Precipitation Processing System (PPS)

NASA Global Precipitation Measurement (GPM)
Geolocation Toolkit
Algorithm Theoretical Basis Document (ATBD)
Version 2.1

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CM FOREWORD

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CHANGE RECORD PAGE

Mod. #	Mod. Date	Description	Section Affected
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Item No.	Location	Summary	Ind./Org.	Due Date

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1.0 INTRODUCTION

1.1 OBJECTIVE

This document describes the algorithms for the Geolocation Toolkit (GeoTK) for the Global Precipitation Measurement (GPM) Mission. The core part of the algorithm uses input orbit ephemeris, spacecraft attitude, and instrument pointing data to compute each pixel latitude and longitude viewed, along with ancillary data such as zenith/incidence and Sun angle data. These calculations are implemented in the GeoTK software subroutines, which will be used for Level 1B (L1B) algorithms for GPM. The details of this software structure, inputs and outputs, are documented in the PPS Geolocation Toolkit Architecture and Design Specification Document (Bilanow, 2012). In addition, this document provides a description of how the sensor alignment angles for input to the algorithm are defined, and how these angles may be adjusted based on misalignments observed after launch.

1.2 GPM SPACECRAFT OVERVIEW

GPM, a follow-on mission to the Tropical Rainfall Measuring Mission (TRMM), is a joint project between the National Aeronautics and Space Administration (NASA) and the Japan Aerospace Exploration Agency (JAXA). An overview of the mission, the science instruments, the orbit and attitude, and various modes of operation and data handling are given in the GPM Core Observatory Performance Requirements (GPM-SE-REQ-0030, 2012).

There are two main science instruments onboard the GPM spacecraft: The GPM Microwave Imager (GMI) and the Dual-frequency Precipitation Radar (DPR). The DPR actually consists of two radars: One is in the Ka frequency band and is called the KaPR, while the other is in the Ku frequency band and is called the KuPR. The accurate co-alignment of the KaPR and KuPR matched beam directions is critical for the mission science objectives.

1.3 GPM SPACECRAFT COORDINATES

There are two GPM coordinate frames that are important for understanding the instrument alignment and the attitude definitions. Figure 1-1 shows the Spacecraft (S/C) and Flight (F) coordinates, which are nominally tilted about 4 degrees relative to each other.

The Spacecraft coordinates are physically defined on the structure by reference points and alignment (mirror) cubes, which are used repeatedly during spacecraft integration and test (I&T) to check the alignments relative to the science instruments and attitude sensors. Attitude sensors are instruments that help detect the orientation of the spacecraft, and each attitude sensor has its own coordinate frame. Key attitude sensors for GPM include the following. Two star trackers measure star positions to detect their orientation in inertial space. An inertial reference unit, also known as a gyro, detects rotations relative to inertial space about four independent axes. Details of all of the attitude sensors and their coordinate frames are outside the scope of this document.

Once on-orbit, the orientation of the spacecraft reference frame is estimated in onboard software, which uses the star tracker and gyro data.

The Flight coordinates are actually defined in an onboard flight software table, which specifies the rotation relative to the Spacecraft coordinates that defines the Flight coordinates. It is the Flight coordinates that are nominally kept closely aligned to local vertical and horizontal.

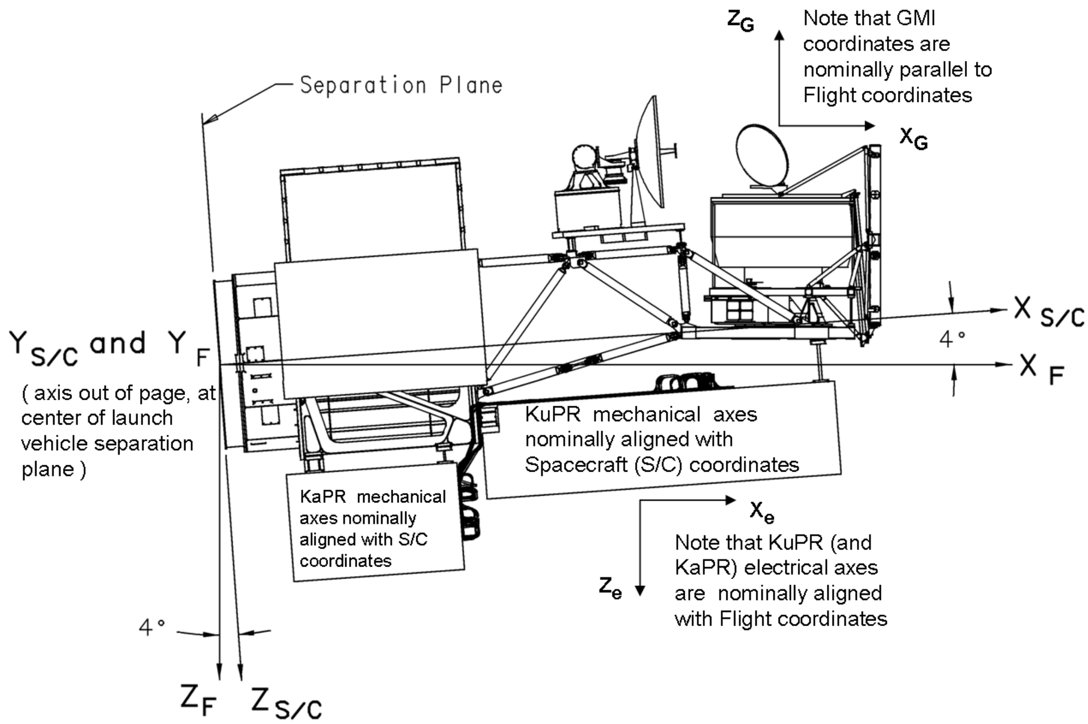

Figure 1-1. GPM Spacecraft-Fixed Coordinate Frames

In the normal mission science mode, GPM is targeted to hold a fixed orientation relative to the local horizontal and geodetic nadir. Nominally the Flight axes are kept level with the local horizontal, although any offset pointing can be targeted. This nominal flight orientation is required so that the Dual-Frequency Precipitation Radars—the Ka-Band Precipitation Radar (KaPR) and the Ku-Band Precipitation Radar (KuPR)—and the GPM Microwave Imager instruments operate with their proper view of the Earth. In the spacecraft design phase, the Spacecraft axes were established in a different orientation than the Flight axes, aligned with the DPR mechanical axes, because of preferences asserted by thermal and structural engineers having to do with integration and testing procedures.

With this approach, the main spacecraft alignment cube and most of the component alignment cubes are oriented along the Spacecraft axes, and in fact the GPM Attitude Control System (ACS) Team has adopted the Spacecraft axes for reference in most of their internal calculations.

Also, the thrusters were designed to operate along the Spacecraft axes, with designers citing slightly better clearances and center of mass, so that the spacecraft will switch to holding the Spacecraft axes along the local horizontal and very close to the nominal velocity direction whenever the thrusters are fired for orbit adjust maneuvers. Thus from the ACS perspective controlling the S/C axes, the normal flight orientation is "pitched up 4 degrees." However, from the science perspective the nominal attitude is defined as near-zero pitch, roll, and yaw with the Flight axes aligned to the geodetic nadir and flight direction. Relative to this attitude, which is the pitch-roll-yaw geodetic attitude reported in the science products, the spacecraft is "pitched down 4 degrees" for the orbit adjust thruster firings before returning to near-zero pitch for science data collection. Hopefully, confusion between these different definitions of the attitude can be avoided throughout the mission by user clarity in their reference frame context. In this document, and in the Level-1 (L1) science products, the orientations in pitch, roll, and yaw will be described relative to the Flight axes, as discussed further in the next subsections.

While the Flight axes are defined as fixed relative to the spacecraft structure, the attitude of the spacecraft is defined by the rotation of the Flight axes relative to a reference frame in space, which is described below.

1.4 GEODETIC REFERENCE FRAME (GRF)

The geodetic reference frame (GRF) is a spacecraft-centered set of reference axes relative to the Earth's geodetic nadir and the spacecraft velocity direction. It is defined as follows in inertial space, given the spacecraft velocity vector, $\mathbf{V}^{\mathbf{I}}$, and the nadir direction vector, $\mathbf{Nd}^{\mathbf{I}}$, in inertial coordinates (where the superscript "I" is used to indicate that the directions are indicated in an inertial frame):

+$\mathbf{Z}^{\mathbf{I}}$ axis is toward the geodetic nadir, $\mathbf{Nd}^{\mathbf{I}}$. (1-1a)

+$\mathbf{Y}^{\mathbf{I}}$ axis is given by the cross product of $\mathbf{Z}^{\mathbf{I}}$ and the spacecraft velocity in inertial space, $\mathbf{V}^{\mathbf{I}}$, ($\mathbf{Z}^{\mathbf{I}}$ x $\mathbf{V}^{\mathbf{I}}$). This axis is nearly along the negative orbit normal direction. (1-1b)

+$\mathbf{X}^{\mathbf{I}}$ axis is given by the cross product of $\mathbf{Y}^{\mathbf{I}}$ and $\mathbf{Z}^{\mathbf{I}}$, ($\mathbf{Y}^{\mathbf{I}}$ x $\mathbf{Z}^{\mathbf{I}}$). This axis is nearly along the spacecraft velocity vector, $\mathbf{V}^{\mathbf{I}}$. (1-1c)

Note that each axis in this orthogonal triad is defined as a unit vector. The geodetic attitude for GPM is defined by rotations from the GRF coordinates to the body Flight axes. More specifically, geodetic yaw, pitch, and roll are defined by a 3-2-1 rotation sequence, respectively.

The construction of the geodetic reference frame using the nadir direction $\mathbf{Nd}^{\mathbf{E}}$ and velocity $\mathbf{V}^{\mathbf{E}}$ in Earth-centered, Earth-fixed (ECEF) coordinates is discussed later (where the superscript "E" is used to indicate that the directions are indicated in ECEF coordinates).

The difference between geodetic and geocentric nadir is illustrated in Figure 1-2, which shows a very exaggerated view of the Earth's oblateness. The Earth shape used in the definition of geodetic nadir is an ellipsoid, and the World Geodetic System-84 (WGS-84) reference ellipsoid is the shape used for GPM and most current Earth geodesy applications. The deviation of mean sea level (which follows the geoid, or gravitational geopotential shape) from the WGS-84 ellipsoid is generally less than 100 meters (Wertz, 1978).

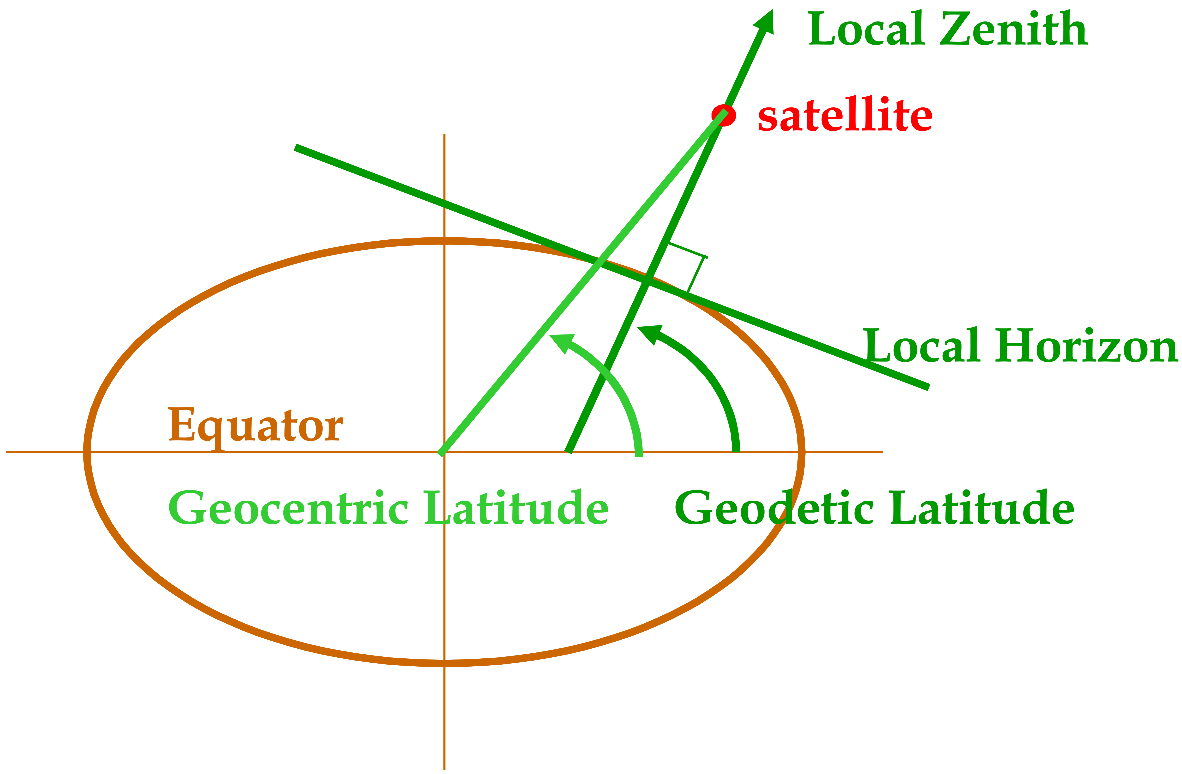

Figure 1-2. Difference Between Geocentric and Geodetic Nadir Directions

The geodetic nadir is a natural pointing orientation for Earth-viewing missions because it is perpendicular to the local horizon and the surface of the Earth. The angular difference between geodetic and geocentric nadir reaches up to about 0.2 degree, with a maximum around 45 degrees latitude, as shown in Figure 1-3.

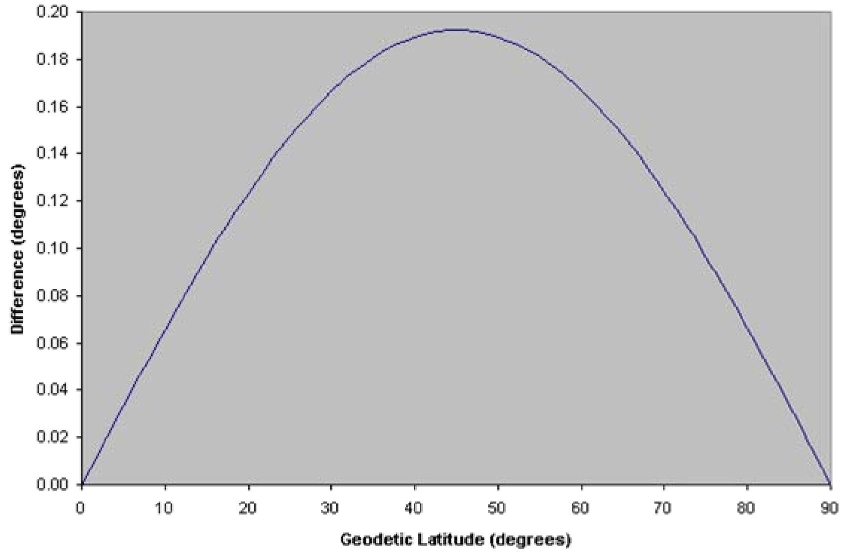

Figure 1-3. Difference Between Geodetic and Geocentric Latitude at the Earth's Surface

1.5 EARTH-CENTERED, EARTH-FIXED (ECEF) COORDINATES

Earth-centered, Earth-fixed coordinates represent X, Y, Z Cartesian coordinates with origin at the Earth center of mass, and fixed relative to Earth latitude-longitude locations. The Z axis is aligned with the International Reference Pole, and the X axis is along the International Reference Meridian, which passes through Greenwich, England.

