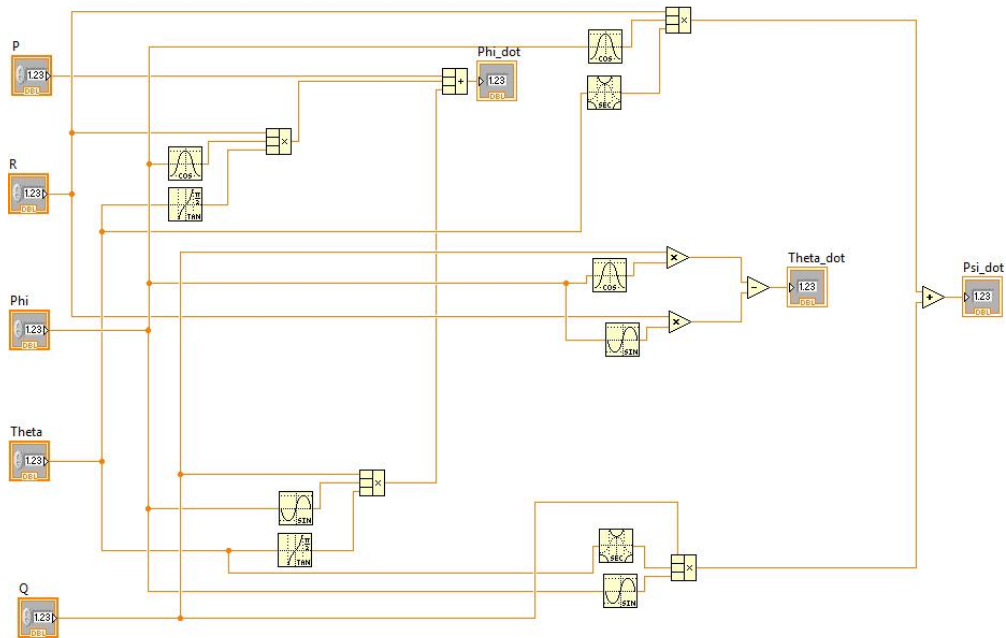


Figure B - 41: Angular Transform Labview Diagram

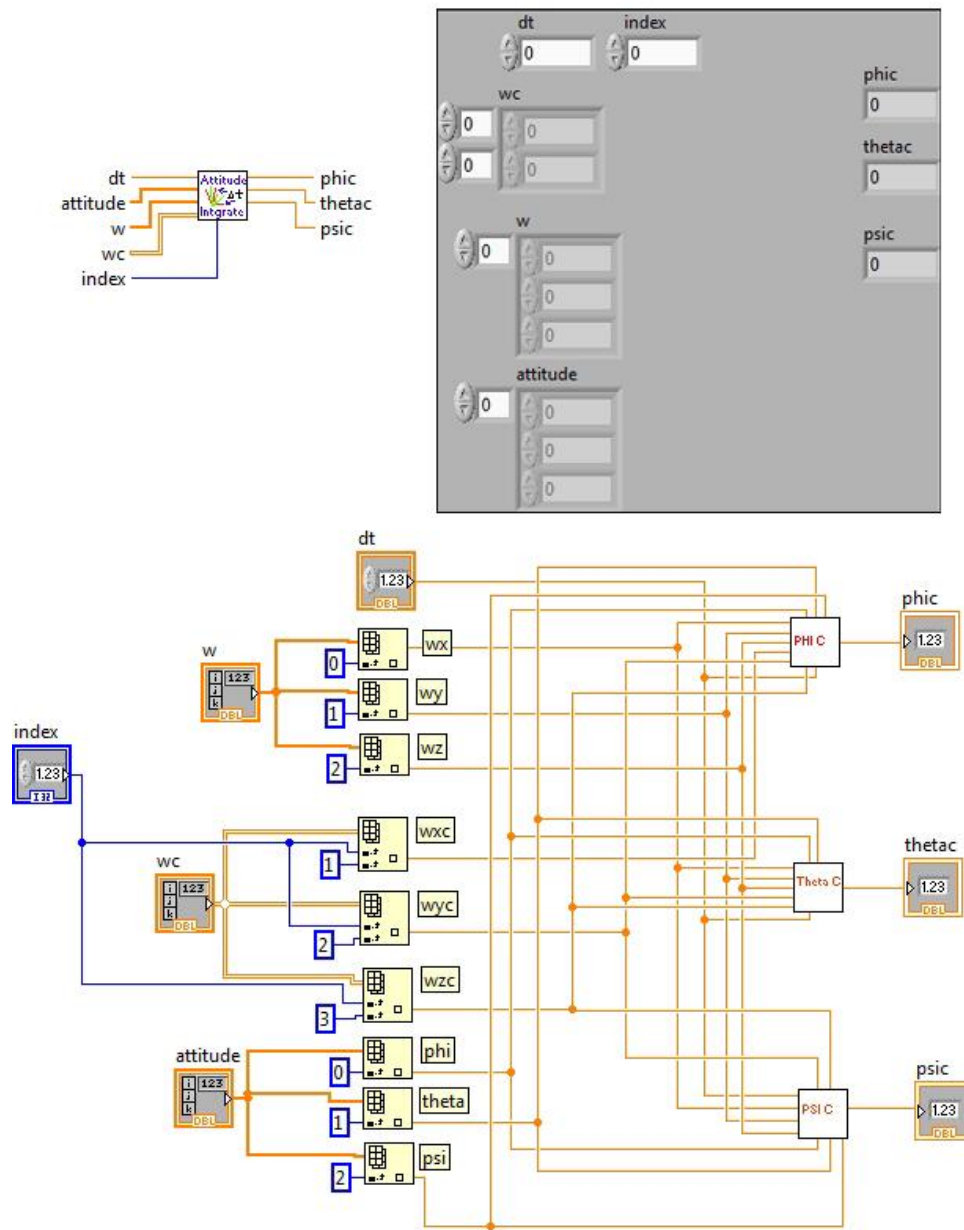


Figure B - 42: Attitude Integration Labview Diagram

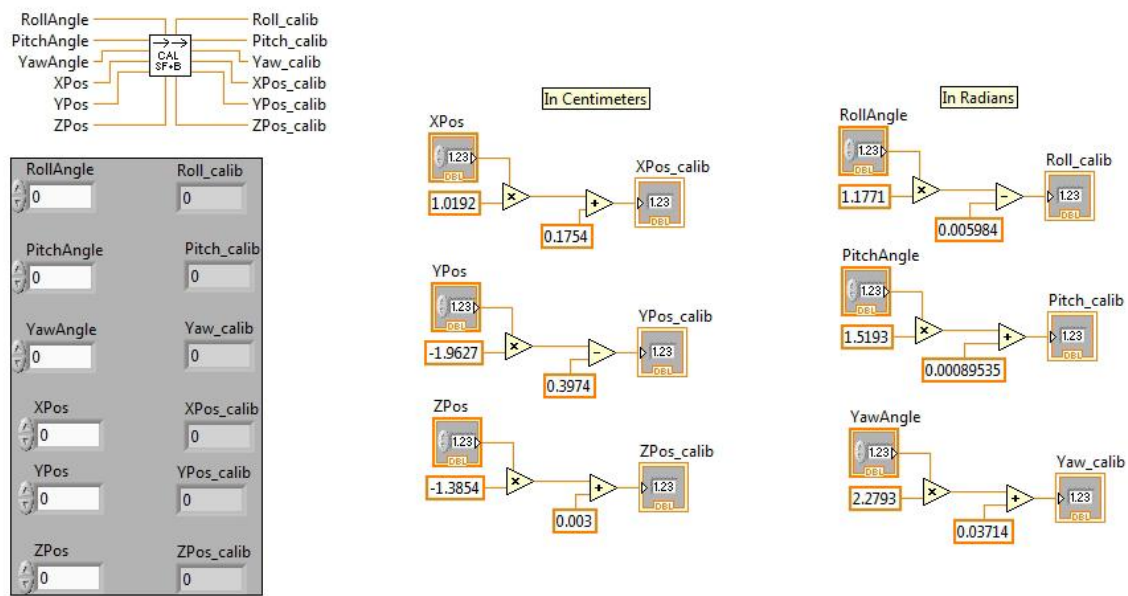


