MRO HIGH RESOLUTION IMAGING SCIENCE EXPERIMENT (HIRISE): INSTRUMENT TEST, CALIBRATION AND OPERATING CONSTRAINTS

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ABSTRACT

HiRISE supports the Mars Reconnaissance Orbiter (MRO) Mission objectives through targeted imaging of nadir and off-nadir sites. Orbital images with high resolution and high signal to noise ratio will be obtained from Mars orbit. The images will have a scale of 25 to 32 cm per pixel from the nominal orbit of 250 x 320 km. HiRISE is a "push-broom" camera with a swath width at 300 km altitude of 6 km in a broad red spectral band and 1.2 km in blue-green and near infrared bands. There are 14 CCD detector chips (2048 x 128 elements each) on the focal plane. The HiRISE camera has a half-meter primary mirror, yet through the use of lightweight glass optics and graphite-composite structures, the mass of the instrument is only 65 kg. A uniform telescope temperature of 20°C is maintained and it's estimated orbital average power consumption is less than 60 W.
The large data volume of a single nominal 20,000 x 40,000-pixel image requires 13 minutes to download to the spacecraft and hours to transmit to Earth. Optical alignment and testing is complicated by gravity sag in the lightweight structure and an IFOV of only 1.0 µrad. Low-noise CCD performance was attained at a rate of 16 Mpix/s. 128 levels of time delay and integration (TDI) is used to achieve a signal-to-noise ratio of >150:1, but requires precision timing in the electronics and a quiet spacecraft.

INTRODUCTION

The primary functional requirement of the HiRISE imager, Figure 1, is to allow identification of both predicted and unknown features on the surface of Mars to a much finer resolution and contrast than previously possible [1], [2]. This results in a camera with a very wide swath width, 6 km at 300 km altitude, and a high signal-to-noise ratio (SNR), >100:1. Generation of terrain maps, with 30 cm vertical resolution, from stereo images requires very accurate geometric calibration. The project limitations of mass, cost and schedule have made the development challenging. In addition, the spacecraft stability [3] must not be a major limitation to image quality. The nominal orbit for the science phase of the mission is a 3pm orbit of 255 by 320 km with periapsis locked to the south pole. The ground track velocity is approximately 3,200 m/s.

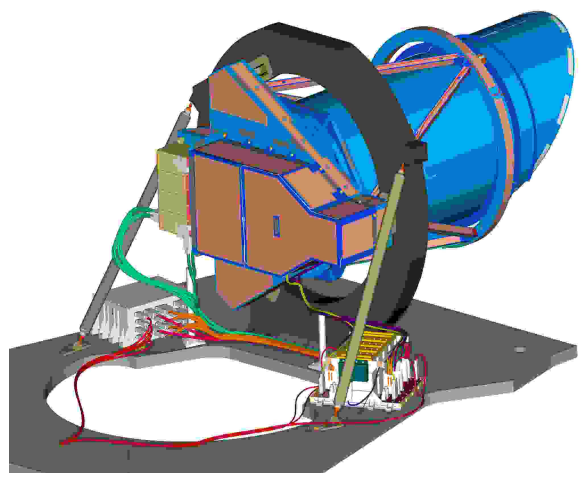

Figure 1: HiRISE imager assembly mounted to spacecraft support ring. The two electronics boxes house the instrument controller and power supply.

HIRISE FEATURES

The HiRISE instrument design features a 50 cm aperture and a detector with 128 lines of Time Delay and Integration (TDI) to create very high signal noise ratio images.

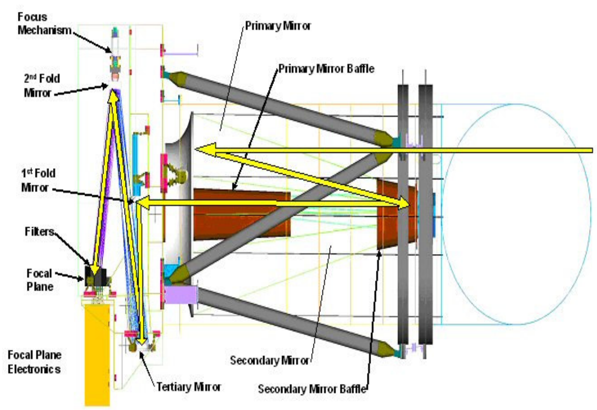

Figure 2: Camera optical path optimized for low mass

The imager design is an all-reflective three mirror astigmatic telescope with light-weighted Zerodur optics and a graphite-composite structure. The Cassegrain objective with relay optic and two fold mirrors is optimized for diffraction-limited performance over the long, narrow field-of-view (FOV) required for "push-broom" scanning and imaging. Filters in front of the detectors provide images in the three wavelength bands: broadband red, blue-green (BG), and near infrared (NIR).