1.6 GEOCENTRIC INERTIAL COORDINATES (GCI)

The geocentric inertial coordinates (GCI), also known as Earth-centered inertial (ECI), represent X, Y, Z Cartesian coordinates with origin at the Earth center of mass, the Z axis along the Earth spin axis, and X axis along the Vernal Equinox. Unless otherwise specified, GCI coordinates discussed in this document will represent "true of date" coordinates, which use the current Earth spin axis and Vernal Equinox direction. It is also common to refer to an epoch date for the inertial coordinates, with "J2000" representing the inertial coordinates for the year 2000 epoch time.

2.0 INSTRUMENT MODEL INPUTS

For each pixel along an instrument scan, the timing of observation and the vector direction viewed in Instrument coordinates (center of the effective field-of-view [CFOV]) are inputs for the GeoTK calculations, and must be generated by the Level 1 software using some telemetry and a model of the sensor observations.

A sample of one approach to calculating these inputs per pixel is according to a simple conical scan path model as described below. This approach was taken for the Tropical Rainfall Measuring Mission (TRMM) Microwave Imager (TMI) and Visible and Infrared Sensor (VIRS) instruments, and could be applied as well for KaPR and KuPR. A summary of parameters used to apply this approach for each of the TRMM rain sensors is shown in Table 2-1.

Table 2-1. Conical Scan Model for TRMM Rain Sensors

Instrument	Axis of Rotation (TRMM S/C Coordinates)	Reference Axis for Zero Phase of Rotation (TRMM S/C Coordinates)	θ Scan Cone Angular Radius	φ Range of Rotation Phase
TMI	(0,0,1) = Z-axis (nadir)	(1,0,0) = +X-axis	49°	-64.4024° to +65.0967° in 208 equal steps
VIRS	(1,0,0) = +X-axis	(0,0,1) = Z-axis (nadir)	90°	-44.86° to +44.87° in 261 equal steps
PR *	(cos 4°, 0, -sin 4°) ~= X-axis pitched up 4 degrees	(0,0,1) = Z-axis (nadir)	94°*	-17° to +17° * in 49 equal steps

*Approximate Precipitation Radar (PR) model; vectors for PR were defined by a table supplied by JAXA.

This approach defined fixed rotation phase angles for each observation in Instrument coordinates, and an instrument alignment matrix was applied to rotate to Spacecraft coordinates. Also, the TRMM instrument approach defined a timing for each pixel, which was fixed relative to the telemetry packet time tags.

2.1 GMI MODEL

A variation on this model is planned for computing the GMI instrument viewing directions, which includes parameters to describe the timing of each observation allowing for variations in the spin rate. The instrument is assumed to rotate at a uniform rate between tachometer data points in the telemetry that are sampled 32 points in each spin, or alternatively a uniform rate is assumed per spin period. The GMI instrument rotates about the instrument +Z axis, which nominally points toward the zenith (opposite the spacecraft +Z axis). The reference axis from which the phase angle of rotation is defined will be the instrument +X axis. The parameters input to this model, derived from spacecraft telemetry, are defined as follows.

Tref = Reference time for start of science sampling for first pixel in scan.
Aref = Angle of rotation when science sampling starts at first pixel in scan.
Tstep = Time between pixel samples (nominally 3.6 milliseconds).
Wrate = Angular spin rate (nominally 32 rpm).
θ = Scan cone angular radius.

The scan cone angular radius, θ, could be defined separately for each channel (i.e., each frequency and separately for horizontal and vertical polarization). In reality each channel, frequency, and polarization will have slightly different effective values for the scan cone angles θ and slightly different starting rotation times Tref and angles Aref. As a practical matter, it is currently expected that the differences will be negligible within each of two groups of frequencies. Channels 1 through 9 all have nominal scan cone angles of 131.5 degrees (180.0 – 48.5), while channels 10 through 13 all have nominal scan cone angles of 134.64 degrees (180.0 – 45.36). Also, nominally the relative timing of the sampling in each of the different channels is adjusted so that Aref is practically the same in all the channels.

The time that each pixel, i, is sampled (starting from index 0) is calculated as:

$$t_i = \text{Tref} + i * \text{Tstep} \qquad (2\text{-}1)$$

The rotation angles, phi, φ_i, at which each pixel, i, are sampled are:

$$\varphi_i = \text{Aref} + \text{Wrate} * (i * \text{Tstep}) \qquad (2\text{-}2)$$

The viewing direction vector is given by:

$$\text{CFOV direction vector, } \mathbf{C}_i = \begin{bmatrix} \sin(\theta)\cos(\varphi_i) \\ \sin(\theta)\sin(\varphi_i) \\ \cos(\theta) \end{bmatrix} \qquad (2\text{-}3)$$

where CFOV is used to indicate the center of the field-of-view of the instrument viewing.

By using the timing information that is hardcoded in the GMI design for the sampling integration time (3.6 ms) and spin rate to get the rotation phase angles of observation, φ_i, this model would still work in the contingency that the spin rate changes slightly from the nominal speed. (A spin rate change might be used during the mission in the unlikely contingency that flexible modes are excited by the nominal 32 rpm rate.)

2.2 KAPR MODEL

A model for the KaPR scan was provided by JAXA (Furukawa, 2011) in which the beam directions (also called ray directions) are given along a 93.943-degree scan cone from a Ka antenna axis, X_{ka}. This was based on antenna near-field measurements. A scan angle phi (Φ) defines the beam directions relative to the Z_{ka} axis positive toward the Y_{ka} axis. The scan angles of each beam are expected to be provided in tables for the alternating matched scan/high-sensitivity (MS/HS) scans of the instrument. There are 25 beams in the MS scan, which will be aligned to match the central beams from the KuPR radar. There are 24 beams in the HS scan that provide high-sensitivity spatial sampling between the MS scans. Phase shift adjustments can be made on-orbit, which will be expected to rotate the scan angle of each of the beams by the same amount. The alignment data and calculations for geolocation adjustments for the relative alignment of the beams are discussed in Section 7.

2.3 KUPR MODEL

A similar model to the KaPR scan was provided by JAXA for the KuPR scan (Furukawa, 2011). The beam directions are given along a 93.755-degree scan cone from the Ku antenna axis, X_{ku} (Masaki, 2012). There is a single normal scan type, labeled NS, with 49 beam directions similar to the TRMM PR instrument. As for KaPR, the scan angles for each beam direction are expected to be provided in a table, based on near-field measurements. It is the middle 25 beams that are expected to match the beam/ray directions of the KaPR MS scans. Since the ground estimated scan cone angle and scan angles are slightly different, all the beams cannot match exactly, but the error from these model differences is negligible compared to other uncertainties. The aim will be to make the nominal nadir-direction beams match as closely as possible.

3.0 ATTITUDE INPUTS

The attitude is provided by the onboard ACS flight software as geodetic pitch, roll, and yaw as specified by GPM Core Observatory Performance Requirements (Level 3). These are offset angles for the Flight coordinates relative to the geodetic reference frame (GRF) defined earlier in Section 1.3. A 3-2-1 Euler sequence for rotations about Z, Y, and X axes for yaw, pitch, and roll, respectively, is used. The GPM attitude in normal mission mode holds zero values of pitch and roll, and a target of 0 or 180 degrees in yaw with respect to the GRF.

3.1 ONBOARD-CALCULATED PITCH, ROLL, AND YAW ATTITUDE HISTORY FILE

Input of the attitude history will be through a file that gives pitch, roll, and yaw values at 0.1 second intervals. Nominally these values will be obtained from spacecraft ancillary 10 Hz telemetry; however, options will be supported to generate simulated values and recalculated, adjusted, or repaired ground estimated values. For example, a simple nominal attitude file for simulation can hold constant zero values for all three axes.

It is expected that linear interpolation can be used to calculate the attitude at any particular time in normal mission mode. Because of the high 10 Hz data rate and limited controller bandwidth onboard, simple linear interpolation should work. Only if there are unusually high levels of spacecraft jitter will this assumption need to be revisited, and this would be an unusual contingency case that would have to be addressed after launch.

This document does not include details of the attitude file reading algorithms. The software design for this algorithm is presented in the PPS GeoBuilder Architecture and Design Specification Document (Hensley, 2011).

3.1.1 Alternate Attitude Representations

There are additional representations used onboard and available in telemetry for specifying the spacecraft orientation. In particular, rotations are often represented by quaternions in spacecraft support, because of their efficiency for calculations and because they avoid singularities that occur for Euler angle representations like pitch, roll, and yaw. Quaternions and other attitude representations are described in Wertz, 1979.

For the star tracker, the attitude is represented as a quaternion giving the rotation from inertial J-2000 coordinates to the tracker's alignment-cube-defined coordinates. The transformation for the tracker coordinates to the Attitude Control System coordinates is provided in the ACS coordinate system document, and refinements might be estimated during the in-orbit checkout mission phase. The onboard ephemeris from the Global Positioning System (GPS) is processed onboard to create a target quaternion, which represents the desired attitude with respect to either a geodetic offset target or an inertial target. The onboard estimated rotation from the target is computed as an error quaternion, which is used for input to the control law for 3-axis control.

For vector processing, as is done for the geolocation calculations herein, it is convenient to represent attitude as a rotation matrix, an orthonormal 3x3 array that represents the projections of one coordinate system's axes onto another coordinate system. The computation of this attitude matrix is discussed later.

3.2 KEY ACS MODE DATA

Some key ancillary data are extracted from the ACS 10 Hz data. Two key ACS mode indicators must be extracted from the ancillary telemetry and will be stored per scan line in the GPM 1B and higher products so that special modes of operation are distinguished from the normal mission configuration. Also, two key status flags about the ephemeris status are extracted. The definitions of these data are described in the following sections.

3.2.1 ACS Control Mode

The ACS control mode gives the basic operating mode for the spacecraft control. There are eight possible values for the control mode:

0 = LAUNCH – Used before injection, and before active control.
1 = RATENULL – Used to stop tumbling after injection or anomaly.
2 = SUNPOINT – Safehold using a minimal sensor set.
3 = GSPM (gyro-less Sun point) – A safehold mode with gyro failure.
4 = MSM (mission science mode) – For normal mission science and calibrations.
5 = SLEW – For a re-orientation maneuver from one target to another.
6 = DELTAH – For contingency momentum dumping, with thrusters.
7 = DELTAV – For a change in velocity, i.e., orbit adjust, with thrusters.

The above control modes are discussed in the GPM Core Observatory Performance Specifications.

Note that mode 4, mission science mode, is the normal mode for science data collection, but the spacecraft can also be in this mode for non-nominal operations. It can be used for various offset pointing for special calibration orientations, or for special inertial pointing orientation. To distinguish that the spacecraft is in a normal science pointing orientation, the following data are also needed.

3.2.2 ACS Target Orientation

The ACS target word, which is downlinked in telemetry, indicates an index into a table of possible target orientations. By agreement between PPS, the Flight Operations Team, ACS, and flight software representatives, it is currently understood that key parts of this table will stay fixed in terms of the specific meaning, so that this word can be used as an indicator when the spacecraft is in normal pointing for operational data collection (Bilanow, 2011).

The actual target values for normal pointing can be adjusted slightly. For example, the target orientation can be changed to slightly adjust the GMI instrument to point its Z axis more exactly at nadir if an undesirable offset is identified.

Details of the configuration management of these tables, and how PPS will be informed of the values, remain to be clarified as of this writing.

Note that the spacecraft ACS control mode will switch to slew mode to change orientation whenever the target attitude is changed. The following provides the currently established numbering for a set of "canned" target orientations of particular interest for normal operations.

0 = S/C Z axis nadir, +X in flight direction.
1 = Flight Z axis nadir, +X in flight direction.
2 = S/C Z axis nadir, -X in flight direction.
3 = Flight Z axis nadir, -X in flight direction.
4 = Flight Z axis nadir, +90 yaw for DPR antenna pattern calibration.
5 = Flight Z axis nadir, -90 yaw for DPR antenna pattern calibration.

The above targets represent fixed offsets from the basic geodetic nadir pointing mode. Up to 12 standard target values can be stored, and in addition new offsets can be loaded by ground command. There is the further option to target fixed orientations in inertial space.

Both the ACS mode and target flag values must be used to distinguish that the spacecraft is in a normal mission pointing orientation, as discussed in Sections 6.4.1 and 6.4.2. The ACS control mode 4, mission science mode, will also be used at some non-nominal orientations.

3.3 NON-SCIENCE MODE EVENTS DURING NORMAL OPERATIONS

There are two important types of events that will occur regularly throughout the mission in which the spacecraft will not be oriented for normal data collection: Yaw turns, and delta-V maneuvers.

3.3.1 Yaw Turns

Like TRMM, the GPM spacecraft is designed to collect data while flying either "forward" (i.e., +X Flight axis along the velocity direction, yaw = 0) or "backward" (i.e., -X Flight axis along the velocity direction, yaw = 180 degrees). The orientation is changed about every 40 days in order to keep the +Y side of the spacecraft in shadow, and the –Y side in sunlight as the orbit precesses. This is done for the purpose of thermal control. For yaw turns, the slew periods are expected to be about 10 minutes. (Note that for GPM there is not a specific "yaw turn" mode as was specified for the TRMM spacecraft. For GPM, a yaw turn is just a slew mode event that goes between one standard yaw target and another.)

3.3.2 Delta-V Maneuvers

The GPM spacecraft will adjust its orbit about once a week to once a month, depending on solar activity, which affects atmospheric drag. There is a special sequence of mode changes that will occur with each orbit adjust.

For a delta-V (orbit adjust), the spacecraft will first slew 4 degrees to an orientation where the Spacecraft axes (rather than "Flight axes") are held level in the geodetic reference frame, i.e., with the instruments pitched down 4 degrees from their normal flight orientation. At this new orientation, the spacecraft will fly in "mission science mode" again for a period of time before entering the actual delta-V mode. It is now expected that for initial orbit operations it will stay in this orientation for about 20 minutes so the new orientation can be verified before the actual thruster firing is commanded. After the thruster firing is done, the spacecraft will slew 4 degrees back to the normal science data collection orientation.

Thus, with an orbit adjust starting from yaw = 0 degree, the following sequence will occur:

ACS_Mode = 4 and Target = 1 (normal yaw 0 science before start of burn sequence).
ACS_Mode = 5 and Target = 0 (slew, pitched down 4 degrees for S/C Z axis at nadir).
ACS_Mode = 4 and Target = 0 (mission mode at new target for a few seconds).
ACS_Mode = 7 and Target = 0 (delta-V for about 30 seconds).
ACS_Mode = 4 and Target = 0 (mission mode again for a few seconds).
ACS_Mode = 5 and Target = 1 (slew, pitched up 4 degrees, returning Flight Z axis to nadir).
ACS_Mode = 4 and Target = 1 (return to normal mission pointing at yaw = 0).

With an orbit adjust starting from yaw = 180 degrees, the following sequence will occur:

ACS_Mode = 4 and Target = 3 (normal yaw 180 science before start of burn sequence).
ACS_Mode = 5 and Target = 2 (slew, pitched down 4 degrees for S/C Z axis at nadir).
ACS_Mode = 4 and Target = 2 (mission mode at new target for a few seconds).
ACS_Mode = 7 and Target = 2 (delta-V for about 30 seconds).
ACS_Mode = 4 and Target = 2 (mission mode again for a few seconds).
ACS_Mode = 5 and Target = 3 (slew, pitched up 4 degrees, returning Flight Z axis to nadir).
ACS_Mode = 4 and Target = 3 (return to normal mission pointing at yaw = 180).

Normal mission science occurs when the ACS mode = 4 and the target = 1 or 3. This normal orientation is also identified for science users by some derived flags, which are discussed in Sections 6.4.1 and 6.4.2.

3.4 EPHEMERIS STALE FLAGS

There are two flags about the status of the ephemeris data that are received with the ACS 10 Hz telemetry packets and that have been specified to be used with the science data processing: The GPS position-velocity-time (PVT) stale flag and the Goddard-Enhanced Onboard Navigation System (GEONS) stale flag. These were generated at the request of the GPM Project based on an action item following a meeting with JAXA.