Figure B - 43: IR-Camera Calibration Correction Labview Diagram

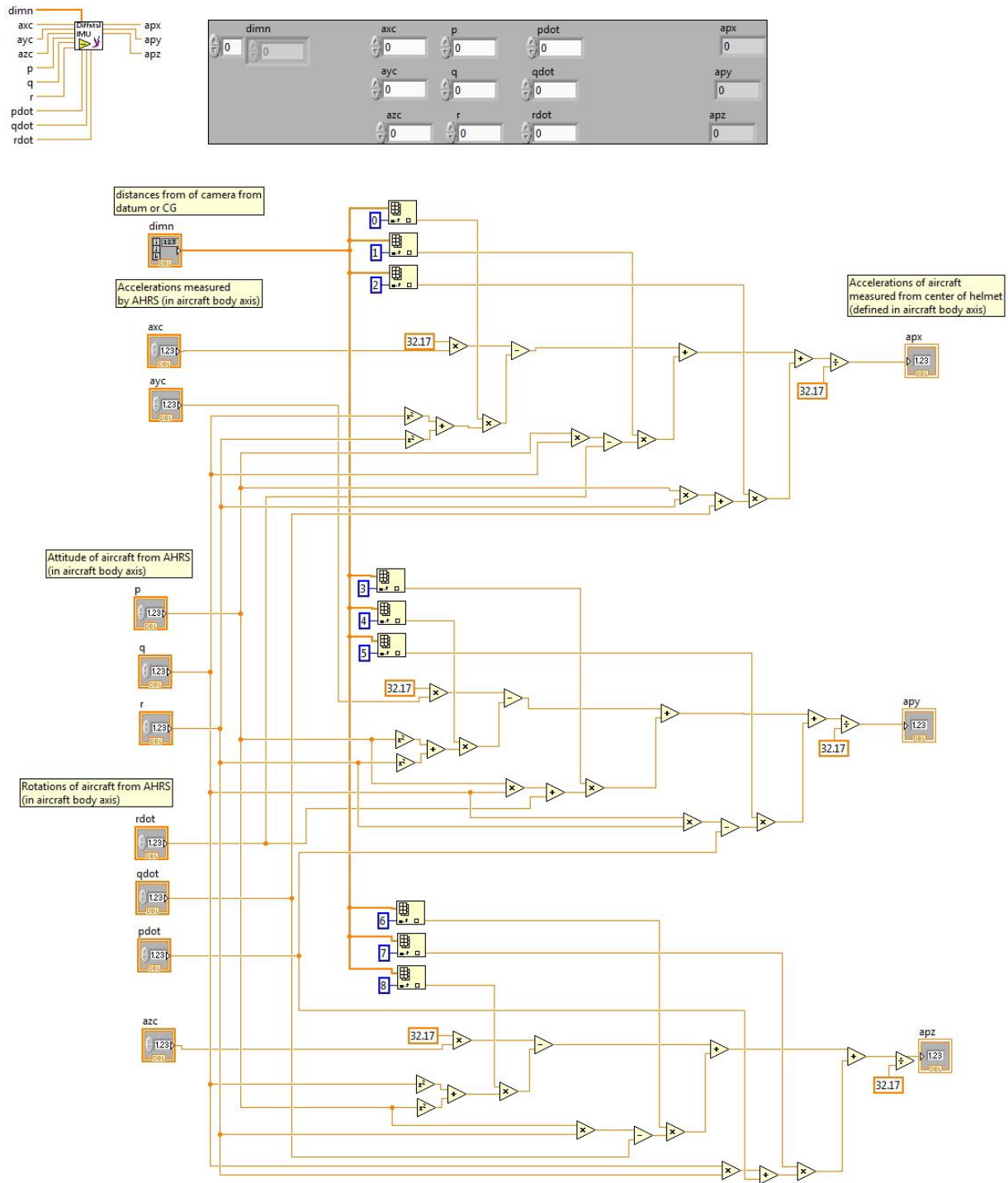


Figure B - 44: Differential IMU Labview Diagram

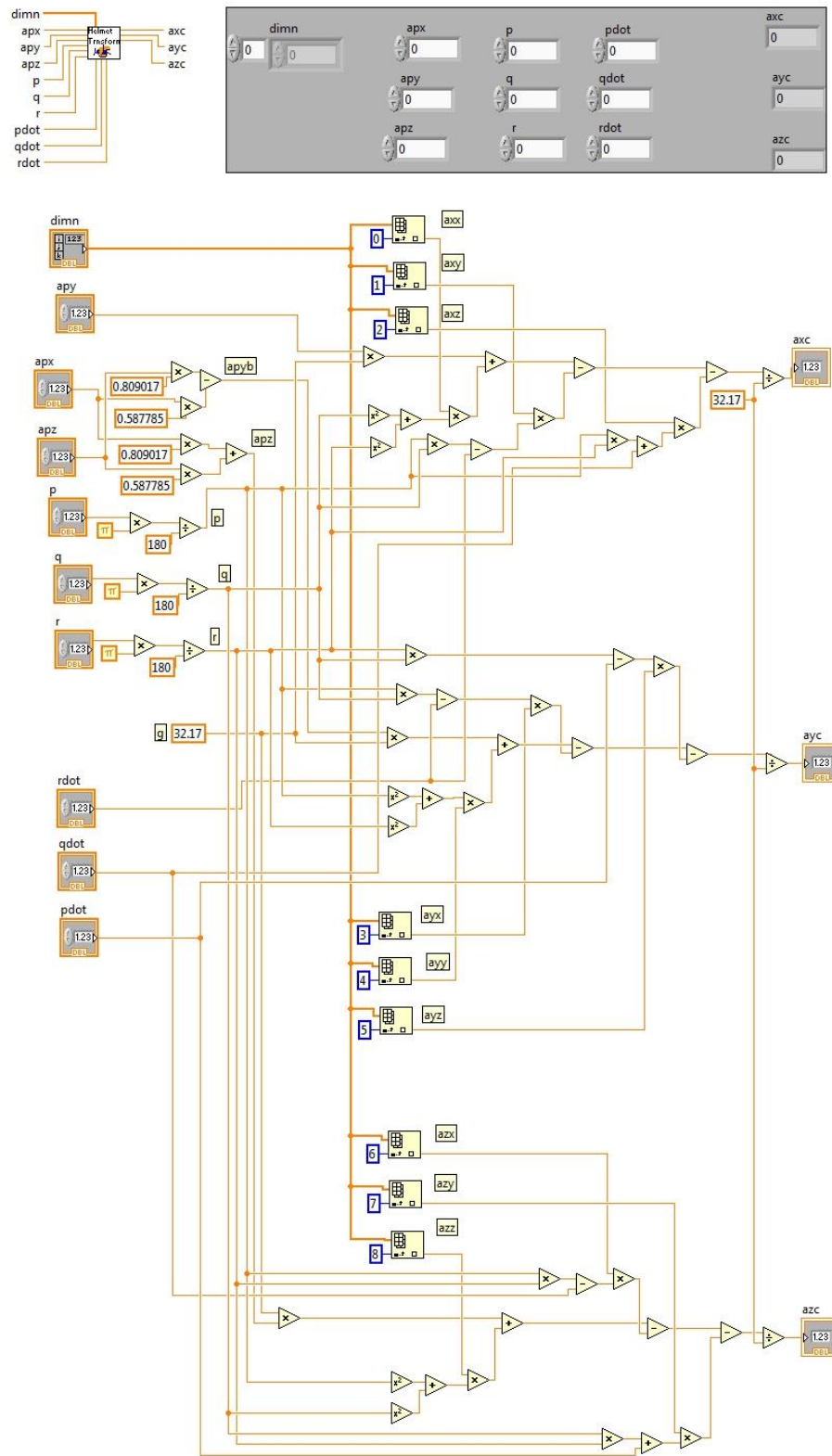


Figure B - 45: Helmet Transformation Labview Diagram