The detector-chip-assemblies (DCA) housing the charge coupled devices (CCDs) are staggered to provide full swath coverage without gaps. Both the BG and NIR bands have two DCAs each, and the red band has ten DCAs.

The primary mirror has a "dual arch" construction for low mass and high rigidity. The optical system provides a diffraction limited modulation transfer function (MTF) on 12 µm pixels for all 14 HiRISE detectors. The color filters are located 30 mm from the detectors for all three bands. This distance avoids problems due to stray light and multiple reflections from the filters in the f/24 quasi-collimated beam. A Lyot stop, located between the tertiary mirror and the second fold mirror, ensures excellent reduction of stray light.

Distortions in the large field of the red channel are very small. As HiRISE points at the Mars' surface, a point in the image will remain in a single CCD TDI column with no cross column smear for all CCDs.

The telescope structure is made of graphite-fiber-reinforced composite. This produces a stiff, lightweight structure with low moisture absorption properties and low coefficient of moisture expansion. The negative coefficient of thermal expansion (CTE) of the composite elements, in conjunction with metallic, positive CTE fittings, is tailored to produce near-zero instrument CTE. Figure 1 shows the flight structure. Note that the external support structure is the metering system between the primary and secondary mirrors and is temperature controlled with heater mats.

TELESCOPE PERFORMANCE

Two primary mirrors were manufactured. See Figure 3. The first one was distorted when the mounts were bonded in place. The bonding design was optimized and the second mirror was installed into the structure and successful results obtained. The goal for the optical performance in the gravity environment was 0.06 waves r.m.s. at 633 nm and a weighted average across the focal plane of 0.045 waves was achieved with the worse case being 0.053 waves at one side of the focal plane.

Figure 3: Primary Mirror in bonding fixture

HIRISE FOCAL PLANE SUBSYSTEM (FPS)

The FPS consists of the DCAs, a focal plane substrate of aluminum–graphite composite material, a spectral filter assembly, and CCD processing/memory modules (CPMMs). Each CCD has 2048 12×12 um pixels in the cross-scan direction and 128 TDI elements (stages) in the along-track direction. The 14 staggered CCDs overlap by 48 pixels at each end as shown in Figure 7. This provides an effective swath width of ~20,000 pixels for the red images and ~4,000 pixels for the blue-green and NIR images.

Using the TDI method increases the exposure time, allowing us to obtain both very high resolution and a high signal-to-noise ratio. As the spacecraft moves above the surface of Mars, TDI integrates the signal as it passes across the CCD detector by shifting the accumulated signal into the next row (line) of the CCD at the same rate as the image moves (see Figure 4). The line rate of 13,000 lines/sec corresponds to a line time of 76 microsec for 250 km altitude. The pixel line time is set to match the ground velocity so that charge from one image region is sequentially clocked into the next corresponding element in the along-track direction. The imager can use 8, 32, 64 or 128 TDI stages (detector elements in the along-track direction) to match scene radiance to the CCD full well capacity. Spacecraft orientation in yaw will compensate for image smearing during the integration period. A practical limit is reached when residual image smear and spacecraft pointing jitter seriously degrade the required resolution. The 128 lines is the largest number of lines that meets all requirements. Images with higher SNR and lower resolution images will be obtained by binning the signal from adjacent lines and pixels, within the CCD, up to a maximum of 16×16 pixels.

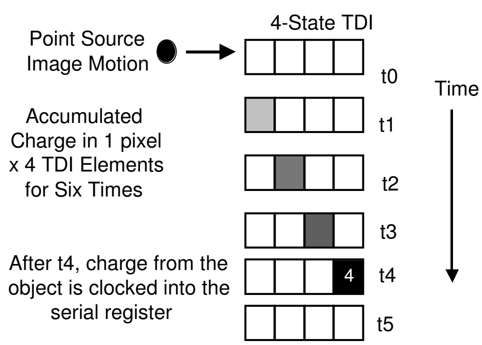

Figure 4: TDI Operation Using a 4-TDI Configuration to Illustrate Charge Accumulation

Assembly pictures of the focal plane are shown in Figures 6 to 10. For additional design details of the HiRISE FPS, see Reference 4.

FPA Geometry

The CCDs were precision located at angles that matched the calculated geometric distortion of the telescope. The measurements of the end fiducials on the CCDs showed less than a pixel error for all CCDs. The vertical locations of the CCDs are shown in Figure 5. Only two CCDs were greater than 80 micrometers from the nominal focal plane. As the focal number is f/24 this error is negligible. Geometric distortion will be measured in flight because the system is not rigid in the earth's gravity. Star clusters Omega-Centauri, M11 and the Plaeides will be observed during cruise to determine the exact geometry of all pixels.