The current description available for these flags in the flight software telemetry spreadsheets reads as follows:

GEONS_STALE	GPS GEONS data are considered stale for science purposes.
PTSOL_STALE	GPS point solutions data are considered stale for science purposes.

Note that the GPS point solutions is what the PPS GeoBuilder utility calls the PVT ephemeris (Section 4.2).

A description of how the flags are implemented in the ACS flight software has been provided as follows (M. Vess, GPM ACS Lead, 2012, personal communication). The stale flags are set by counting the number of ACS cycles (every 0.1 second) since the last valid point solution or GEONS update, and checking against database-specified values. Currently that database value is 6,000 cycles, or 600 seconds (10 minutes). This value can be changed based on future discussions and science user input. The GEONS updates occur every 5 seconds for the propagated state even though the GEONS solution is only updated every 30 seconds. There are separate thresholds for PVT and GEONS stale checks, but currently they are the same. The ACS software does not use these flags; they are provided for telemetry information based on the request from the project. The use of these flags in derived flags for the GPM science products is discussed in Section 6.4.2.

4.0 EPHEMERIS INPUTS

There are three types of onboard ephemeris from the GPM core observatory, and several types of ground ephemeris that can be generated.

All of the onboard ephemerides are derived from Global Positioning System data. The raw data are obtained onboard via GPS receivers and processed with software referred to as the GPS Navigator (NAV) system. The GPS Navigator is being developed at NASA Goddard Space Flight Center (GSFC) to support both the GPM and the Magnetospheric Multi-Scale Mission (MMS) Projects. Within the NAV flight software, the Core GPS System (CGS) takes the standard GPS approach of computing position and velocity and time, or "PVT point solutions," using pseudo range and range-rate data derived from signals from the GPS constellation of satellites. These are designed to be computed every 5 seconds. These data are then filtered by the Goddard Enhanced Onboard Navigation System (GEONS) software to obtain a more accurate ephemeris, with solutions output every 30 seconds.

4.1 ONBOARD PROPAGATED (OBP) EPHEMERIS

The onboard ACS software obtains the spacecraft position and velocity from the onboard GPS Navigator software, using either the PVT point solutions every 5 seconds, or the GEONS smoothed ephemeris solutions every 30 seconds.

After obtaining the position and velocity from the GPS Navigator, the ACS software propagates the orbit with a simple orbit model in order to obtain the position and velocity every 0.1 second (a 10 Hz rate). The position and velocity are needed to compute the reference frame for the geodetic target attitude, and thus compute the target quaternion for the onboard attitude control. The GPS Navigator accuracy is specified in coordination with the accuracy of the onboard propagator so that adequate pointing control will be maintained with the propagated ephemeris.

PPS documents now refer to this as the onboard propagated (OBP) ephemeris. Nominally every fifth value computed onboard is downlinked in spacecraft telemetry. It is included in the 2 Hz combined telemetry packet. It provides a subsampled record of the vectors used in the onboard processing.

4.2 GPS POSITION-VELOCITY-TIME (PVT) EPHEMERIS

The GPS Navigator PVT point solutions are included in telemetry, and will be captured in PPS ephemeris files. By post-processing the data, PPS does not need to propagate the ephemeris as required onboard, and thus can obtain better ephemeris at any time simply by interpolating and/or smoothing the PVT data. The positions obtained from the Navigator are expected to be accurate in the range of tens of meters; however, the velocities computed are somewhat more noisy than those obtained from a standard definitive ephemeris, so that should be kept in mind in the use of these data.

The GPS Navigator will be able to find the spacecraft position and provide valid PVT within minutes of being turned on after any outage. The NAV PVT solutions are expected to be quite reliable and could meet mission requirements on their own. Some early onboard GPS systems have been subject to glitches such as loss of signal from enough satellites, so many past missions have also included a system to filter and check the GPS data. The onboard ACS system will do some checking of the PVT data before use in the OBP results, and the GEONS system onboard will provide a more complex filter for the NAV PVT solutions to provide a more accurate estimate of the orbit.

4.3 GEONS EPHEMERIS

The GEONS software filters the PVT data using a Kalman filter with an accurate Earth geopotential model and drag parameters to provide more accurate position and velocity estimates. The new solutions are computed every 30 seconds, although the propagated state is output every 5 seconds. This should be the most accurate of the onboard ephemeris sources. However, it may take a longer time to obtain the high accuracy after any outage, and it will take some time to converge to a good orbit solution after delta-V (orbit adjust) maneuvers. Currently it is not clear how much the GPM Project Office will support efforts for on-orbit analysis, calibration, and verification of the GEONS accuracy. It is not presumed to be necessary to meet the minimum mission requirements, and therefore extensive validation currently is not required.

4.4 GROUND EPHEMERIS

Several types of standard ground ephemeris files are available, and in several different formats. Traditionally at GSFC, standard ephemeris products have output data at 60-second intervals. It is expected that will continue with products for GPM. Since a ground model is typically based on a very accurate orbit model fit to adequate data, the outputs are generally free from noise. This will be most accurate over the period of data that is fit, and this is generally referred to as the “definitive arc” of an ephemeris. The model output becomes progressively less accurate as it is used to predict into the future. This is because any errors are gradually amplified and geodynamic variables that change, such as atmospheric drag on the spacecraft, cannot be predicted accurately.

A standard 30-day predicted ephemeris file will be available to PPS. Some 8-hour definitive ephemeris products are now planned for internal GPM Mission Operations Center (MOC) usage with the Attitude Ground System (AGS), but it is not yet known if these will ever be made available to PPS.

The Consultative Committee for Space Data Systems (CCSDS) Orbital Ephemeris Message (OEM) is an ASCII text format for providing ephemeris output, and it is planned to be used as a convenient standard.

4.5 PPS COMBINED EPHEMERIS FORMAT

A format has been chosen for PPS operational use that combines all the onboard data with an OEM-like ground format at 60-second intervals all in one file. The file length will normally cover a span just long enough for production of standard granules of data—typically 1 orbit for standard products.

5.0 GEOLOCATION CALCULATIONS

The main function of geolocation is to compute pixel latitude and longitude, Earth incidence angles, and other ancillary geolocation-related information based on the input orbit and attitude and sensor model data.

A top-level view of the basic geolocation computations is provided as follows. These computations are done for each pixel of each scan. The beam direction vector, for each pixel field-of-view, D_S, is taken in Instrument (Sensor) coordinates, and transformed to the view direction in ECEF coordinates, D_E. The beam direction vector is defined as the effective center of the weighted average antenna pattern. It defines the center of the effective field-of-view (EFOV), including time-averaging, e.g., of DPR pulse repetitions or GMI sample integration. It gives the line-of-sight direction for which geolocation parameters are calculated. This transformation involves three steps enumerated below (with the name of the rotation indicated in parentheses):

1. Instrument to Flight coordinates (sensor alignment, $[S]^T$ rotation).
2. Flight coordinates to GRF coordinates (attitude, $[A]^T$ rotation).
3. GRF coordinates to ECEF coordinates (new reference frame, [N] rotation).

This is represented mathematically by a sequence of matrix rotations:

$$D_E = [N]\,[A]^T\,[S]^T\,D_S \tag{5-1}$$

where the direction vectors, D, are column vectors; the brackets [ ] indicate 3x3 matrices; and the superscript T represents the transpose of matrix. The orthogonal 3x3 matrices, [N], [A], and [S], which represent rotations, are described in detail in subsequent sections. They are applied to the vector cumulatively from right to left.

This view direction in ECEF is then intersected with the model of the oblate Earth and the latitude and longitude of intersection location is determined. The local Vertical, North, and East coordinate directions at this pixel location are calculated, from which the incidence angles and satellite azimuth can be calculated. Also, the solar zenith and azimuth and a Sun glint angle are computed in the coordinates of this pixel location. Details are provided in the following subsections.

5.1 INITIALIZATION COMPUTATION; ALIGNMENT MATRIX

Certain computations need to be done only once, and not for every pixel of every scan. These are summarized in this subsection. In particular, alignment information input as Euler angles is used to construct the sensor rotation matrix [S] for the transformation from Flight coordinates to Instrument (or Sensor) coordinates. The inputs for this computation are summarized as follows (and these input data are stored with product metadata for reference). Some details about how the alignment information is obtained and may be modified is discussed in Section 7.

For maximum flexibility, a general utility subroutine is used that will handle the alignment input in any of the 12 possible Euler rotation sequences: 1-2-3, 2-3-1, 3-1-2, 1-3-2, 3-2-1, 2-1-3, 1-2-1, 1-3-1, 2-1-2, 3-1-3, 2-3-2, or 3-2-3. This allows the convenience of specifying a sequence that makes sense for any particular sensor. This subroutine provides the rotation matrix from the aforementioned Flight coordinates to the Instrument coordinates. The computation is based on applying the rotations about the axes specified in the order given.

For example, the KaPR beam direction vector inputs will be expected in the Instrument mechanical coordinates, which are nominally tilted about 4 degrees up in pitch (about Y) relative to the Flight coordinates as defined for GPM. (See Section 7 for discussion of the Instrument coordinates.) Since it is often useful to apply the largest rotation first, the following nominal 2-1-3 sequence has been proposed for the KaPR alignment inputs:

Alignment angle - First rotation (deg)	:4.0
Alignment angle - Second rotation (deg)	:0.0
Alignment angle - Third rotation (deg)	:0.0
Euler rotation sequence - First rotation axis	:2
Euler rotation sequence - Second rotation axis	:1
Euler rotation sequence - Third rotation axis	:3

For a 2-1-3 sequence with a first rotation angle phi, Φ, about **Y**, a second rotation angle theta, Θ, about the new **X**, and a third rotation angle psi, Ψ, about the new **Z**, the matrix [S] for the rotation of a vector from Flight coordinates to Instrument coordinates is computed as:

$$[S] = \begin{bmatrix} c\Psi c\Phi + s\Psi s\Theta s\Phi & s\Psi c\Theta & c\Psi s\Phi + s\Psi s\Theta c\Phi \\ -s\Psi c\Phi + c\Psi s\Theta s\Phi & c\Psi c\Theta & s\Psi s\Phi + c\Psi s\Theta c\Phi \\ c\Theta s\Phi & -s\Theta & c\Theta c\Phi \end{bmatrix} \qquad (5\text{-}2)$$

where c and s represent the cosine and sine functions, cos( ) and sin( ) respectively, applied to the indicated angles. The formulas for this matrix for any of the 12 possible Euler sequences are defined in Wertz, 1978, Appendix E, page 764.

5.2 PIXEL GEOLOCATION DIRECT REFERENCING

This section describes the computation of the geolocation, i.e., latitude and longitude, of a pixel location on the Earth given the beam direction in Instrument/Sensor coordinates. The term “direct referencing” refers to the process of calculating the location given input of the view direction, while the term “inverse referencing” refers to calculating the Instrument data coordinates, e.g., scan line and pixel number, given a location on the Earth. An inverse reference application, such as developed for TRMM, is not currently planned for GPM. Inverse referencing could be computed, for example, by linear interpolation within a scene using the direct-referenced latitude and longitude values once the neighboring pixels are located, but the details of this technique are not a topic discussed in this document.

The direct referencing calculation described here is executed for an entire granule for every scan pixel by pixel for the imaging sensors. This process includes obtaining the spacecraft ephemeris and spacecraft attitude data.

Since the GPM direct referencing method uses a basic Earth ellipsoid shape model for geolocation, it does NOT include effects of topography or cloud heights in computing the exact geodetic latitude and longitude of observed locations. These effects must be considered by the data users for cases in which they might be important in viewing cloud tops or mountain tops at high slant paths—but most of the time very little error is introduced by this simplification.

The relationship between the spacecraft position vector from Earth's center to the spacecraft (P), the look direction vector from the sensor to the pixel (D), and the target vector from Earth center to the pixel (G) on an oblate Earth is shown in Figure 5-1.

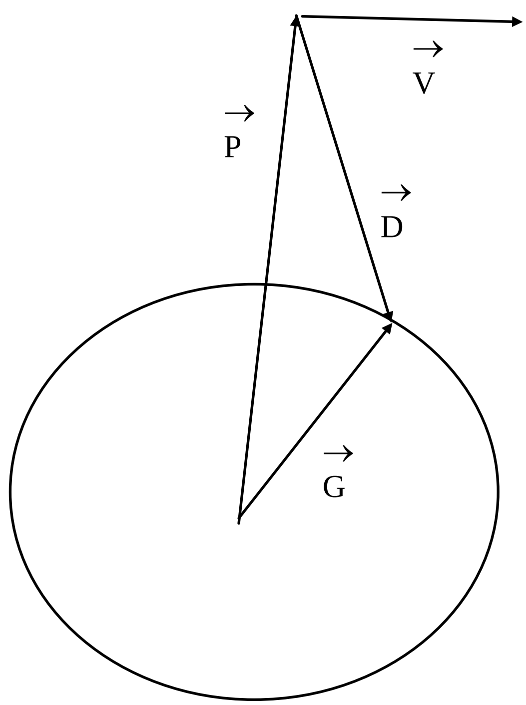

Figure 5-1. Relationships Between Vectors Involved in the Geolocation Calculations

Calculations done for each pixel are outlined as follows:

- Get the spacecraft position and velocity.
- Compute the geodetic nadir vector.
- Compute the geodetic reference frame.
- Get the attitude angles and calculate the attitude rotation matrix.
- Calculate the rotations from sensor to Earth coordinates.
- Calculate the center of the field-of-view (CFOV) unit vector.

- Intersect the CFOV direction with the Earth WGS-84 ellipsoid model.
- Derive geodetic latitude and longitude at the intersection point.
- Compute East, North, and Up coordinates at the pixel location.
- Compute zenith (incidence) and azimuth angles to the spacecraft.
- Get the Sun direction in inertial coordinates and convert to ECEF coordinates.
- Compute Sun zenith and azimuth angles and Sun glint angle.

These steps are described in more detail in the following sections.

5.2.1 Interpolate Attitude to Requested Time and Compute Attitude Rotation Matrix

The attitude at the specified time is obtained from the attitude file as described in Section 3.1. If attitude data are missing, the results will be flagged and fill data will be inserted.

Linear interpolation will be applied to the input data. This is calculated as follows. Given the attitude Euler angles, a_1 and a_2, (representing pitch, roll, or yaw) for the times, t_1 and t_2, closest to the time, t, of interest, the attitude at time t, a_t, is calculated as:

$$a_t = a_1 + (t - t_1)\,(a_2 - a_1)\,/\,(t_2 - t_1) \tag{5-3}$$

The attitude angles are defined (and computed onboard the spacecraft) as a 3-2-1 Euler rotation sequence, providing information for the transformation from the geodetic reference frame (see below) to the Flight axes. This sequence means that a rotation is applied about Z, then Y, and then X.

The formula for the attitude matrix [A] (for the rotation of a vector from GRF coordinates to Flight coordinates) for the 3-2-1 sequence, where the first rotation angle phi, Φ, is about **Z**, the second rotation angle theta, Θ, is about the new **Y**, and the third rotation angle psi, Ψ, is about the new **X**, is given by (Wertz, 1978, pg 764):

$$[A] = \begin{bmatrix} c\Theta c\Phi & c\Theta s\Phi & -s\Theta \\ -c\Psi s\Phi + s\Psi s\Theta c\Phi & c\Psi c\Phi + s\Psi s\Theta s\Phi & s\Psi c\Theta \\ s\Psi s\Phi + c\Psi s\Theta c\Phi & -s\Psi c\Phi + c\Psi s\Theta s\Phi & c\Psi c\Theta \end{bmatrix} \tag{5-4}$$

where c and s represent the cosine and sine functions, cos( ) and sin( ) respectively, applied to the indicated angles.