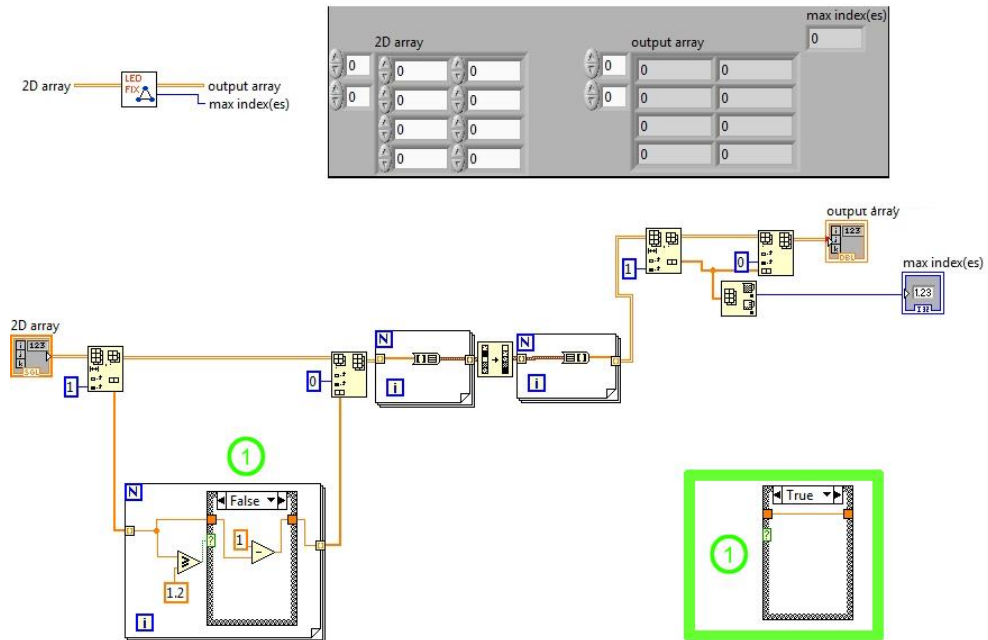


Figure B - 46: LED Clip Fix Labview Diagram

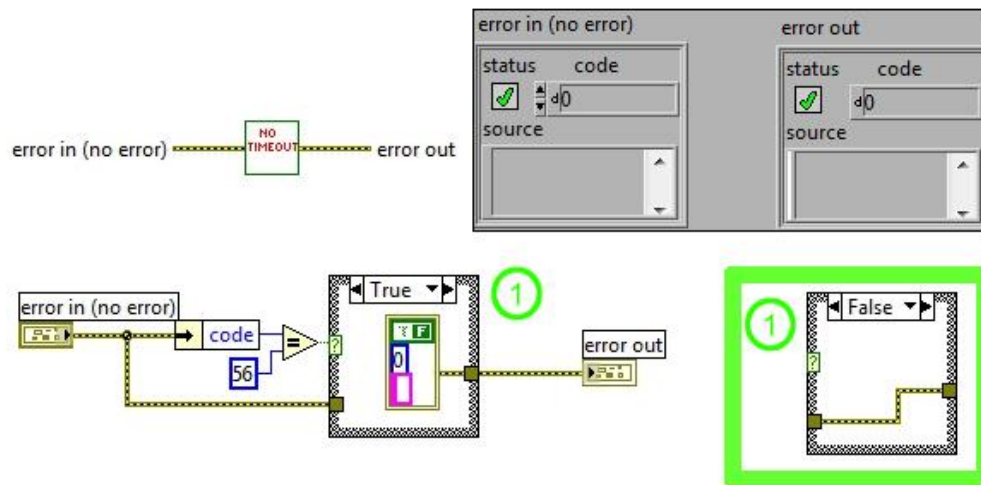


Figure B - 47: No Timeout Error Labview Diagram

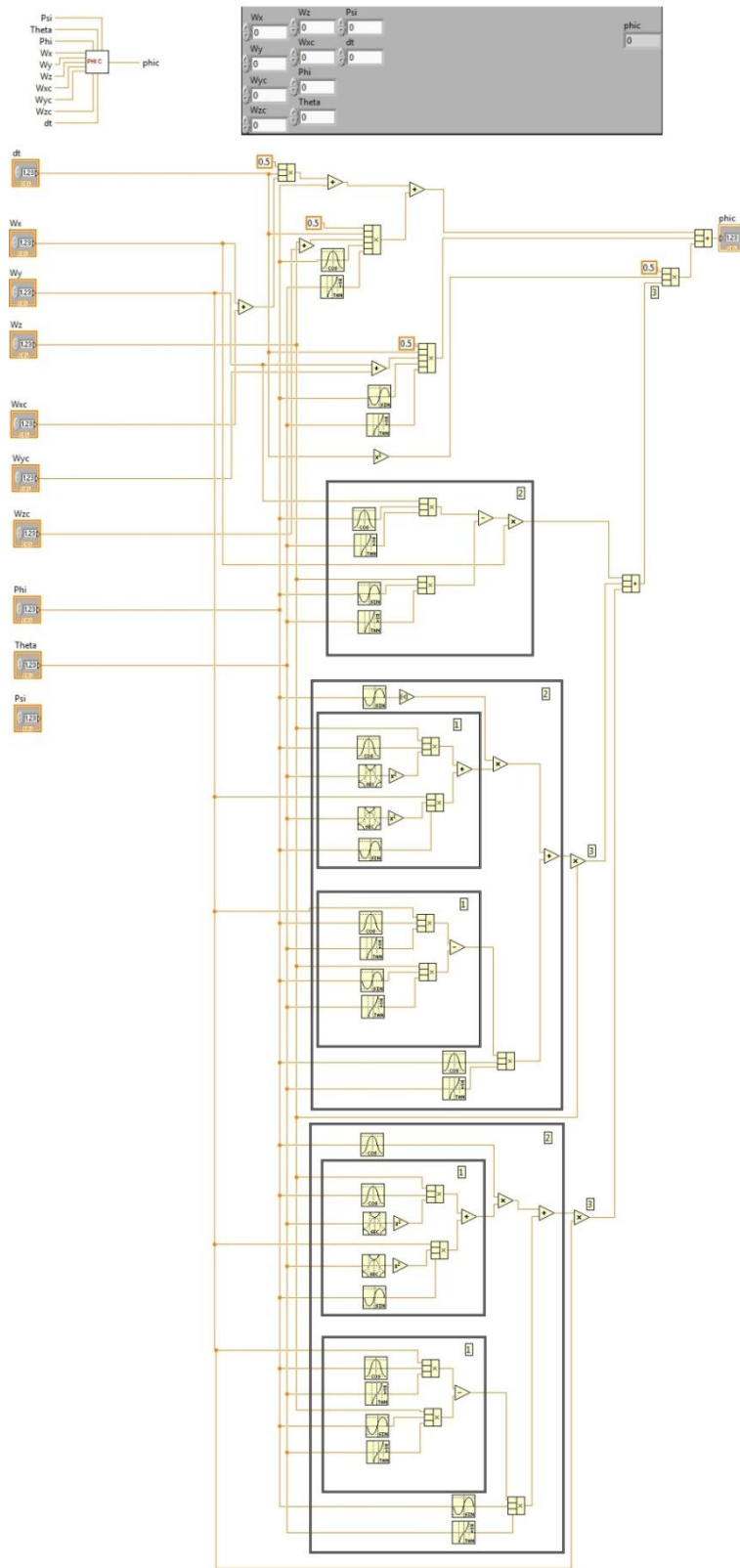


Figure B - 48: Phi Correction Labview Diagram

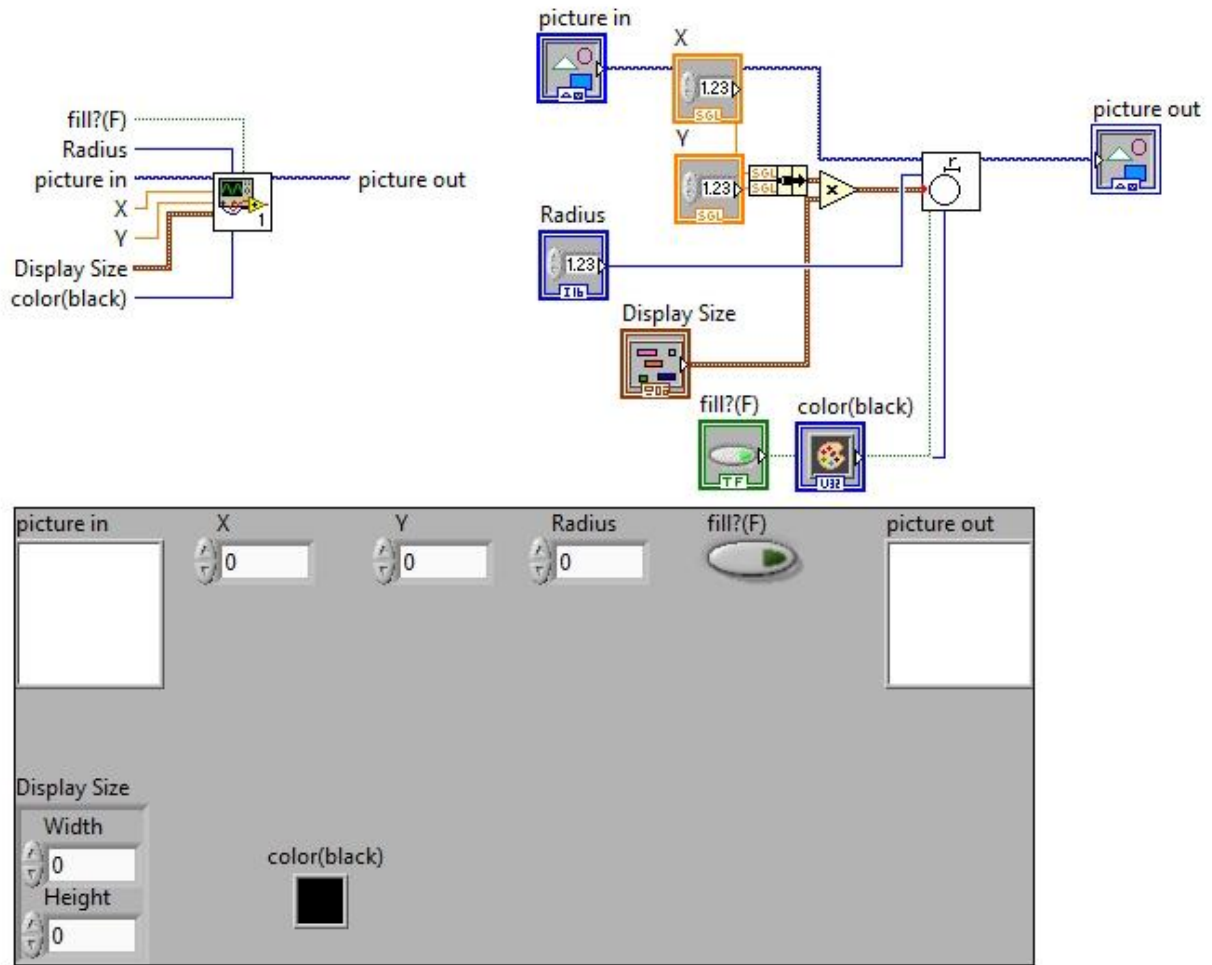


Figure B - 50: Draw Scaled Circle Labview Diagram

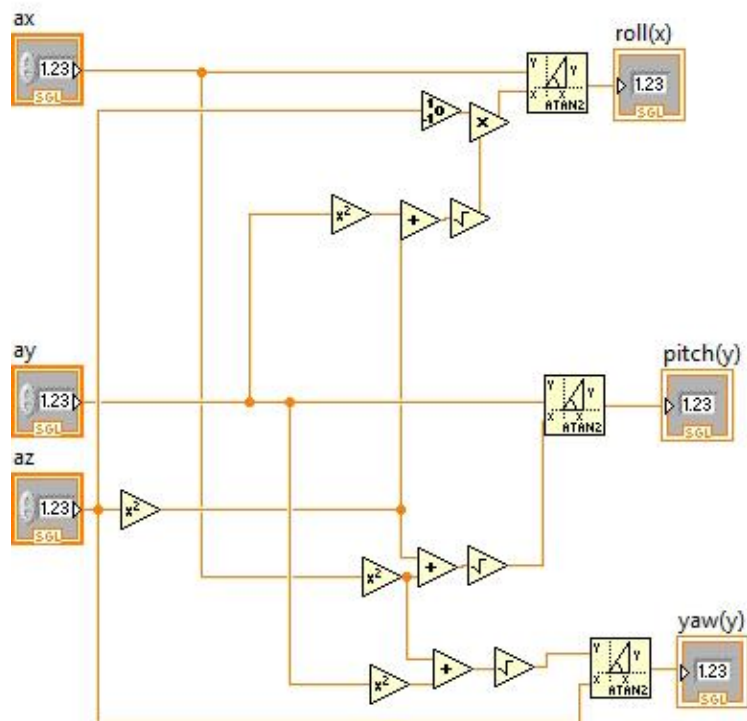
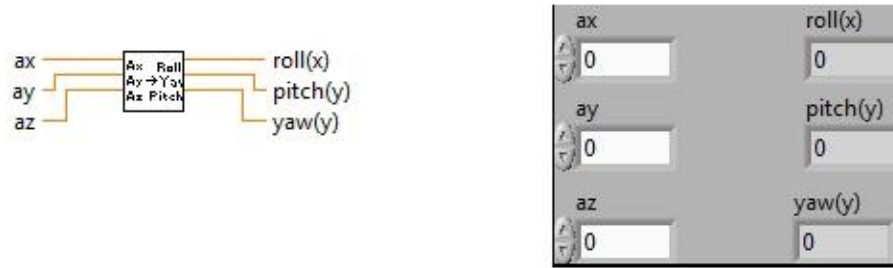