5.2.2 Calculate Instrument Attitude Euler Angles

The orientation of the Instrument coordinates relative to the geodetic reference frame is defined as the "instrument attitude." This incorporates the alignment information, including any adjustments that might be made after launch, to specify the orientation of the Instrument axes relative to the geodetic coordinates. The transformation matrix from the geodetic GRF coordinates to the Instrument coordinates is given by:

$$\mathbf{[M] = [S]\,[A]} \tag{5-5}$$

Assuming a 3-2-1 Euler sequence definition for the attitude, the Euler angles InstrYaw, $\mathbf{\Phi_{IY}}$, InstrPitch, $\mathbf{\Theta_{IP}}$, and InstrRoll, $\mathbf{\Psi_{IR}}$, representing yaw, pitch, and roll respectively, are calculated as:

$$\text{InstrYaw} = \mathbf{\Phi_{IY}} = \text{atan2}(\,\mathbf{[M]_{12}}, \mathbf{[M]_{11}}\,) \tag{5-6a}$$
$$\text{InstrPitch} = \mathbf{\Theta_{IP}} = \text{asin}(\,\mathbf{-[M]_{13}}\,) \tag{5-6b}$$
$$\text{InstrRoll} = \mathbf{\Psi_{IR}} = \text{atan2}(\,\mathbf{[M]_{23}}, \mathbf{[M]_{33}}\,) \tag{5-6c}$$

where the FORTRAN or C “atan2” function is used to define the “arctan” function over all four quadrants based on the sign of the two input arguments. The subscripts on the matrix $\mathbf{[M]_{rc}}$ give the elements in the specified row, **r**, and column, **c**.

5.2.3 Interpolate Spacecraft Position and Velocity in ECEF

The position and velocity vectors of a spacecraft at a specified time are obtained by reading the selected ephemeris source as discussed in the previous section. A two-point cubic interpolation is used, fitting the position and velocity at the two ephemeris sample points surrounding the specified time. Using adjacent position and velocity vectors ($\vec{P}_1$ and $\vec{V}_1$ at time t_1, $\vec{P}_2$ and $\vec{V}_2$ at time t_2) ($t_1 < t_2$), compute the cubic polynomial coefficients as follows:

$$\vec{c}_0 = \vec{P}_1 \tag{5-7a}$$
$$\vec{c}_1 = \vec{V}_1 * \Delta t \tag{5-7b}$$
$$\vec{c}_2 = 3 * \vec{P}_2 - 3 * \vec{P}_1 - 2 * \vec{V}_1 * \Delta t - \vec{V}_2 * \Delta t \tag{5-7c}$$
$$\vec{c}_3 = 2 * \vec{P}_1 - 2 * \vec{P}_2 + \vec{V}_1 * \Delta t + \vec{V}_2 * \Delta t \tag{5-7d}$$

where $\Delta t = t_2 - t_1$.

Then the interpolated position and velocity vectors are computed as follows:

$$\vec{P}_s = \vec{c}_0 + \vec{c}_1 * t + \vec{c}_2 * t^2 + \vec{c}_3 * t^3 \tag{5-8a}$$

$$\vec{V}_s = (\vec{c}_1 + 2 * \vec{c}_2 * t + 3 * \vec{c}_3 * t^2) / \Delta t \tag{5-8b}$$

where $\vec{P}_s$ and $\vec{V}_s$ are the position and velocity vectors at sample time t_s ($t_1 \le t_s \le t_2$), and where $t = (t_s - t_1) / \Delta t$ gives the fractional time from t_1 to t_2 for which t_s is the desired sample time.

5.2.4 Compute the Geodetic Nadir Direction

The geodetic nadir will be calculated optionally according to two methods for analysis and verification purposes. Investigation shows that the differences in the results are very small and are insignificant relative to other error sources.

This is illustrated in Figure 5-2, where the maximum angular difference in nadir pointing direction is less than an arcsecond. The difference in terms of calculated latitude for the nadir point is less than 20 meters in distance.

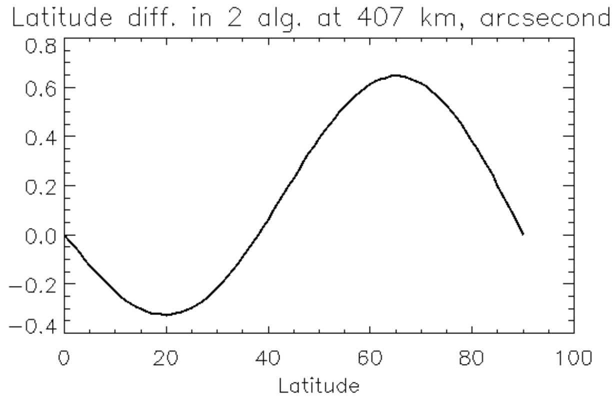

Figure 5-2. Differences in Latitude of Geodetic Nadir Calculated from Two Methods

5.2.4.1 Patt Method

The first method applied was adopted from TRMM data processing. It is the method developed by Patt and Gregg, 1994, which was initially used for the Sea-viewing Wide field-of-view Sensor (SeaWiFS)/OrbView-2 data processing. It includes an approximation, but was deemed adequate for SeaWiFS and TRMM applications.

5.2.4.2 Zhu Method

An alternative algorithm for this calculation published in a review article by Zhu, 1994, and originally developed by Heikkinen. The accuracy is estimated to be on the order of nanometers. This approach was chosen by the GPM ACS Team for implementation on the onboard code for GPM. This code was obtained and adopted for PPS processing also. This method will nominally be used in the ground computations as well for consistency.

5.2.5 Construct the Geodetic Reference Frame (GRF) in ECEF Coordinates

The geodetic reference frame (GRF) is constructed using the nadir direction, $\mathbf{Nd^E}$, and the spacecraft position, $\mathbf{P^E}$, velocity, $\mathbf{V^E}$, in ECEF coordinates as follows. Essentially this is done by constructing each of the axes of the GRF frame ($\mathbf{X^{GE}}$, $\mathbf{Y^{GE}}$, $\mathbf{Z^{GE}}$) as defined in ECEF coordinates. The superscript $^{\mathbf{GE}}$ is used here to indicate that the GRF coordinate basis vectors are defined in the ECEF coordinates.

Although ECEF is formally defined as "Earth centered" the GRF is "spacecraft centered." The ECEF coordinates used here to define the GRF axes are considered as parallel to the ECEF directions, but centered at the spacecraft. The $\mathbf{Z^{GE}}$ unit vector is defined as along the geodetic nadir, $\mathbf{Nd^E}$, whose calculation is discussed in Section 5.2.4.

Since ECEF is a rotating frame relative to inertial space and the GRF is an inertial frame, the velocity vector, $\mathbf{V}^{\mathbf{E}}$, in ECEF must be adjusted for the Earth sidereal rotation rate (Patt, 1994) to provide an effective velocity relative to inertial coordinates, V′. Let the Earth sidereal rate be given by omega, $\mathbf{\Omega_E}$, in radians per second. The components of V′ are given by:

$$V'_x = \mathbf{V}^{\mathbf{E}}_x - \mathbf{\Omega_E}\ \mathbf{P}^{\mathbf{E}}_y \quad (5\text{-}9a)$$
$$V'_y = \mathbf{V}^{\mathbf{E}}_y + \mathbf{\Omega_E}\ \mathbf{P}^{\mathbf{E}}_x \quad (5\text{-}9b)$$
$$V'_z = \mathbf{V}^{\mathbf{E}}_z \quad (5\text{-}9c)$$

The normalized cross product of $\mathbf{Z}^{\mathbf{GE}}$ and the adjusted velocity V′ give the $Y^{\mathbf{GE}}$ direction.

$$Y^{\mathbf{GE}} = \mathbf{Z}^{\mathbf{GE}} \times V' \ / \ (\ \| \ \mathbf{Z}^{\mathbf{GE}} \times V' \ \| \ ) \quad (5\text{-}10)$$

The cross product of $Y^{\mathbf{GE}}$ and $Z^{\mathbf{GE}}$ gives $X^{\mathbf{GE}}$.

$$X^{\mathbf{GE}} = Y^{\mathbf{GE}} \times Z^{\mathbf{GE}} \quad (5\text{-}11)$$

Since $Y^{\mathbf{GE}}$ and $Z^{\mathbf{GE}}$ are already normalized and perpendicular, $X^{\mathbf{GE}}$ is also a unit vector.

These vectors, $X^{\mathbf{GE}}$, $Y^{\mathbf{GE}}$, and $Z^{\mathbf{GE}}$, are combined to make the GRF to ECEF rotation matrix, [**N**], as follows:

$$[N] = \begin{bmatrix} X_x^{GE} & Y_x^{GE} & Z_x^{GE} \\ X_y^{GE} & Y_y^{GE} & Z_y^{GE} \\ X_z^{GE} & Y_z^{GE} & Z_z^{GE} \end{bmatrix} \quad (5\text{-}12)$$

where the components of the vectors are indicated by the subscripts.

The inverse rotation, ECEF to GRF, is given by the transpose $[N]^T$. It can be seen by inspection that $[N]^T$ applied to a vector in ECEF gives the dot product, or projection, of the ECEF vector onto each of the ECEF coordinate constructions of the GRF axes, thus giving the vector components in the GRF frame.

5.2.6 Rotate the Instrument View Direction to ECEF

Starting with the direct line of sight in Instrument/Sensor coordinates, D_S, the rotation to ECEF can be done in three steps as described at the beginning of Section 5, using equation 5-1. First the transpose S^T of the sensor alignment matrix S rotates the view in Instrument coordinates to GPM Flight coordinates. Second, the transpose A^T of the attitude matrix A provides the rotation from Flight coordinates to the GRF (geodetic reference frame) coordinates. Finally, the GRF to ECEF rotation matrix N provides the transformation to ECEF. This gets the instrument view direction in spacecraft-centered ECEF coordinates, $\mathbf{D_E}$. The line-of-sight (or ray path) in this direction can then be intersected with the Earth.

Note that the attitude matrix definition here uses the convention (Wertz, 1978) whereby the rotation direction gives "reference frame to body frame." In this case, the reference frame is given by the GRF (geodetic) coordinates and the body frame is the GPM Flight coordinate frame (which is defined as fixed relative to the spacecraft body). The sensor alignment matrix is defined (using Euler angle inputs) using an analogous convention where "reference to body" in this case means the rotation from the Flight coordinates (a "reference" coordinate relative to the spacecraft structure) to the instrument "body frame." As for all rotation matrices, which are orthogonal, the matrix transpose provides the inverse transformation, so rotations in the opposite direction are readily computed.

5.2.7 Intersect With Oblate/Ellipsoid Earth Model

The intersection of the instrument beam direction, D_E, with the Earth ellipsoid model is calculated as described below. The equation for the ellipsoidal Earth surface in ECEF x, y, z coordinates can be written as:

$$x^2 + y^2 + z^2 / (1-f)^2 = r^2 \qquad (5\text{-}13)$$

where

f	=	flattening factor = $1 - (r_p / r)$
r	=	Earth's equatorial radius
r_p	=	Earth's polar radius

Referring to Figure 5-1, we define the spacecraft position in ECEF, **P**, a line of sight to a point on the Earth, **D**, and the vector from Earth center to the view vector intersection, **G**.

$$\mathbf{G} = \mathbf{P} + \mathbf{D} \qquad (5\text{-}14a)$$

Let **E** be the vector from the satellite to the Earth Center, which is the negative of **P**. Our formulation (from the TRMM heritage implementation) uses **E**, giving the **G** as:

$$\mathbf{G} = \mathbf{D} - \mathbf{E} \qquad (5\text{-}14b)$$

We express **D** as the unit direction vector $\mathbf{D_E}$ times a scalar distance d_X, i.e., $\mathbf{D} = d_X \mathbf{D_E}$. We want to solve for d_X where the line-of-sight intersects the ellipsoid, i.e. where the following vector equation gives x, y, z positions on the ellipsoid:

$$\mathbf{G} = \begin{bmatrix} x \\ y \\ z \end{bmatrix} = \begin{bmatrix} d_X D_{EX} - E_X \\ d_X D_{Ey} - E_y \\ d_X D_{EZ} - E_Z \end{bmatrix} \qquad (5\text{-}14c)$$

By substituting each component of equation (5-14c) into equation (5-13) and regrouping terms, we get a quadratic equation for d_X.

By solving the quadratic equation we get the distance to the intersection points between the line and the ellipsoid. The solution for the quadratic equation is:

$$d_X = [- B \pm (B^2 - 4AC)^{1/2}] / 2A \qquad (5\text{-}15)$$

where

$$A = (D_{EX}^2 + D_{Ey}^2) / r^2 + (D_{EZ}^2 / r_p^2) \qquad (5\text{-}16)$$

$$B = -2 (D_{EX} E_X + D_{Ey} E_y) / r^2 + (D_{EZ} E_Z / r_p^2) \qquad (5\text{-}17)$$

$$C = (E_X^2 + E_y^2) / r^2 + (E_Z^2 / r_p^2) - 1 \qquad (5\text{-}18)$$

If we get two real solutions for d_X, then the smaller one will be the required value for d_X. If we get only one solution, the line-of-sight is tangent to the ellipsoid. If we get no real solution, then the line of sight is missing the ellipsoid. The distance d_X is stored for use in DPR L1 products.

5.2.8 Derive Geodetic Latitude and Longitude

The target position vector, G in ECEF coordinates, from the Earth center to the point where the line of sight, D_S, intersects the ellipsoid is computed using equation (5-14). The coordinates of the geodetic system, i.e., geodetic latitude and longitude, are computed as follows:

$$\tan(lat_c) = G_{\mathbf{z}} / (G_{\mathbf{x}}^{\mathbf{2}} + G_{\mathbf{y}}^{\mathbf{2}})^{1/2} \qquad (5\text{-}19)$$

$$\tan(lat_d) = \tan(lat_c) / (1 - f)^2 \qquad (5\text{-}20)$$

$$= G_{\mathbf{z}} / (1 - f)^2 (G_{\mathbf{x}}^{\mathbf{2}} + G_{\mathbf{y}}^{\mathbf{2}})^{1/2} \qquad (5\text{-}21)$$

where

lat_c = geocentric latitude
lat_d = geodetic latitude

Geocentric and geodetic longitudes are the same and they are calculated as follows:

$$\tan(lon_d) = \tan(lon_c) = G_{\mathbf{y}} / G_{\mathbf{x}} \qquad (5\text{-}22)$$

$$lon_d = lon_c = \arctan (G_{\mathbf{y}} / G_{\mathbf{x}}) \qquad (5\text{-}23)$$

where

lon_c = geocentric longitude

lon_d = geodetic longitude

And note that the atan2 function can be used in implementations to resolve the quadrant of the longitude angle around +/- 180 degrees.

5.3 ANCILLARY CALCULATIONS PER PIXEL

Ancillary data computed on a per-pixel basis include the zenith and azimuth angles of the satellite from the pixel view, and the zenith and azimuth angles of the Sun.

5.3.1 Satellite Zenith and Azimuth Angles

The satellite zenith angle is defined as the angle between a vector pointing vertically up at a local position and a vector to the position of the satellite. The computation method is obtained from Patt and Gregg (1994), as implemented for TRMM C code. At each pixel location, the local unit vectors in the East, North, and Up (or Vertical) directions, U^E, U^N and U^U respectively, are computed. The direction vector, D^{Sat}, from the pixel to the satellite is computed from the satellite position vector P minus the surface pixel location vector G, normalized to unit length. The dot product of D^{Sat} with the East, North, and Up direction unit vectors gives the components of D^{Sat} in the East-North-Up coordinates, $D^{Sat}{}_E$, $D^{Sat}{}_N$, $D^{Sat}{}_U$. Given these component values, the zenith angle, or angle between the local vertical and the direction to the satellite, is given by:

$$\text{zenith angle} = \arctan\{(D^{Sat}{}_N{}^2 + D^{Sat}{}_E{}^2)^{1/2} / D^{Sat}{}_U \} \tag{5-24}$$

The zenith angle is also known as the Earth Incidence Angle (EIA), or simply “incidence angle.” A zero zenith angle means the satellite is directly overhead, while a 90-degree zenith angle would be tangent to the horizon. Note that this calculation does not include effects for atmospheric refraction, which are assumed to be negligible for the typical instrument viewing angles.