Figure B - 51: Modified Subroutine Roll Pitch Yaw Labview Diagram

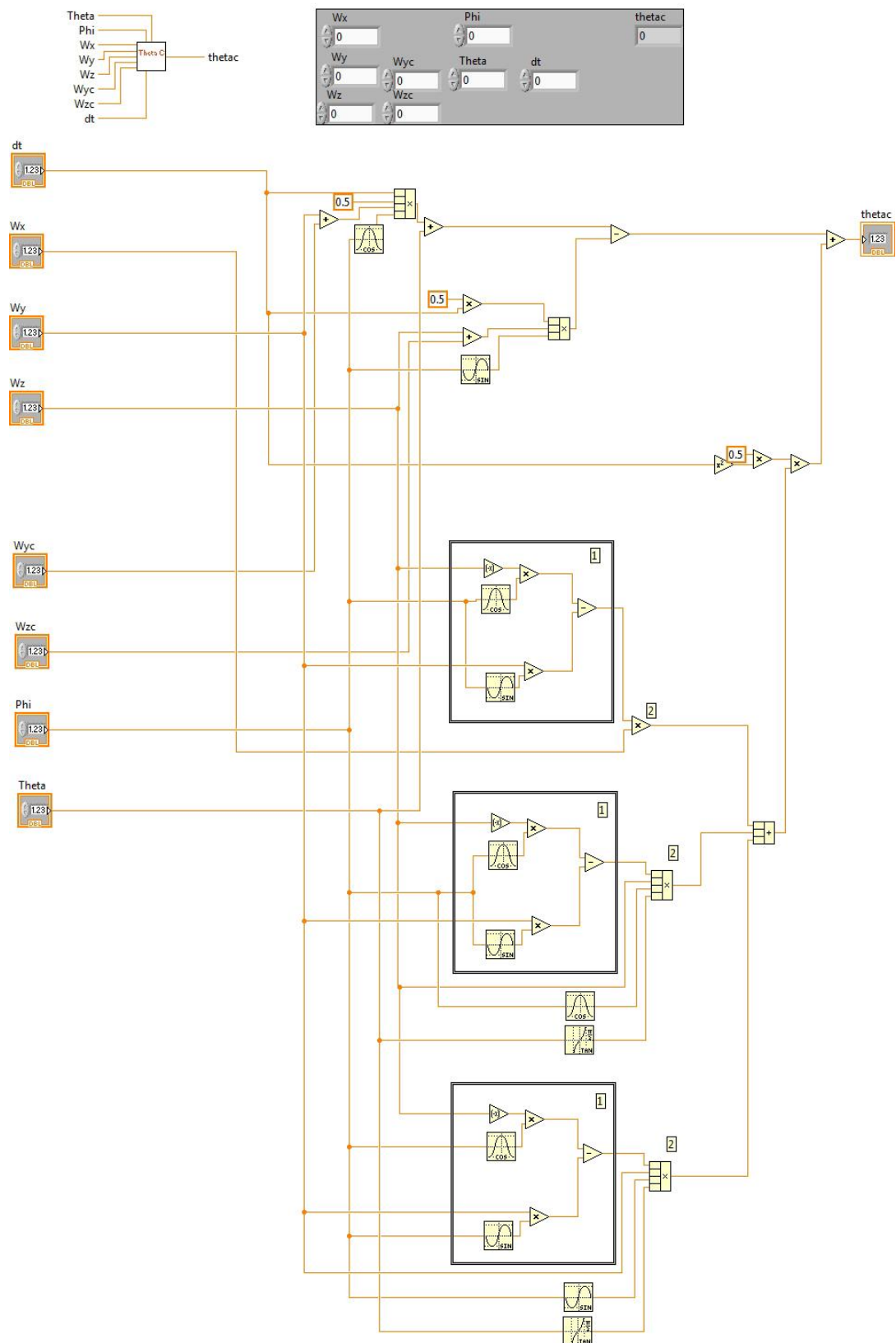


Figure B - 52: Theta Correction Labview Diagram

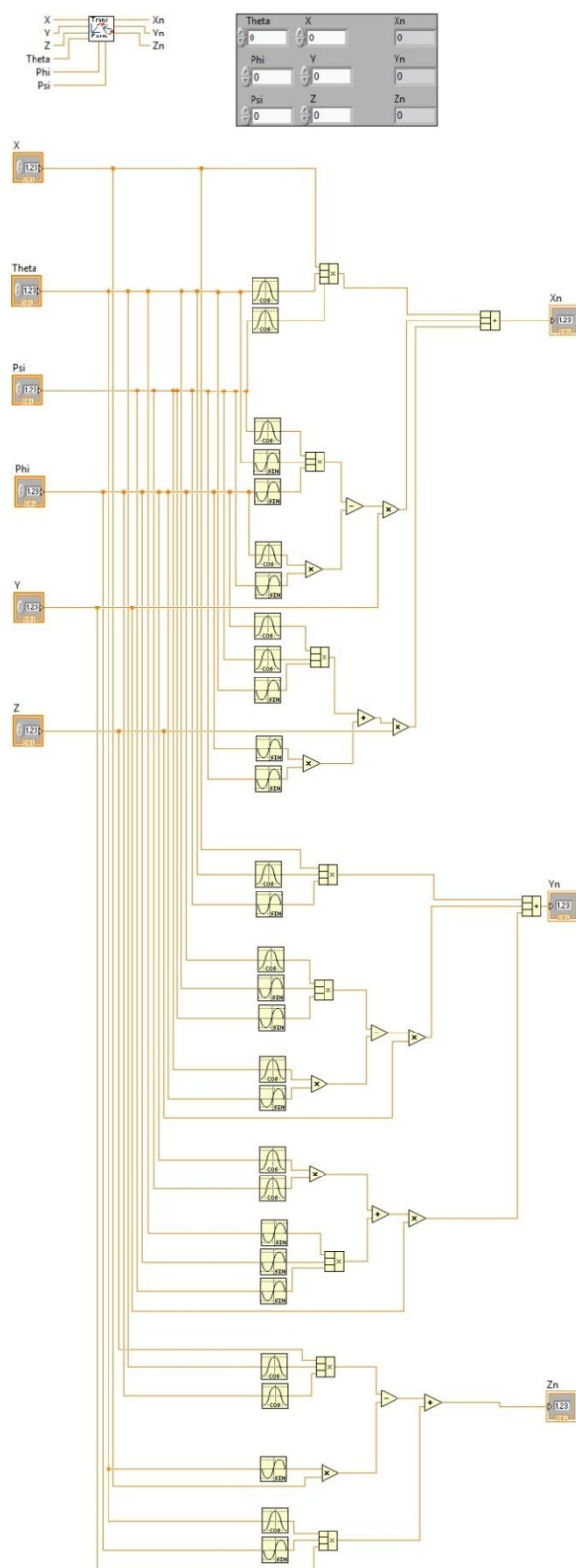


Figure B - 53: Transformation Labview Diagram

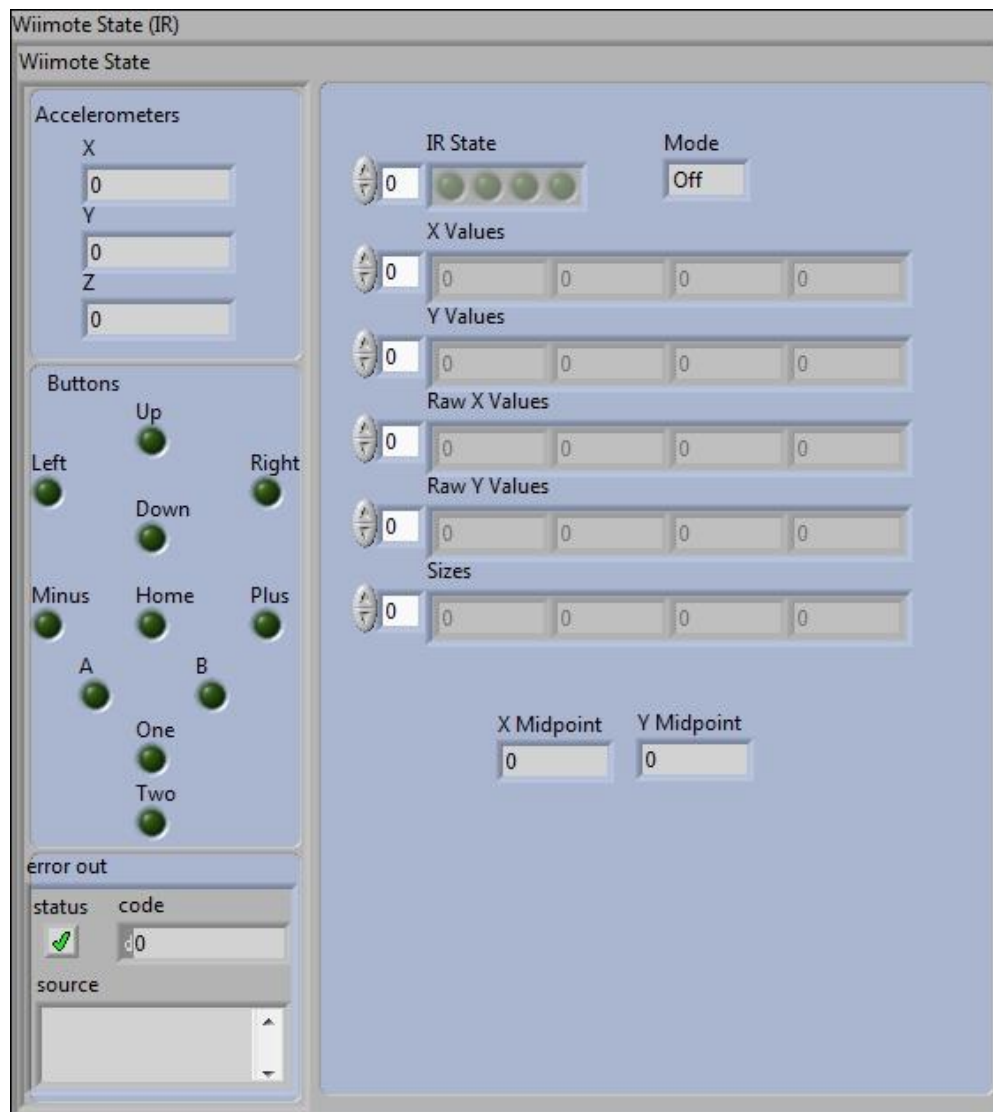


Figure B - 54: WiiMote State (IR) Labview Control[26]