The satellite azimuth angle is the rotation phase about vertical from the North direction:

$$\text{azimuth angle} = \text{atan2}(D^{Sat}{}_E , D^{Sat}{}_N) \tag{5-25}$$

where the atan2 function is used to define the azimuth in all four quadrants, +/- 180 degrees. Notice that zero azimuth is defined to the North, and the measurements are positive in a clockwise direction to the East.

5.3.2 Solar Zenith and Azimuth Angles

The determination of the solar zenith and azimuth angles is essentially the same as the satellite zenith and azimuth, with the Earth-to-Sun vector in ECEF coordinates, D^{Sun}, replacing the pixel-to-spacecraft vectors. Thus we require the Earth-to-Sun vector, which will be computed using a model developed by Patt and Gregg (1994) and implemented in the Pathfinder Project and used for the TRMM Mission.

5.3.3 Sun Glint Angle

The Sun glint angle is defined here as the angular separation between the surface reflected view vector and the Sun direction. It is equivalently the angle between the reflected Sun vector and the satellite direction at the pixel location. The reflection is considered a specular (mirror-like) reflection off the plane tangent to the ellipsoid at the pixel surface location. The reflected Sun vector, D^{Refl}, simply changes the sign of the East and North components in East-North-Up coordinates, i.e.:

$$D^{Refl}{}_E = -D^{Sun}{}_E \qquad (5\text{-}26a)$$

$$D^{Sun}{}_N = -D^{Sun}{}_N \qquad (5\text{-}26b)$$

$$D^{Refl}{}_U = D^{Sun}{}_U \qquad (5\text{-}26c)$$

Then the Sun glint angle is computed from the arccosine of the dot product of the reflected Sun vector and the satellite direction.

$$\text{Sun glint angle} = \arccos\left(D^{Refl}{}_E * D^{Sat}{}_E + D^{Sun}{}_N * D^{Sat}{}_N + D^{Refl}{}_U * D^{Sat}{}_U\right) \qquad (5\text{-}27)$$

6.0 ANCILLARY CALCULATIONS PER SCAN

Various calculations are made and data are stored on a per-scan basis by the Geolocation Toolkit (Bilanow, 2012). For discussion here, these are grouped into four areas: Navigation data, Sun data, Moon data, and derived flags.

6.1 NAVIGATION DATA GROUP

The following items are stored in 1B products under the group name Navigation:

- Time of mid-scan.
- Spacecraft position.
- Spacecraft velocity.
- Spacecraft latitude.
- Spacecraft longitude.
- Spacecraft altitude.
- Geocentric roll.
- Geocentric pitch.
- Geocentric yaw.
- Geodetic roll.
- Geodetic pitch.
- Geodetic yaw.
- Greenwich Hour Angle.

The methods by which these are obtained are discussed below.

6.1.1 Time of the Mid-Scan

The mid-time for each scan is computed as the average of the times for the first pixel and the last pixel in the scan. All the per-scan navigation data are taken at this mid-time of the scan.

6.1.2 Spacecraft Position and Velocity

This is obtained by interpolating vectors in the ephemeris files. A cubic interpolation is nominally used based on the positions and velocities nearest the point in time desired.

6.1.3 Spacecraft Latitude and Longitude

These are computed by the same method by which the pixel latitude and longitude are computed, except that the line-of-sight direction for intersection with the Earth ellipsoid is taken as the geodetic nadir direction from the spacecraft. This is also referred to as the subsatellite position. The time sequence of the subsatellite position is the ground track of the satellite.

6.1.4 Spacecraft Altitude

The equations for the intersection with the Earth ellipsoid also derive the distance, d_X (see equation 5-15), from the spacecraft to the surface intersection. When the direction is the geodetic nadir, this gives the geodetic altitude.

6.1.5 Spacecraft Geodetic Roll, Pitch, and Yaw

The geodetic roll, pitch, and yaw are computed by the onboard flight software (for the geodetic to Flight coordinate rotation, translated to a 3-2-1 Euler representation), and included in the downlink ancillary telemetry at 10 Hz. Files of the attitude history are generated for each granule of processing by the PPS GeoBuilder software. These data are read and interpolated to the desired time nominally using linear interpolation. In this way the geodetic attitude angles are obtained for the scan mid-time for the navigation data output. The angles are also used to compute the attitude matrix at the scan mid-time using the approach described in Section 5.2.1.

6.1.6 Spacecraft Geocentric Roll, Pitch, and Yaw; Orbital Geocentric Coordinates (OGC)

The geocentric roll, pitch, and yaw are defined relative to the orbital geocentric coordinate system. This is defined similar to the geodetic coordinates as described in Section 5.2.5 except that the Z, nadir direction, is taken toward the center of the Earth (rather than perpendicular to the surface—the difference is illustrated in Figure 1-2). Some missions (such as TERRA and AQUA) use the geocentric definition for pitch, roll, and yaw, and also control the spacecraft to point its Z axis to the geocentric nadir. On the TRMM Mission, control was essentially to the geodetic nadir, as is control for GPM; however, reporting of the attitude in the navigation data gave the geocentric pitch, roll, and yaw (as specified by top-level Earth Observation System Data Information System requirements, which were initially flowed down to TRMM). The different definitions of pitch, roll, and yaw have led to confusion in numerous circumstances for data users. In order to help clarify the difference and provide users with additional information, both definitions of pitch, roll, and yaw are provided here, and values using both definitions are supplied for the mid-scan time as included in the Level 1 file specifications for GPM.

The orbital geocentric coordinates (OGC) are constructed using the spacecraft position, $\mathbf{P^E}$, velocity, $\mathbf{V^E}$, in ECEF coordinates as follows. Essentially this is done by constructing each of the axes of the OGC frame ($\mathbf{X^O}$, $\mathbf{Y^O}$, $\mathbf{Z^O}$) as defined in ECEF coordinates.

The Z axis of the orbital coordinates in ECEF, $\mathbf{Z^O}$, is in the geocentric nadir direction, which is a unit vector directly opposite the spacecraft position vector, $\mathbf{P^E}$:

$$\mathbf{Z^O} = -\mathbf{P^E} \,/\, (\, \| \mathbf{P^E} \| \,) \tag{6-1}$$

where $\| \mathbf{P^E} \|$ indicates the magnitude of the position vector $\mathbf{P^E}$.

The Y axis of orbital coordinates is opposite the orbit normal direction in inertial space, and the X axis is in the orbit plane in inertial space close to the inertial velocity direction. Since ECEF is a rotating frame relative to inertial space, the velocity vector, $\mathbf{V^E}$, in ECEF must be adjusted for the Earth sidereal rotation rate (Patt, 1994) to provide an effective velocity relative to inertial coordinates, V′. Let the Earth sidereal rate be given by omega, $\mathbf{\Omega_E}$, in radians per second. The components of V′ are given by:

$$V'_x = \mathbf{V^E}_x - \mathbf{\Omega_E}\ \mathbf{P^E}_y \qquad (6\text{-}2a)$$
$$V'_y = \mathbf{V^E}_y + \mathbf{\Omega_E}\ \mathbf{P^E}_x \qquad (6\text{-}2b)$$
$$V'_z = \mathbf{V^E}_z \qquad (6\text{-}2c)$$

The normalized cross product of $\mathbf{Z^{GE}}$ and the adjusted velocity V′ give the $Y^{\mathbf{GE}}$ direction.

$$Y^{\mathbf{O}} = \mathbf{Z^O} \times V' \ / \ (\ \| \ \mathbf{Z^O} \times V' \ \| \ ) \qquad (6\text{-}3)$$

The cross product of $Y^{\mathbf{O}}$ and $Z^{\mathbf{O}}$ gives $X^{\mathbf{O}}$.

$$X^{\mathbf{O}} = Y^{\mathbf{O}} \times Z^{\mathbf{O}} \qquad (6\text{-}4)$$

Since $Y^{\mathbf{O}}$ and $Z^{\mathbf{O}}$ are already unit vectors and perpendicular, $X^{\mathbf{O}}$ also computes as a unit vector.

These vectors, $X^{\mathbf{O}}$, $Y^{\mathbf{O}}$, and $Z^{\mathbf{O}}$, are combined to make the OGC to ECEF rotation matrix, [**O**], as follows:

$$[O] = \begin{bmatrix} X_x^O & Y_x^O & Z_x^O \\ X_y^O & Y_y^O & Z_y^O \\ X_z^O & Y_z^O & Z_z^O \end{bmatrix} \qquad (6\text{-}5)$$

The inverse rotation, ECEF to OGC, is given by the transpose, $\mathbf{[O]^T}$. It can be seen by inspection that $\mathbf{[O]^T}$ applied to a vector in ECEF gives the dot product, or projection, of the ECEF vector onto each of the ECEF coordinate constructions of the OGC axes, thus giving the vector components in the OGC frame.

The rotation from the Geodetic Reference Frame (GRF) as defined earlier in Section 5.2.5 to the OGC frame can be computed by the matrix product of the ECEF-to-Geocentric matrix times the Geodetic-to-ECEF matrix:

$$[\text{Rotation Geodetic to Geocentric}] = \mathbf{[G^{GD2GC}]} = \mathbf{[O]\ [N]} \qquad (6\text{-}6)$$

The “geocentric attitude” rotation, which gives the rotation from the OGC reference frame to the GPM Flight body axes, can be computed in two steps going from the geocentric to geodetic frames and then from geodetic to the Flight coordinates. Thus the “geocentric” attitude matrix is given by the product “geodetic” attitude matrix [A] times the transpose of the Geodetic to Geocentric matrix.

$$[\mathbf{A}^{\mathbf{OGC}}] = [\mathbf{A}]\ [\mathbf{G}^{\mathbf{GD2GC}}]^{\mathbf{T}} \tag{6-7}$$

The geocentric yaw, pitch, and roll angles are defined as the respective Euler rotation angles of a 3-2-1 sequence, and are computed from this matrix as:

$$\text{Geocentric Yaw} = \mathbf{\Phi_{GC}} = \text{atan2}(\ [\mathbf{A}^{\mathbf{OGC}}]_{\mathbf{12}},\ [\mathbf{A}^{\mathbf{OGC}}]_{\mathbf{11}}\ ) \tag{6-8}$$

$$\text{Geocentric Pitch} = \mathbf{\Theta_{GC}} = \text{asin}(\ \mathbf{-[\ A^{OGC}]_{13}}\ ) \tag{6-9}$$

$$\text{Geocentric Roll} = \mathbf{\Psi_{GC}} = \text{atan2}(\ [\mathbf{A}^{\mathbf{OGC}}]_{\mathbf{23}},\ [\mathbf{A}^{\mathbf{OGC}}]_{\mathbf{33}}\ ) \tag{6-10}$$

where the FORTRAN or C "atan2" function is used to define the arctangent function over all four quadrants based on the sign of the two input arguments. The subscripts on the matrix $[\mathbf{A}^{\mathbf{OGC}}]_{\mathbf{rc}}$ give the elements in the specified row, **r**, and column, **c**.

6.1.7 Calculate Greenwich Hour Angle

One approximate method is provided for calculating the Earth Rotation Angle, or Greenwich Hour Angle (GHA). Currently it is used to rotate the Sun and Moon vectors, obtained in inertial coordinates, to ECEF coordinates. High accuracy in the calculation of the Sun or Moon direction is not needed. A high accuracy alternative method might be implemented in the future by reading an Earth Orientation Parameters file, which can be obtained from the International Earth Rotation Service (IERS). However, this alternative is not described here.

The Greenwich Hour Angle, h, is calculated with a polynomial approximation:

$$h = 100.4606184 + 0.9856473663\ d + 2.908 \times 10^{-13}\ d^2 \tag{6-11}$$

where

d = days since J2000, with time-of-day expressed as fractional days

and

$$d = jd - 2451545.50 + fd \tag{6-12a}$$

$$jd = 367 * iy - 7 * (iy + (im+9)/12) / 4 + 275 * im / 9 + id + 1721014 \tag{6-12b}$$

where

jd	=	the Julian day of the calendar date
iy, im, id	=	the integer year, month, and day, respectively
fd	=	the fractional day

Nutation correction for the mean sidereal time can be applied as follows:

$$gha = h + d\psi * \cos(\varepsilon) + fd * 360.0 \tag{6-13}$$

Other parameters are calculated as follows:

$$l_S = 280.46592 + 0.9856473516\ d \tag{6-14}$$

$$g_S = 357.52772 + 0.9856002831\ d \tag{6-15}$$

$$l_m = 218.31643 + 13.17639648\ d \tag{6-16}$$

where

l_S	=	mean solar longitude (degrees)
g_S	=	Sun mean anomaly (degrees)
l_m	=	Moon mean longitude (degrees)

The nutation in longitude is computed as follows:

$$d\psi = [-17.1996 \sin(\Omega_m) + 0.2062 \sin(2\Omega_m) - 1.3187 \sin(2l_S) + 0.1426 \sin(g_S) - 0.2274 \sin(2l_m)] / 3600 \tag{6-17}$$

and

$$\Omega_m = 125.04452 - 0.0529537648\ d \tag{6-18}$$

where

Ω_m = ascending node of the Moon's orbit

The obliquity of the ecliptic ε is given by:

$$\varepsilon = \varepsilon_M + d_\varepsilon \tag{6-19}$$

$$\varepsilon_M = 23.439291 - 3.560 \times 10^{-7}\ d \tag{6-20}$$

$$d_\varepsilon = [9.2025 \cos(\Omega) + 0.5736 \cos(2l_S)] / 3600 \tag{6-21}$$

where

ε_M	=	mean obliquity
d_ε	=	nutation in obliquity

6.2 SUN DATA GROUP

The following items are stored in the Sun data group:

- Solar beta angle.
- Phase from orbit midnight.
- Sun-Earth separation angle.
- Earth angular radius.
- Orbit phase of eclipse exit.
- Orbit rate.
- Time since eclipse entry.
- Sun vector in body (Flight) coordinates.

Calculation of these additional items is described below.

6.2.1 Solar Beta Angle

The solar beta angle gives the elevation of the Sun from the satellite orbit plane, with positive being toward the positive orbit normal direction. The solar beta angle calculation involves the computation of the Sun vector, which will be carried out using heritage TRMM code.

The unit orbit normal, **N**, is given by cross product of the position and velocity vectors in inertial coordinates.

$$\mathbf{N} = \mathbf{P} \times \mathbf{V} \qquad (6\text{-}22)$$

Let **S** be the unit Sun vector and p is the dot product of **N** and **S**.

$$p = \mathbf{N} \bullet \mathbf{S} \qquad (6\text{-}23)$$

Now the solar beta angle, B, is just the complement of the arccosine of this dot product:

$$B = 90 - \cos^{-1}(p) \qquad (6\text{-}24)$$

Since the solar beta angle for GPM will vary between +/- 88.5 degrees (this is the sum of the 65-degree TRMM orbit inclination and the 23.5-degree tilt of the ecliptic plane from the Earth equatorial plane, which defines the total possible variation of the Sun from the orbit plane), there is no need for special checks for p being near plus or minus one. Better accuracy may be obtained by using a cross product formula near +/- 90 beta angles; however, the accuracy is not significantly degraded at 88.5 degrees, and high accuracy is not needed for this calculation.

6.2.2 Phase From Orbit Midnight

To get the Sun phase from orbit midnight, we first compute the Sun vector in orbital coordinates. Orbital coordinates are defined with Z toward the geocentric nadir, X toward the velocity direction in inertial coordinates, and Y opposite the orbit normal direction. Let $\mathbf{S_{Ox}}$, $\mathbf{S_{Oy}}$, $\mathbf{S_{Oz}}$ be the X, Y, and Z components, respectively, of the Sun in orbital coordinates. The Sun angular phase from orbit midnight is given by:

Sun phase from orbit midnight = atan2 ($\mathbf{S_{Ox}}$ / $\mathbf{S_{Oz}}$) (6-25)

6.2.3 Sun-Earth Separation Angle

Since the Earth direction in orbital coordinates is along the Z direction, the separation angle of the Sun from the Earth is given by:

Sun-Earth separation angle = **asin ($\mathbf{S_{Oz}}$)** (6-26)

6.2.4 Earth Angular Radius

The Earth angular radius is calculated from a spherical Earth model approximation using the local geodetic height, h, (computed earlier) and the subsatellite Earth radius. A formula for the Earth radius, R_E, as a function of latitude is given in Wertz (1978), page 99, equation (4-14).

Then:

Earth angular radius, rho = asin (R_E / (R_E + h)) (6-27)

6.2.5 Orbit Phase of Eclipse Exit

The orbit phase of eclipse exit is estimated based on the spherical trigonometry formula given on page 91 (5-3) in Wertz (1991).

Phase of eclipse exit = acos (cos(rho) / cos(beta)) (6-28)

6.2.6 Orbit Rate

The instantaneous angular orbit rate is computed based on the velocity component that is perpendicular to the position vector.

6.2.7 Time Since Eclipse Entry

The time since eclipse entry is computed based on the estimated orbit rate and the phase angle spanned since eclipse entry.

6.2.8 Sun Vector in Body Coordinates

Using the various rotation matrices computed for coordinate transformations, the Sun vector, which is initially obtained in inertial (J2000) coordinates, can be converted to ECEF, and Instrument body coordinates as needed.

6.3 MOON VECTORS

The Moon direction is optionally calculated in inertial and Instrument coordinates. This is for the use of the GMI instrument algorithm for the purpose of checking for Moon interference in the cold sky calibration view direction. The method of Van Flandern and Pulkkinen (1979) is used to get the Moon vector in inertial space. The vector is rotated to ECEF coordinates using the calculated GHA. Finally, the rotation matrices calculated as part of the geolocation calculations are used to rotate this vector to Flight coordinates and Instrument coordinates.

6.4 DERIVED FLAG DATA

There are several derived flags that the Geolocation Toolkit is responsible for generating to be used in Level 1 products (the variable names in the file specifications are noted in parentheses):

- Spacecraft orientation (scOrientation).
- Pointing status (pointingStatus).
- Geolocation error (geoError).
- Geolocation warning (geoWarning).

The intent of these flags is discussed in the following sections. A key part of the first three flags is that when they are non-nominal, they trigger a bad "dataQuality" flag, which is described in the PPS GPM File Specifications for GPM Products (2012) (this includes Level 1 products).

6.4.1 Spacecraft Orientation

The intent of this flag is to show the yaw orientation of the spacecraft, as was discussed in Section 3.3.1. This flag is similar to a spacecraft mode flag from the TRMM telemetry, although it was modified by science user input to have meaningful angle values for users. Basically, it currently has:

0 +X forward (yaw 0)
180 -X forward (yaw 180)
-8000 Non-nominal pointing
-9999 Missing

As was discussed in Section 3.2, the following combinations of ACSmode and target define the normal orientations yaw 0 and 180:

Yaw = 0 (+X forward) → (ACSmode = 4 and target = 1)
Yaw = 180 (-X forward) → (ACSmode = 4 and target = 3)

The yaw orientations of -90 or +90 currently are not distinguished in this flag. They could be added if desired by users, but it is noted that this information is carried in the target orientation data from the spacecraft (see Section 3.2.2), which is also stored in products for each mid-scan time. The +90 and -90 yaw orientations currently are expected to be used only once, or rarely, for a special calibration maneuver, and they do not represent an orientation that needs to be distinguished for normal data processing.

6.4.2 Pointing Status

The intent of the pointing status flag for GPM was to incorporate other flags so that the GPM "dataQuality" flag being incorporated in Level 1 products could look to one particular value to determine whether the spacecraft is in a normal operational pointing status for science data use. On TRMM, the single acsMode flag provided adequate information, but for GPM it was realized early that both the ACS mode and target flags (see Section 3.2) would be needed to clarify operational pointing. Later, the ephemeris stale flags (see Section 3.4) were picked up to provide additional information.

The conditions for normal pointing based on the ACS mode and target flags are essentially the same as those giving a spacecraft orientation of either yaw = 0 or yaw = 180, i.e.:

Either (ACSmode = 4 and target = 1) or (ACSmode = 4 and target = 3) (6-29)

The stale flags (see Section 3.4), it turns out, have somewhat limited usage because the ACS has implemented some backup options in which the spacecraft data could be valid even if these flags are set. For example, the spacecraft could fly using ground uplinked ephemeris as a backup mode, and if this is done successfully, the lack of GPS data would not be a condition for flagging all of the science data. Thus, the following limited usage of these flags is recommended based on the currently available information and available spacecraft telemetry: A "bad" (non-zero) pointing status will be set for following two conditions:

1. If PVT ephemeris is in use for PPS data processing, and the PVT stale flag is set, then pointingStatus is flagged.
2. If GEONS ephemeris is in use for PPS data processing, and the GEONS stale flag is set, then pointingStatus is flagged.

The details of this flag are defined in the GeoTK design document (Bilanow, 2012), and the values are summarized in the PPS GPM File Specifications for GPM Products (2012).

6.4.3 Geolocation Error

The intent of the geolocation error flag is simply to indicate conditions in which the geolocation calculations failed. For example, if ephemeris or attitude data cannot be obtained for a requested time span, the result just provides fill data. The flag helps indicate the cause of the problem for diagnostic purposes. The details of this flag are defined in the GeoTK design document (Bilanow, 2012), and the bit values are summarized in the PPS GPM File Specifications for GPM Products (2012).

6.4.4 Geolocation Warning

The intent of the geolocation warning flag is simply to indicate conditions that users might want to be warned about (for example, if the spacecraft attitude is outside of its usual limits for a period of time, or if a switch was made to a backup ephemeris source). The details of this flag are defined in the GeoTK design document (Bilanow, 2012), and the bit values are summarized in the PPS GPM File Specifications for GPM Products (2012).

7.0 ALIGNMENT AND CO-ALIGNMENT DATA HANDLING

This section discusses how ground-measured and ground-computed data for the KuPR and KaPR instruments can be used to obtain the angles for incorporation into the GeoTK alignment inputs. First, key intermediate coordinates in the ground alignment calculations are defined, along with the rotations between them that are important in defining the alignments used in flight. Then the initial on-orbit alignment angles are defined, and it is described how adjustments can be incorporated after launch. It is noted that some of these details are preliminary, awaiting GPM Project and JAXA review.

7.1 INTERMEDIATE COORDINATES FOR KAPR ALIGNMENT

A schematic of KaPR and Spacecraft body coordinates systems, and relationships between them, is shown in Figure 7-1. The various coordinate systems are defined briefly as follows.

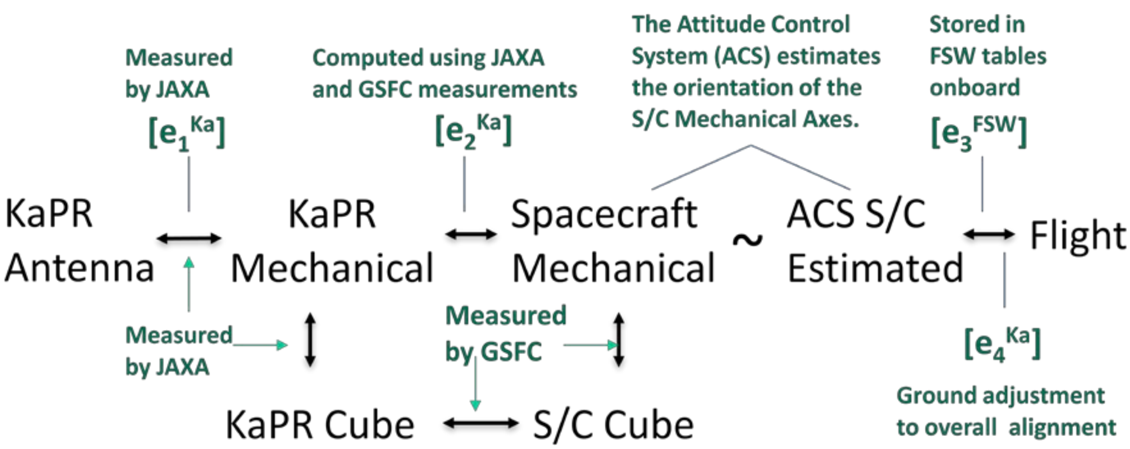

Figure 7-1. Schematic of KaPR and GPM Spacecraft Coordinate Relationships

KaPR Antenna Coordinates: These define reference axes for the beam steering, based on the axis about which phase adjustments change the beam pointing, and a reference direction for the nominal nadir beam (Furukawa, 2011).

KaPR Mechanical Coordinates: These define reference axes based on the measurements of the physical mechanical structure of the KaPR unit.

KaPR Cube Coordinates: The faces of the mirror cube attached to the KaPR structure have a fixed relationship with the KaPR mechanical coordinates, which are measured by JAXA.

S/C Cube Coordinates: The spacecraft master reference cube (MRC) that is used for alignment measurements has a fixed relationship with the Spacecraft coordinates.

Spacecraft Coordinates: The S/C coordinate reference frame is established by reference points and laser tracking on the spacecraft structure, and provides the key central coordinates for all hardware integrated onto the structure.

ACS S/C Estimated Coordinates: This is the orientation of the S/C coordinate frame as estimated by the Attitude Control System (ACS) during spacecraft flight. These are nominally presumed to be exactly the same S/C coordinates as measured on the ground. However, they must be estimated from star tracker calibration data and alignment measurements of the star tracker relative to the master reference cube. These are noted as separate "coordinates" to highlight the potential uncertainties introduced by ground and flight calibrations.

Flight Coordinates: The Flight coordinates represent the spacecraft's Flight body axes that are at a fixed offset from the S/C coordinates. These coordinates were discussed in Section 1.3 and illustrated in Figure 1-1.

Key rotations between these coordinate systems are represented by the following matrices:

$[\mathbf{e_1}^{\mathbf{Ka}}]$ The rotation matrix from KaPR antenna coordinates to KaPR mechanical coordinates, defined so that given a beam direction, $\mathbf{D_A}$, in antenna coordinates, the direction in mechanical coordinates, $\mathbf{D_M}$, is given by:

$$\mathbf{D_M} = [\mathbf{e_1}^{\mathbf{Ka}}]\, \mathbf{D_A} \tag{7-1}$$

This rotation is measured by JAXA in ground testing (Furukawa, 2011).

$[\mathbf{e_2}^{\mathbf{Ka}}]$ The rotation matrix from KaPR mechanical coordinates to S/C mechanical coordinates, defined so that given a beam direction, $\mathbf{D_M}$, in KaPR coordinates, the direction in S/C mechanical coordinates, $\mathbf{D_{S/C}}$, is given by:

$$\mathbf{D_{S/C}} = [\mathbf{e_2}^{\mathbf{Ka}}]\, \mathbf{D_M} \tag{7-2}$$

This rotation is computed from a combination of JAXA-measured and NASA-measured data via the mirror alignment cubes on the instrument and spacecraft structure.

$[\mathbf{e_3}^{\mathbf{FSW}}]$ The rotation matrix from ACS estimated S/C coordinates to Flight coordinates, defined so that given a beam direction, $\mathbf{D_{ACS/C}}$, in ACS S/C coordinates, the direction in Flight coordinates, $\mathbf{D_F}$, is given by:

$$\mathbf{D_F} = [\mathbf{e_3}^{\mathbf{FSW}}]\, \mathbf{D_{ACS/C}} \tag{7-3}$$

This rotation is stored in onboard flight software tables (in quaternion form) based on ground alignment measurements taken before launch. It nominally provides just the 4-degree rotation about the pitch axis between these coordinates as described in Section 4.2. However, it may be slightly different from that value in order to make use of ground alignment measurements and provide consistency with the best estimate for the attitude targeting that will fly the spacecraft with the instruments precisely oriented as desired. There is an understanding between PPS and ACS support personnel (Bilanow, 2011) that this matrix will not be changed in the onboard tables after launch, because it would change the definition of the reported onboard geodetic pitch, roll, and yaw.

It is assumed for the purpose of discussion here that the S/C coordinates and ACS estimated Spacecraft coordinates are the same (or coincident), and so:

$$\mathbf{D}_{\mathbf{S/C}} = \mathbf{D}_{\mathbf{ACS/C}} \tag{7-4}$$

Thus the total rotation from antenna to Flight coordinates is nominally given by:

$$\mathbf{D}_{\mathbf{F}} = [\mathbf{e}_3^{\mathbf{FSW}}]\,[\mathbf{e}_2^{\mathbf{Ka}}]\,[\mathbf{e}_1^{\mathbf{Ka}}]\;\mathbf{D}_{\mathbf{A}} \tag{7-5}$$

This represents the nominal conversion based on the ground measurements and onboard loads. However, it is useful to add an additional term, $[\mathbf{e}_4^{\mathbf{Ka}}]$, to take into account any errors that are detected in the instrument alignment after launch. We choose to place this term as a last step adjusting the rotation to Flight coordinates.

$$\mathbf{D}_{\mathbf{F}} = [\mathbf{e}_4^{\mathbf{Ka}}]\;[\mathbf{e}_3^{\mathbf{FSW}}]\,[\mathbf{e}_2^{\mathbf{Ka}}]\,[\mathbf{e}_1^{\mathbf{Ka}}]\;\mathbf{D}_{\mathbf{A}} \tag{7-6}$$

This term, $[\mathbf{e}_4^{\mathbf{Ka}}]$, can represent a correction for a rotation error in any or all of the other terms. The placement is chosen so that it represents a change relative to Flight coordinates, which are the simplest to implement in corrections to Euler angle inputs to the GeoTK software. It means the errors, in terms of a pitch, roll, and/or yaw adjustment to the alignment Euler representations, can be interpreted as adjustments to the nominally reported attitude telemetered from the spacecraft.

In subsequent subsections, some possible post-launch corrections to the alignment angles used in the GeoTK software are discussed in terms of pitch and roll corrections. The $[\mathbf{e}_4^{\mathbf{Ka}}]$ term is just a matrix representation of the corrections that are applied to the ground alignments.

Obviously, one possible source of error in this end-to-end rotation is the fact that the onboard Attitude Control System cannot be expected to estimate directions exactly in the original ground-measured S/C coordinates. However, it will not be distinguishable, after launch, where errors may have occurred in ground measurements of any of the rotations. It is possible that errors are also introduced by "launch shock" or thermal distortion of the structure, although prelaunch analysis limits the expected magnitude from these other sources. A discussion of the error budget for the instrument alignment is provided in the GPM Project Pointing and Alignment Specification (Harper, 2012).

7.2 INTERMEDIATE COORDINATES FOR KUPR ALIGNMENT

It is simply noted here that the intermediate coordinates for KuPR exactly follow the same pattern as for KaPR. The following matrices represent the equivalent rotations to those explained for KaPR above:

$[e_1^{Ku}]$ The rotation from KuPR antenna coordinates to KuPR mechanical coordinates.