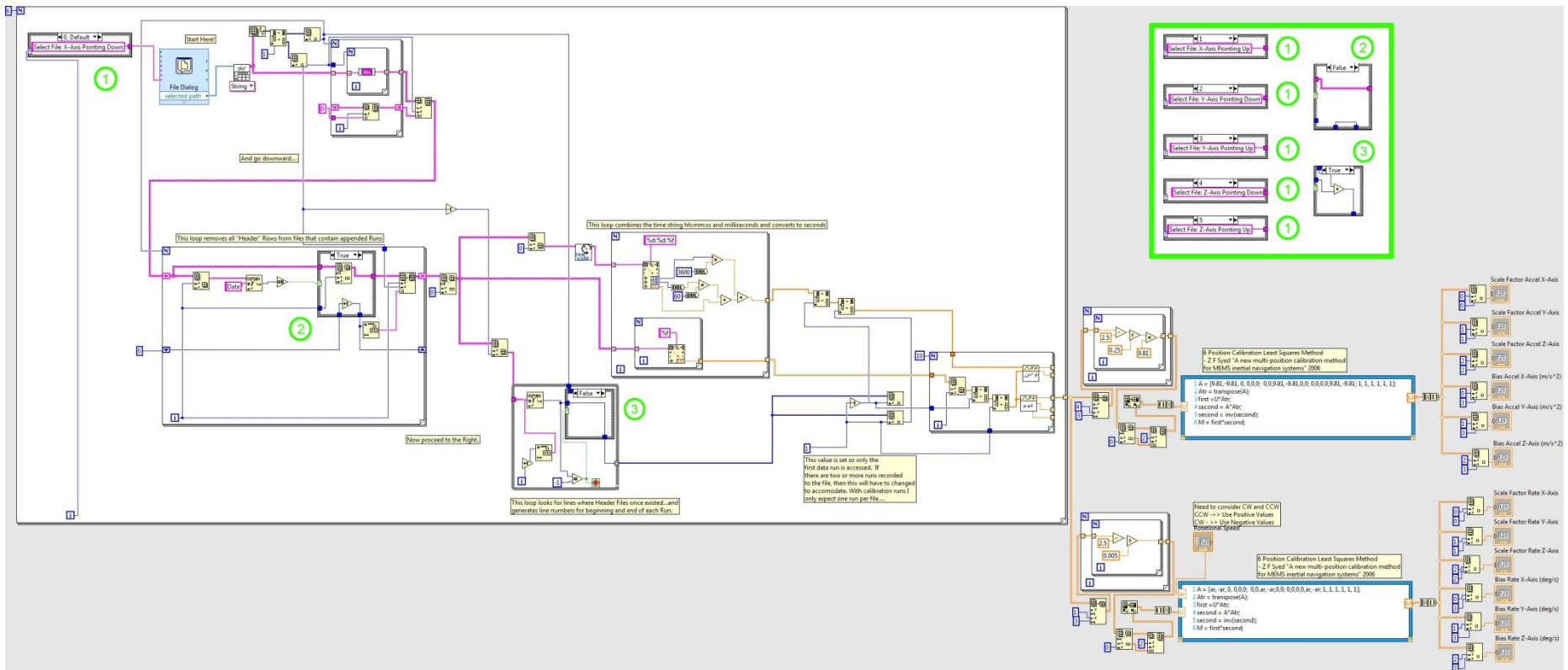


Figure B - 55: Six Position Method Labview Diagram

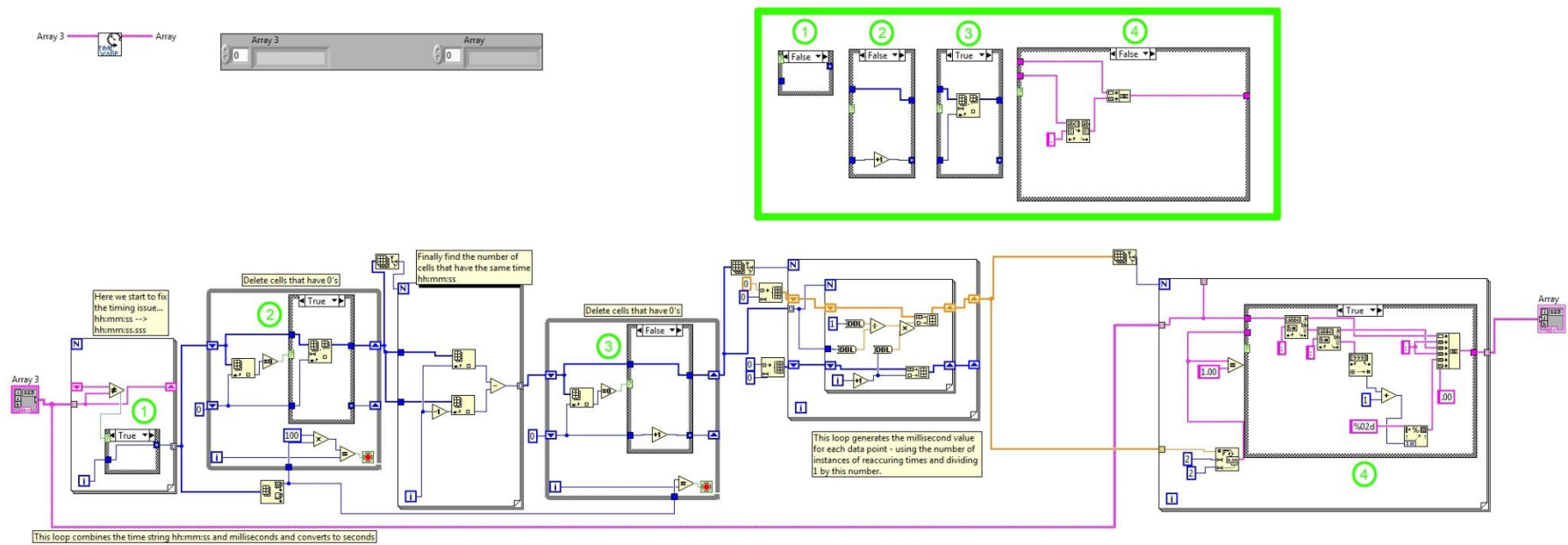


Figure B - 56: Time Fix for Six Position Method Labview Diagram

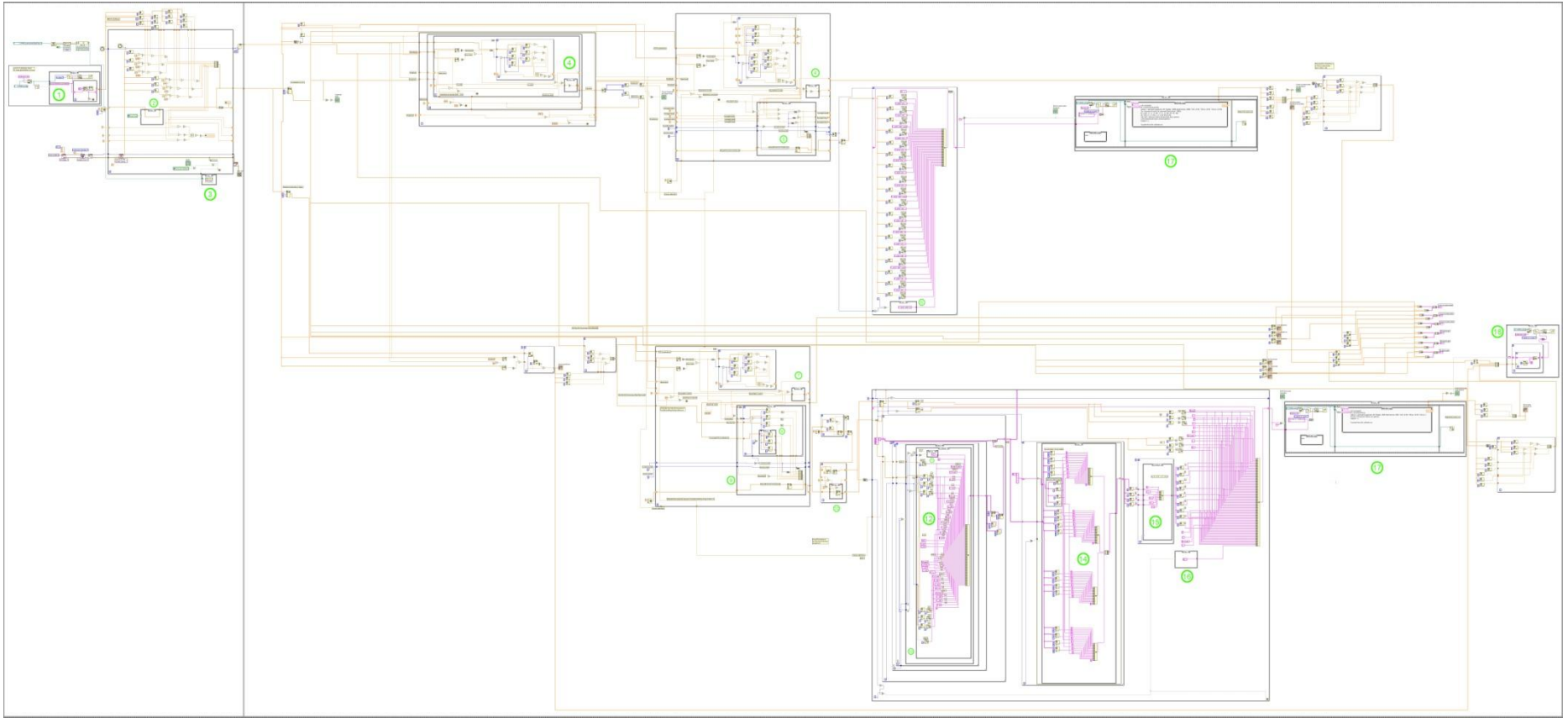


Figure B - 57: Multi-Position Calibration Method Labview Diagram