$[e_2^{Ku}]$ The rotation from KuPR mechanical coordinates to S/C mechanical coordinates.

$[e_4^{Ku}]$ The rotation adjustment for KuPR alignment.

In Figure 2-1, Ku can be substituted for Ka to show the equivalent relationships.

7.3 INCORPORATION OF GROUND ALIGNMENT DATA AT LAUNCH

The alignment measurements between the master reference cube and the instruments and key ACS components on the spacecraft (like the star trackers) are made at many times using theodolites during the integration and test (I&T) schedule. Measurements are taken after instrument integration, before and after vibration tests, before and after thermal tests, and after shipping to the launch site. A decision will be made before launch by the GPM Project and JAXA as to the best numbers that should be used for the GeoTK software, and what numbers will be loaded into the flight software tables.

The alignment inputs to the GeoTK software incorporate the total rotation between Flight coordinates and the Instrument coordinates in which ray pointing vectors are input. JAXA currently plans to input the ray vectors in the KaPR and KuPR mechanical coordinates. Therefore, the matrices are combined as follows to get the desired rotation matrix representations for the Instrument-to-Flight rotation, which is labeled $[\mathbf{S}]^T$ in equation (5-1):

For KaPR:

$$[S^{Ka}]^T = [e_3^{FSW}]\,[e_2^{Ka}] \tag{7-7}$$

And for KuPR:

$$[S^{Ku}]^T = [e_3^{FSW}]\,[e_2^{Ku}] \tag{7-8}$$

Note that for the GeoTK design, the angles input for the alignment matrix, **[S]**, represent a Flight-to-Instrument rotation, which is the inverse rotation of Instrument-to-Flight, $[\mathbf{S}]^T$ (given by the transpose matrix, since rotation matrices are orthogonal).

The inputs to the GeoTK software (Bilanow, 2012) use an Euler angle representation for any selected Euler rotation sequence. Generally the Euler angle representation provides more meaningful input for users. So, given the matrix representation above, an Euler sequence remains to be chosen, and the proper rotation angles that define the matrix must be computed. It is usually best to choose the largest rotation first to avoid coupling in the interpretation of the other rotations. In this case, the largest rotation is the nominal 4-degree rotation about the Y (or pitch) axis. Roll and yaw alignment offsets are expected to be small. A 2-1-3 sequence has been used initially in tests, but it is noted that any sequence can be used, just so it is correctly defined and applied.

GPM Project representatives have noted informally that they can provide the alignment data for the GeoTK software in any rotation representation desired; however, the procedures for documentation and delivery of the final values remain to be determined by the GPM Project. It will be useful if the transformations can be checked by multiple parties. If a matrix representation is available, tools are already available in the Geolocation Toolkit library (e.g., GeoTKeuler.c) to convert a matrix representation to any desired Euler sequence. The formulas are given in Wertz, 1978.

It should be noted that the final ground alignment measurements will not be expected to create conditions such that both sensors are nominally looking in exactly the same direction in S/C coordinates. It also should be noted that the formulation for the nadir beam direction in antenna coordinates (Furukawa, 2011) constrains the Ka and Ku nadir beams to match for the nominal alignments given, so the e1 rotations used here should represent changes from those nominal alignments. It is useful to keep track of the expected difference in the two beams pointing. We will want to track this rotational difference for the nominal nadir pointing beam for the instruments. The total rotation from Ka antenna coordinates to Ku antenna coordinates follows the following sequence of rotations:

$$[\mathbf{e}^{\mathbf{Ka\text{-}Ku}}] = [\mathbf{e}_1{}^{\mathbf{Ku}}]^{\mathbf{T}}\,[\mathbf{e}_2{}^{\mathbf{Ku}}]^{\mathbf{T}}\,[\mathbf{e}_2{}^{\mathbf{Ka}}]\,[\mathbf{e}_1{}^{\mathbf{Ka}}] \tag{7-9}$$

Since Ka and Ku beams are nominally closely aligned, this comes out as a small rotation, and the aforementioned tools for computing an Euler sequence of rotations from this matrix can readily show components of the expected beam offsets. Alternatively, small angle approximations can be computed from appropriately selected matrix components, e.g.:

Ka to Ku pre-launch estimated nadir beam offset in roll:

$$\boldsymbol{\varepsilon}_{\mathrm{roll}} = \mathrm{asin}\,(\,[\mathbf{e}^{\mathbf{Ka\text{-}Ku}}]_{\mathbf{23}}\,) \tag{7-10}$$

Ka to Ku pre-launch estimated nadir beam offset in pitch:

$$\boldsymbol{\varepsilon}_{\mathrm{pitch}} = \mathrm{asin}\,([\mathbf{e}^{\mathbf{Ka\text{-}Ku}}]_{\mathbf{31}}\,) \tag{7-11}$$

where the subscripts following the matrix brackets indicate the row and column of the matrix. Small differences can be tolerated, or adjustments can be made in the initial on-orbit instrument operation to compensate for the ground-measured misalignment. These adjustments can be accomplished by phase code data loads and/or Ku/Ka timing adjustments, which are discussed in the next section.

7.4 CO-ALIGNMENT ADJUSTMENTS ON ORBIT

JAXA is planning measurements using the Active Radar Calibrator (ARC) to estimate any errors in the in-flight co-alignment for the KaPR and KuPR radars (Furukawa, 2010). Two parameters can be adjusted in the instrument operation to achieve better co-alignment:

1. In roll, phase code data adjustments can shift the beam direction (about the antenna coordinate X axis).
2. In pitch, Ku/Ka timing delays can provide an equivalent effect in the beam surface geolocation.

These adjustments will implicitly provide information about the relative alignment errors in the instrument data. These alignment adjustments need to be incorporated into the GeoTK inputs in order to provide the most optimal geolocation calculations. This section discusses how the adjustments to the GeoTK alignment angles can be made based on the phase code data adjustments and Ku/Ka timing delays.

7.4.1 Phase Code Data Adjustments

Changes in the phase code data introduce a rotation in the beam direction about the antenna coordinate X axis. Roughly this is about the roll axis, except for the approximately 4-degree offset between antenna and Flight coordinates. The difference in an offset at nadir between rotation about the Flight coordinate X axis and about the antenna coordinate X axis is just given by the ratio of the cosine of 4 degrees, which is about 0.9976, so this difference is generally negligible compared to the accuracy to which adjustments can be computed using flight data.

A scenario for roll adjustments is illustrated in Figure 7-2. Initially it is assumed that the Ka and Ku radars are both exactly aligned with the Spacecraft axes for simplicity of illustration. Initially the GeoTK software will be reporting the beams co-located. Now we suppose that the ARC results show that the beams are not actually co-located, but the Ku beam is observed to be shifted to the right on the ground track, relative to the velocity direction (into the page in the illustration). A phase code data load can be planned for the Ku radars that shift the beam views back to the left. Thus, the nadir-looking beam is not along the Ku antenna coordinate Z axis direction as originally set, but is shifted to the left so that it actually aligns with the associated Ka beam. The phase code data load will shift the beams to match, and the adjusted beam directions in Ku mechanical coordinates will be input to the GeoTK software. However, the computed geolocation will not match until alignment adjustments are also included in the GeoTK software.

The alignment adjustment needed in the GeoTK roll axis Euler angle will be almost exactly equal to the phase code data adjustment about the antenna X axis. A small numerical difference can be calculated (as discussed further below) due to the different axes of the antenna and Flight coordinates (as noted in the paragraph above).

The Geolocation Toolkit alignment roll parameter adjustment is the same sign as the phase code data shift adjustment, as verified by test cases run by JAXA (Yoshida, 2012). The sense of rotation is opposite as seen in Figure 7-2, but a sign change is added by the rotation convention for inputs to the GeoTK software (entering angles for [S] rather than $[S]^T$ as noted in Section 5.2.5). The JAXA definitions (Furukawa, 2011) have the beams shift to the right for a positive phase shift, so the illustration in Figure 7-2 is for a negative phase shift. All of the sign conventions will be confirmed with further end-to-end analysis and testing.

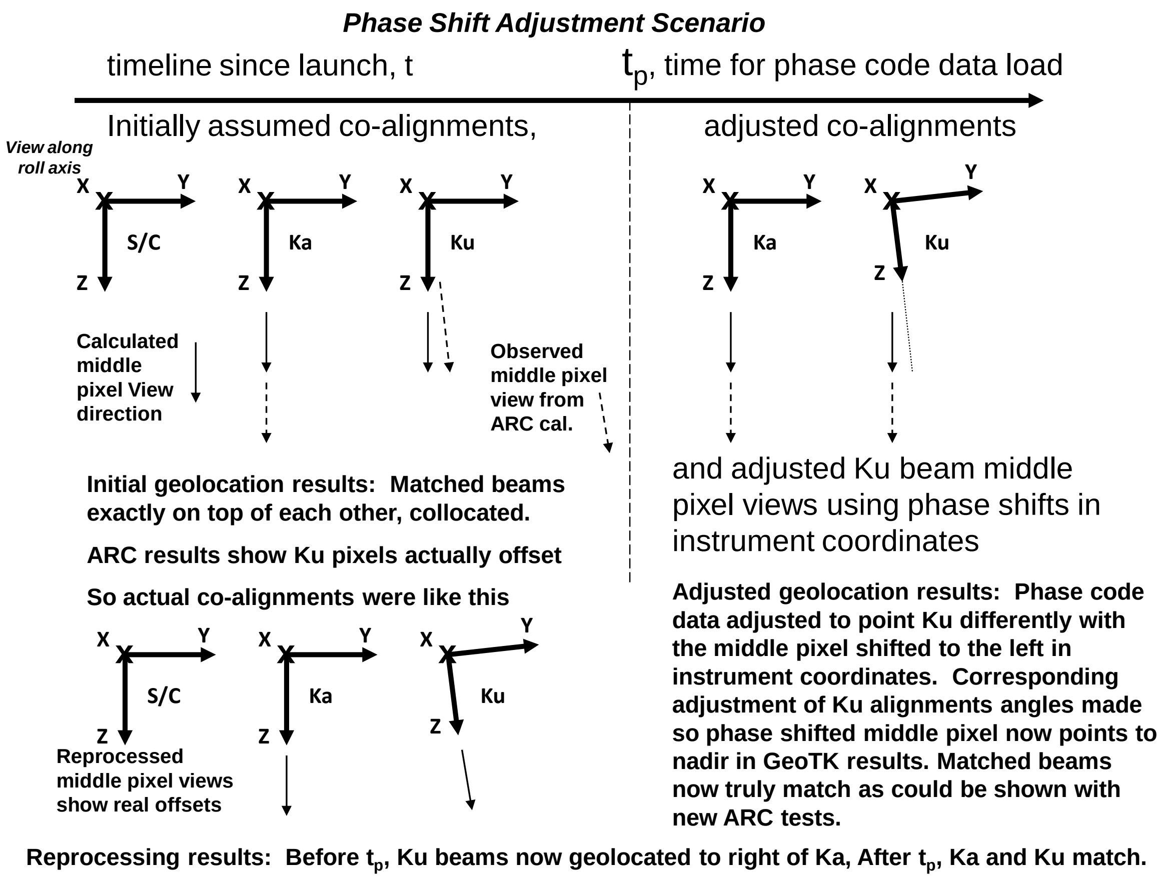

Figure 7-2. Schematic Diagram of DPR Phase Code Data Adjustment Scenario

From co-alignment information taken from ARC measurements, it will not be known if the error is due to Ku beams offset in one direction, or Ka beams offset in the other direction. Which beam is shifted is unimportant for the purpose of getting co-aligned geolocation, i.e., matched latitude and longitude coordinates.

Either may be chosen for adjustment onboard the spacecraft, just so the relative offsets match the relative offsets of the beams in the sensor Spacecraft body coordinates. If information is obtained on the absolute alignment of either KaPR or KuPR, it will be important that adjustments be made to both to maintain the correct relative co-alignment. This is discussed further in Section 7.5.

Calculation of alignment angle adjustments based on a phase code data adjustment can be derived as follows. To a close approximation, the adjustment needed is that which locates the shifted beam at the same offset in roll at which the other beam is looking. Suppose it is the KuPR beam that is being shifted, as illustrated in Figure 7-2, and the KuPR phase code data are loaded to shift in all the beam rotation angles of delta-phi, $\Delta\Phi = -0.1$ degree. Further, suppose for simplicity that the original co-alignment was perfect ($\boldsymbol{\varepsilon}_{roll} = 0.0$). In this case, this means that the KuPR alignment rotation in roll, $\boldsymbol{\Delta a_{roll}^{Ku}}$, must be shifted by -0.1 degree (to a value of -0.1 degree if it was initially 0.0 degree).

In the case in which there was some initial misalignment modeled in the beam directions for the geolocation calculations, say based on the ground alignment, this misalignment offset must be removed from the beam adjustment, giving an adjustment in Ku alignment about the Ku antenna coordinate X axis, or roll axis by the formula:

$$\boldsymbol{\Delta a_{roll}^{Ku} \approx \Delta\Phi^{Ku} - \varepsilon_{roll}} \qquad (7\text{-}12)$$

Where delta-phi, $\boldsymbol{\Delta\Phi^{Ku}}$, is the KuPR phase shift adjustment, $\boldsymbol{\varepsilon_{roll}}$ is the initial roll misalignment modeled in the beam directions for GeoTK calculations, and the computed delta-alignment-in roll, $\boldsymbol{\Delta a_{roll}^{Ku}}$, is the adjustment to the KuPR alignment angle that is associated with roll for the selected alignment angles' Euler sequence. This is the second alignment angle for the 2-1-3 sequence used in the Section 5-1 example. The "approximately equal" symbol is used in this equation because (for the nominal alignment Euler sequence used) the roll axis for alignment adjustments is shifted by 4 degrees from the axis of phase shift adjustments. The exact formulation is discussed below with the full matrix formulation for the adjustment calculation, but this approximation error is expected to be insignificant.

Thus, for example, if $\boldsymbol{\varepsilon}_{roll} = 0.02$ degree, and $\boldsymbol{\Delta\Phi^{Ku}}$ is -0.1, this gives:

$$\boldsymbol{\Delta a_{roll}^{Ku}} = (-0.1 - 0.02) = -0.12 \text{ degree}) \qquad (7\text{-}13)$$

This is needed because the ARC calibration estimates the overall adjustment in the phase code data needed, assuming they are aiming to be aligned exactly. Thus, we want to adjust relative to assumed geolocation estimates that would have them co-located exactly. It should be noted that this adjustment assumes the $\boldsymbol{\varepsilon_{roll}}$ offset is due to differences in the e_2^{Ka} and e_2^{Ku} alignments on the spacecraft. (Our current understanding of the model, Furukawa 2011, is that the e_1 rotations from KaPR and KuPR do not contribute to any difference in the nominal nadir beam directions.)

If the e_1 rotations create significant difference in the Instrument coordinate nadir beam directions, then additional bookkeeping will be needed to distinguish the different contributions to the offsets for the GeoTK alignment adjustment versus the adjustments to input beam directions calculated by JAXA for input to the Geolocation Toolkit (GeoTK) software.

The alignment adjustment results can be checked with the GeoTK software by simply verifying that the calculated geolocation (latitude and longitudes) is co-located for the matched pixels after the new alignments are applied.

A matrix representation of the co-alignment calculation is presented as follows, giving a more exact adjustment formula. Initially, perfect co-alignment (i.e., $[\mathbf{e}^{Ka\text{-}Ku}] = \mathbf{I}$ or $\boldsymbol{\varepsilon}_{roll} = \boldsymbol{\varepsilon}_{pitch} = 0$) implies:

$$[\mathbf{e}_2^{Ka}]\,[\mathbf{e}_1^{Ka}] = [\mathbf{e}_2^{Ku}]\,[\mathbf{e}_1^{Ku}] \qquad (7\text{-}14)$$

Equivalently in Flight coordinates:

$$[\mathbf{e}_3^{FSW}]\,[\mathbf{e}_2^{Ka}]\,[\mathbf{e}_1^{Ka}] = [\mathbf{e}_3^{FSW}]\,[\mathbf{e}_2^{Ku}]\,[\mathbf{e}_1^{Ku}] \qquad (7\text{-}15)$$

Or, showing the matrices used in GeoTK inputs:

$$[\mathbf{S}^{Ka}]^T\,[\mathbf{e}_1^{Ka}] = [\mathbf{S}^{Ku}]^T\,[\mathbf{e}_1^{Ku}] \qquad (7\text{-}16)$$

If we introduce an adjustment to the $[\mathbf{e}_1^{Ku}]$ rotation by a new rotation, $[\mathbf{e}_0^{Ku}]$, and aim for perfect co-alignment under this condition, this may be written:

$$[\mathbf{S}^{Ka}]^T\,[\mathbf{e}_1^{Ka}] = [\mathbf{S}^{Ku}]^T\,[\mathbf{e}_1^{Ku}]\,[\mathbf{e}_0^{Ku}] \qquad (7\text{-}17)$$

However, we want a new representation of the alignment rotation $[\mathbf{S}^{Ku}]$, call it $[\mathbf{S}^{Ku2}]$, that we can use in our original form, multiplying just $[\mathbf{e}_1^{Ku}]$ to calculate the overall rotation to Flight coordinates; i.e., we want:

$$[\mathbf{S}^{Ku2}]^T\,[\mathbf{e}_1^{Ku}] = [\mathbf{S}^{Ku}]^T\,[\mathbf{e}_1^{Ku}]\,[\mathbf{e}_0^{Ku}] \qquad (7\text{-}18)$$

The value needed for the new rotation matrix is computed by post-multiplying above by $[\mathbf{e}_1^{Ku}]^T$:

$$[\mathbf{S}^{Ku2}]^T = [\mathbf{S}^{Ku}]^T\,[\mathbf{e}_1^{Ku}]\,[\mathbf{e}_0^{Ku}]\,[\mathbf{e}_1^{Ku}]^T \qquad (7\text{-}19)$$

Using this matrix approach, $[\mathbf{e}_0^{Ku}]$ can be constructed to represent the phase code data change about the Ku antenna coordinate X axis, and the new alignment matrix $[\mathbf{S}^{Ku2}]$ calculated. The aforementioned utilities (GeoTKeuler.c) can be used to get the alignment angle inputs needed for the updated GeoTK inputs. The changes from the previous alignment angles can be noted, and the changes as represented by an $[\mathbf{e}_4^{Ka}]$ matrix representation of the change can be calculated if desired.

7.4.2 Ku/Ka Timing Adjustments

Changes in the Ku/Ka timing of the beams introduce an along-track shift in the geolocation of the beams similar to the effect of a pitch shift in the beam direction. This is illustrated in Figure 7-3. For example, suppose it is found that the beams initially arrive at an ARC site with the Ka radar beam center passing over the ARC site 0.1 second ahead of the Ku radar beam, because the Ka radar is looking ahead (pitched up toward the flight direction). A delay would then be introduced to the Ku radar pulses of 0.1 second so that corresponding radar pulses are directed to the same location for future data sampling.

The Ka and Ku pixel center locations would have been calculated initially to be at the same location at the same time (although that did not reflect reality). After the delay is introduced and if no associated alignment changes are made, then pixel centers would be calculated as separated by about 700 meters along track due to the 0.1 second time delay. However, the fact that delay was introduced so the ARC would detect the associated beams (for pixel locations being sampled) at the same location (with detection at the slight time delay as noted) means that there is a relative misalignment of about 0.1 degree between the two beams about the pitch axis.

Either Ka has a positive pitch bias (as illustrated) or Ku has a negative pitch bias in this case. (The case illustrated has GPM flying at 0 degree yaw orientation with the +X Spacecraft axis pointing forward.) Either bias can be applied to correct the relative misalignment.

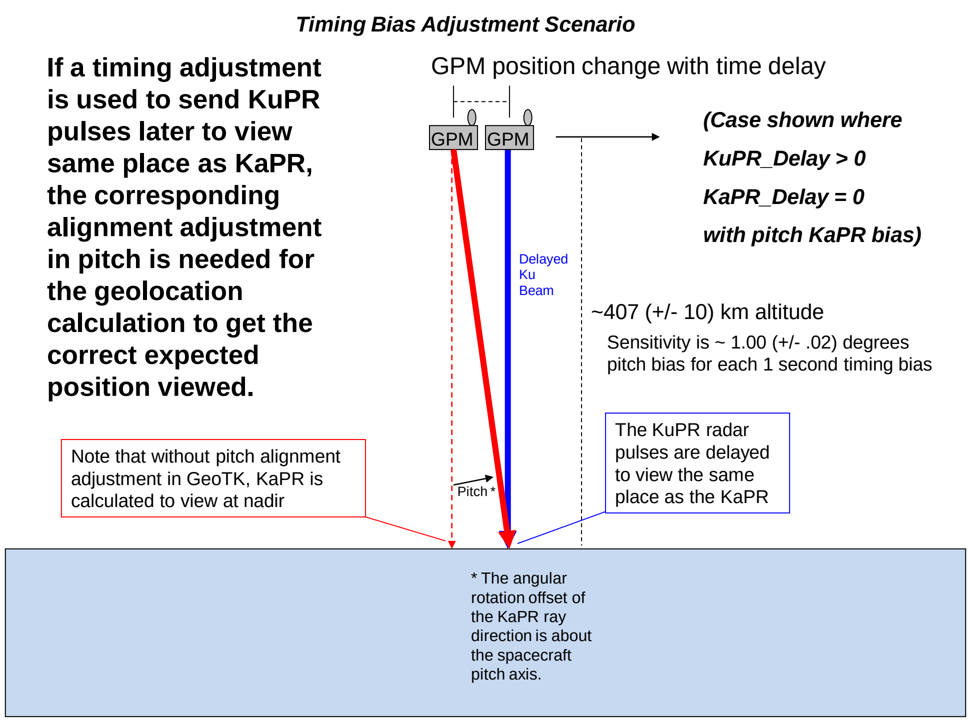

Figure 7-3. Timing Bias Adjustment Scenario

The Ka/Ku timing delay applied actually needs to change sign when the GPM spacecraft changes its yaw orientation (which occurs roughly monthly), and this will be done by applying the delay alternately to KaPR or KuPR in command loads to the spacecraft with each yaw turn. However, the alignment adjustment needed does not change sign with the yaw turn; the properly selected alignment adjustment should remain constant throughout the mission.

The effect of the pitch offset on the Ku/Ka timing delay at the surface changes slightly with geodetic altitude variations of the spacecraft, and to a lesser extent due to orbit non-circularity (eccentricity). At the mean altitude of 407 km, the orbit period is about 95 minutes. The subsatellite location moves at about 7 kilometers per second. More precisely—calculated from one orbit simulation—it moves about 6.99 km/sec at high latitudes and about 7.03 km/sec over the equator. These values should be checked over the typical variations that may occur in orbit altitude and eccentricity, but this is probably very close to the usual range. The geodetic altitude (above the oblate Earth) is required to stay between 397 to 419 km. It varied between 402 and 418 km in the simulation where the orbit rate was just noted. The alignment shift associated with a Ku/Ka timing delay can be estimated from simple trigonometry based on the along-track motion and the altitude. It is very close to 1 degree of alignment shift associated with 1 second of delay; however, it was calculated to be between 0.96 to 1.01 degrees/second for the noted sample orbit simulation. The calculations are noted here for an indication of the variability in the sensitivity. The actual sensitivity and its variation will be verified with further analysis and simulations.

Since the Ku/Ka timing delay effect is directly about the pitch axis, the pitch adjustments calculated from the sensitivity noted above can be applied directly to the alignment errors. The same type of calculation as was shown for the roll effects at the end of the last subsection can be used to calculate the alignment adjustments including pitch effects more precisely if needed. There is not the effect as in roll for the 4-degree offset of mechanical and Flight coordinates, but there will be some coupling if roll and pitch adjustments are applied at the same time, although this is expected to be generally negligible.

The Ku/Ka timing delay illustrated (applied to Ku with +X forward, yaw = 0, and then applied to Ka during –X forward, yaw = 180 degrees) can be corrected by a positive alignment adjustment in the pitch axis alignment angle, $\mathbf{\Delta Y^{Ka}}$, for KaPR (as illustrated), or a negative adjustment in the KuPR alignment, $\mathbf{\Delta Y^{Ku}}$, as shown in tests (Yoshida, 2012). The opposite pattern of Ku/Ka timing delay would be adjusted by pitch alignment angle adjustments of opposite sign, as will be verified with further analysis and tests as needed.

7.5 MAINTAINING ESTIMATED CO-ALIGNMENT WITH ALIGNMENT CHANGES

If independent checks from sources like coastline checks indicate an overall alignment adjustment in the KaPR and/or KuPR sensors, adjustments to both sensor alignments may be needed. It is likely that the co-alignment may be known to better accuracy than the absolute alignment of either sensor. It is discussed briefly here how the relative co-alignment can be maintained while adjustments are made to the absolute alignment.

Suppose for example that an adjustment to the KaPR absolute alignment is estimated based on some coastline checks. These checks would indicate pitch and roll adjustments relative to the Flight coordinates. To a very close approximation, exactly the same pitch and roll adjustments could be applied to both of the Ka and Ku alignment matrices, and the relative alignments would be kept practically the same. For a detailed matrix calculation, we can translate these pitch and roll adjustments to an error matrix, which we will call $[\mathbf{e}_4^{\mathbf{AdjDPR}}]$. Thus, the new alignment matrix, $[\mathbf{S}^{\mathbf{Ka3}}]$, to be used in the Geolocation Toolkit for the KaPR processing would be calculated from:

$$[\mathbf{S}^{\mathbf{Ka3}}]^{\mathbf{T}} = [\mathbf{e}_4^{\mathbf{AdjDPR}}]\,[\mathbf{S}^{\mathbf{Ka}}]^{\mathbf{T}} \qquad (7\text{-}20)$$

A new alignment matrix for KuPR would be calculated the same way:

$$[\mathbf{S}^{\mathbf{Ku3}}]^{\mathbf{T}} = [\mathbf{e}_4^{\mathbf{AdjDPR}}]\,[\mathbf{S}^{\mathbf{Ku}}]^{\mathbf{T}} \qquad (7\text{-}21)$$

The alignment angles associated with this matrix representation can be calculated using standard tools as noted earlier, so it is straightforward to calculate exactly the new alignment angles associated with any alignment adjustment in the Flight coordinates. Tools to verify the numbers for the exact combination of rotations will be useful for analysis and support of the mission. However, it is reiterated that to a very close approximation, the same adjustments can simply be made to the same axes for these nearly co-aligned instruments.

8.0 REFERENCES

1. Bilanow, S., "PPS Geolocation Toolkit Architecture and Design Specification Document Algorithm Theoretical Basis Document," PPS Publication, PPS_610.2_P520, Version 2, NASA GSFC, 2012.

2. Bilanow, S., "Operations Working Group Meeting Notes," NASA GSFC, September 23, 2011.

3. Furukawa, K., "Study of Beam Matching Measurement Accuracy Using Radar Calibrator for GPM Dual Frequency Precipitation Radar," GPM/DPR Project Team, Technical Document Number SBG-090004, January 14, 2010.

4. Furukawa, K., "GeoTK DPR Alignment Information," Slides Presented at GSIM #11, August 19, 2011.

5. "GPM Core Observatory Performance Requirements," NASA GSFC, GPM-SE-REQ-0030, Revision E, February 16, 2012.

6. Harper, A., "Global Precipitation Measurement (GPM) Project Pointing and Alignment Specification," GPM-MSE-SPEC-0018, Revision D, NASA GSFC, February 16, 2012.

7. Hensley, M., "PPS GeoBuilder Architecture and Design Specification Document," PPS Publication, PPS_610.2_P530, Version 2, NASA GSFC, 2011.

8. Masaki, T., and Yoshida, N., "DPR L1B Status," Slides Presented at GSIM #14, July 11, 2012, 14.

9. Patt, F. S., and Gregg, W., 1994, "Exact Closed-Form Geolocation Algorithm for Earth Survey Sensors," Int. J. Remote Sensing, Vol. 15, No. 18, 3719-3734.

10. Patt, F.S., 2002, "NASA Technical Memorandum 2002-206892, Volume 16, Navigation Algorithms for the SeaWiFS Mission," Section 2.1, Orbit Integration and Interpolation, http://oceancolor.gsfc.nasa.gov/cgi/postlaunch_tech_memo.pl?16.

11. "Precipitation Processing System Global Precipitation Measurement File Specification for GPM Products," Version 1P6, NASA GSFC, March 23, 2012.

12. Van Flandern, T. C., and Pulkkinen, K. F., "Low-Precision Formula for Planetary Positions," Ap. J. Supplement Series, 1979, 41:391-411.

13. Wertz, J. R., Ed., 1978, "Spacecraft Attitude Determination and Control," D. Reidel Publishing Company.

14. Wertz, J. R., Ed., 1991, "Space Mission Analysis and Design," Kluwar Academic Publishers.

15. Yoshida, N., and Masaki, T., "1B_GeoTK_Input_Sample_NewSimulation_V02," JAXA/EORC, RESTEC Slides, January 26, 2012.

16. Zhu, Juie, "Conversion of Earth-Centered Earth-Fixed Coordinates to Geodetic Coordinates," Aerospace and Electronic Systems, IEEE Transactions on Vol. 30, No. 3, July 1994, 957-961.

9.0 ACRONYMS

ACS	Attitude Control System
AGS	Attitude Ground System
ARC	Active Radar Calibrator
ASCII	American Standard Code for Information Exchange
ATBD	Algorithm Theoretical Basis Document
CCB	Configuration Control Board
CCSDS	Consultative Committee for Space Data Systems
CFOV	Center of Field-of-View
CGS	Core GPS System
CM	Configuration Management
CMO	Configuration Management Office
DPR	Dual-Frequency Precipitation Radar
ECEF	Earth-Centered, Earth-Fixed
ECI	Earth-Centered Inertial
EFOV	Effective Field-of-View
EIA	Earth Incidence Angle
F	Flight
GCI	Geocentric Inertial Coordinates
GEONS	Goddard-Enhanced Onboard Navigation System
GeoTK	Geolocation Toolkit
GHA	Greenwich Hour Angle
GMI	GPM Microwave Imager
GPM	Global Precipitation Measurement
GPS	Global Positioning System
GRF	Geodetic Reference Frame
GSFC	Goddard Space Flight Center
GSIM	Ground Systems Interface Meeting
I&T	Integration and Test
IERS	International Earth Rotation Service
JAXA	Japan Aerospace Exploration Agency
KaPR	Ka-Band Precipitation Radar
KuPR	Ku-Band Precipitation Radar
L1	Level 1
L1B	Level 1B
MMS	Magnetospheric Multi-Scale Mission
MOC	Mission Operations Center
MRC	Master Reference Cube
MS/HS	Matched Scan/High Sensitivity
NASA	National Aeronautics and Space Administration
NAV	GPS Navigator System
OBP	Onboard Propagated
OEM	Orbital Ephemeris Message

OGC	Orbital Geocentric Coordinates
PPS	Precipitation Processing System
PR	Precipitation Radar
PVT	Position Velocity Time
SeaWiFS	Sea-viewing Wide field-of-view Sensor
S/C	Spacecraft
TMI	TRMM Microwave Imager
TRMM	Tropical Rainfall Measuring Mission
VIRS	Visible and Infrared Sensor
WGS-84	World Geodetic System-84