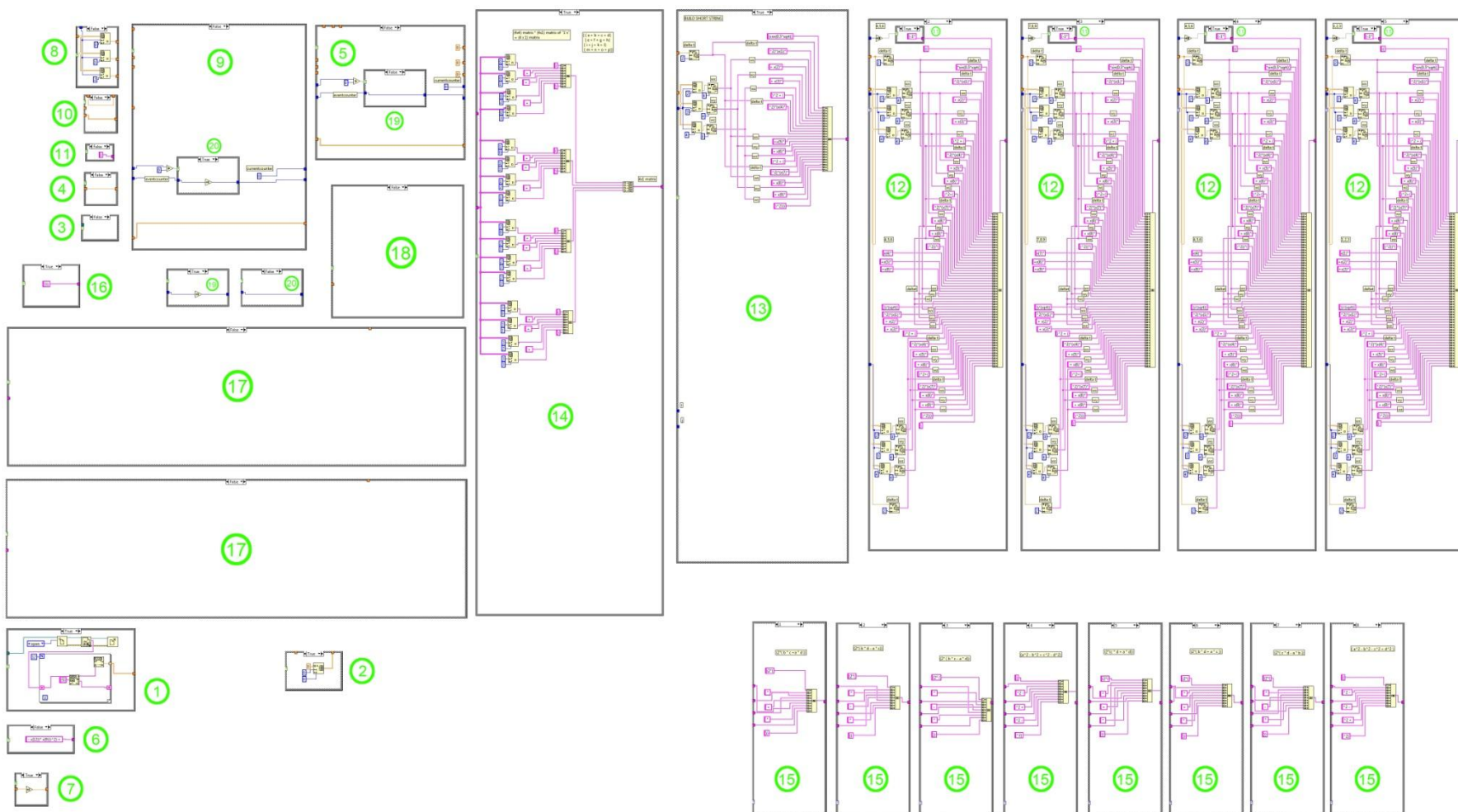


Figure B - 58: Multi-Position Calibration Method Labview Diagram Alternate Cases

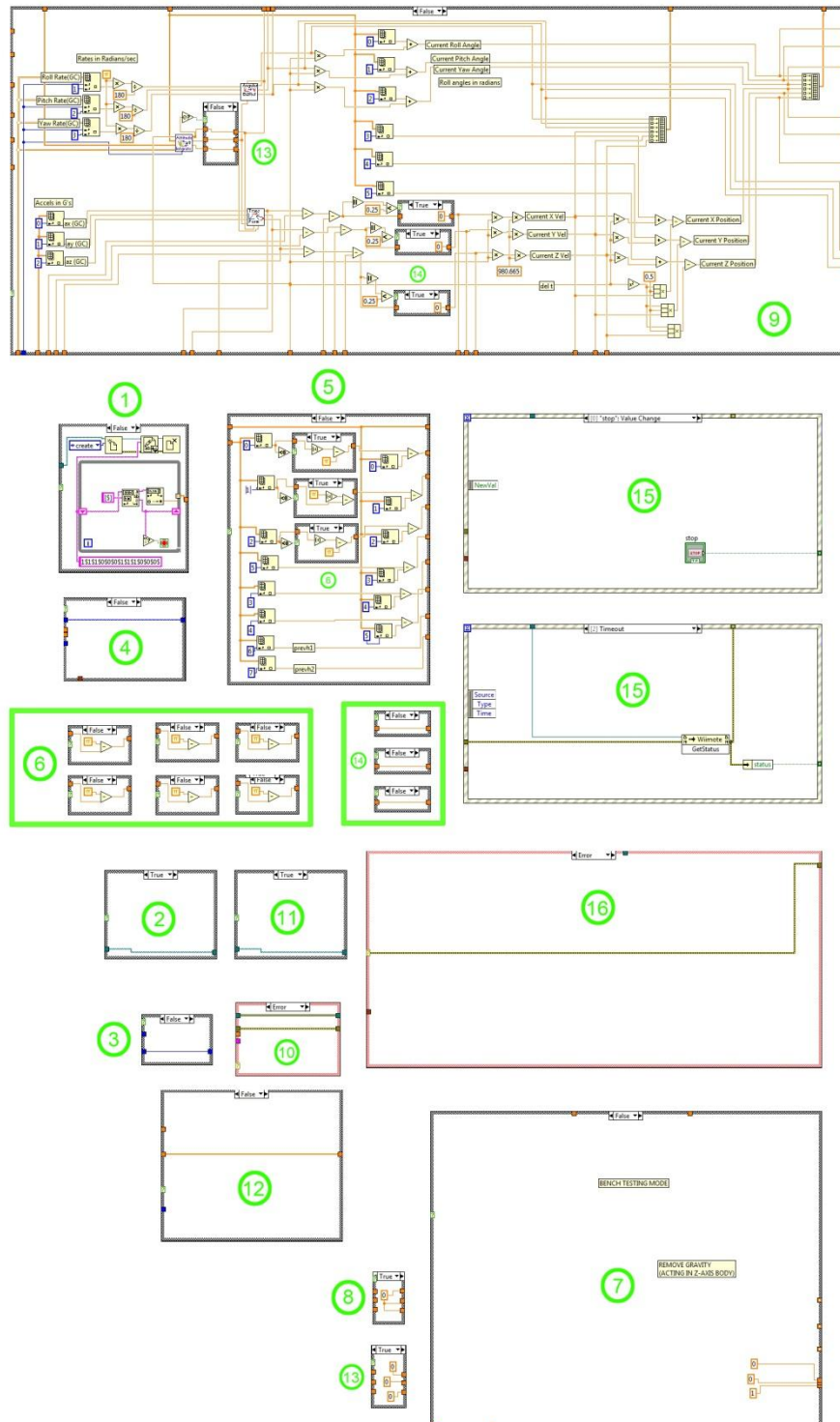


Figure B - 60: PHIS Data Acquisition Labview Diagram Alternate Cases

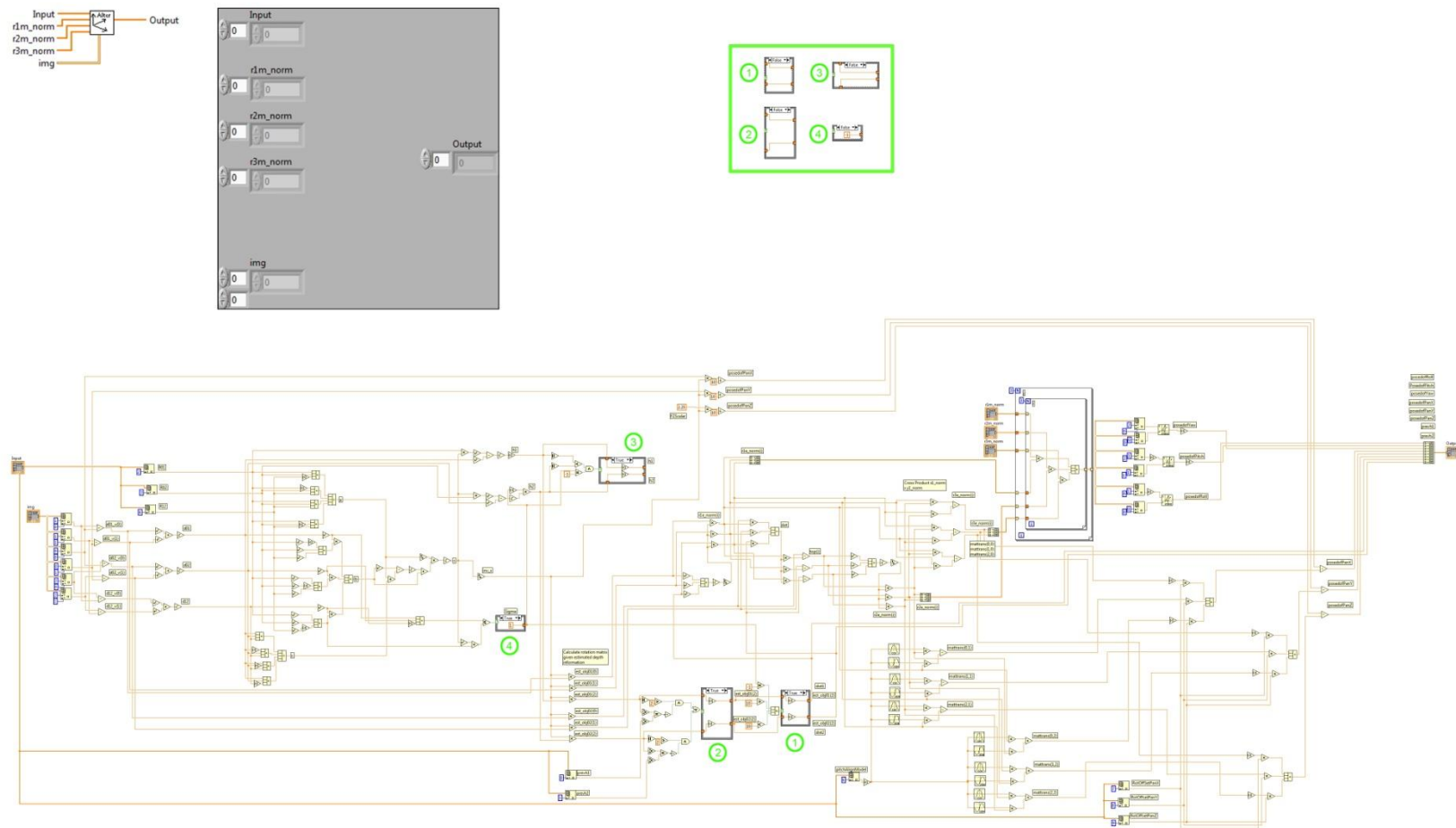


Figure B - 61: Alter 3D Pose Determination Labview Diagram

Vita

Michael A Kamp was born in Ft. Stewart, Georgia. He studied Aerospace Engineering at the University of Cincinnati and was awarded a Bachelor's Degree in June 2007. He then enrolled at the University of Tennessee Space Institute for an advanced degree.