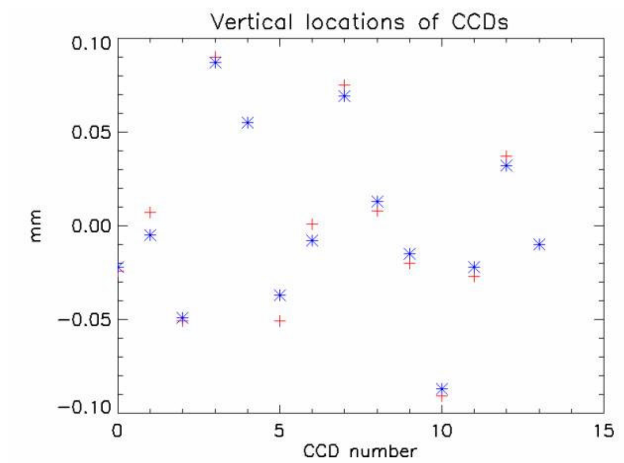

Figure 5: Measurements of CCD vertical position on focal plane substrate. Left +, and right * side fiducials

Figure 6: 14 CCDs mounted to FPA substrate

Figure 7: Mounting of center 6 CCDs, showing custom headers for wirebonds to buffer amplifier boards

Figure 8: Focal Plane Assembly about to be mounted to Focal Plane Electronics

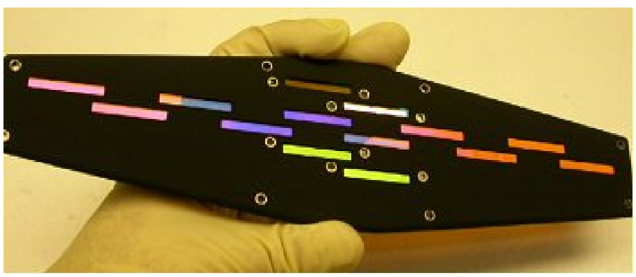

Figure 9: Filter assembly

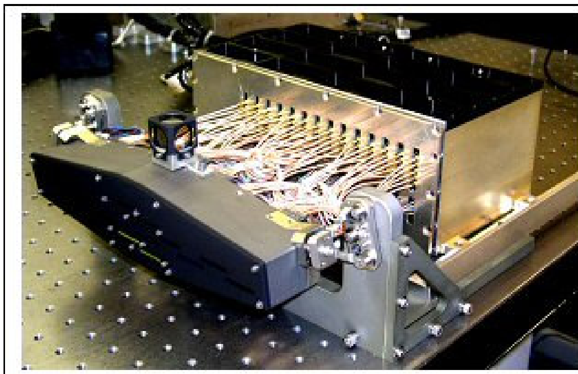

Figure 10: Focal Plane Assembly with filter cover in place

CCD Performance

Noise

The random noise is close to the desired values, 80 electrons r.m.s., against a specification of 90 electrons r.m.s. There is a significant amount of fixed pattern noise that has been demonstrated to be correctable in processing.

Gain

The CCD gain was measured using an Fe_{55} radio-active source. The gain ranged from 19 to 26 electrons per digital number (DN).

Miscellaneous Problems

The CCDs have two problems a slight amplifier light emission and an injected signal into the last gate that is different for both channels. The light emission is comparable with the dark current and easily corrected. The offsets in the two channels, about 1000DN, are only significant when using the lookup table (LUT) to compress the 14 bit data to 8 bits. The instrument design is such that the same LUT must be used for both CCD channels. It is the channel difference in offset that limits the selection of LUT. Some channels are only 20DN difference while the maximum is 200DN.

Practical Limitations of TDI Imaging

Length of integration column

The longer the integration column the better the signal to noise ratio. The maximum number of TDI stages, 128, was set to achieved the required signal to noise ratios. The other TDI stages, 64, 32 and 8, allow for the higher signal levels created when binning is used.
To preserve resolution the image must track exactly down the column. This determines the accuracy of the spacecraft control system. We are using a quarter of a pixel as the requirement to be met. For 128 TDI lines, this translates to +/-2 milli-radians.
The second major consideration is spacecraft jitter that will cause loss of resolution if the amplitude of the jitter is in excess of a quarter of a pixel in the integration time. The integration time is about 100 microseconds times the number of TDI lines used. This means the jitter amplitude must be less than 0.25 micro-radian r.m.s. in 12.8 milliseconds. In the event that spacecraft jitter or alignment degrade resolution a shorter column length can be used at the expense of degraded signal-to-noise ratio.

Reverse clocking

The CCD is cleared of charge by reverse clocking the parallel register. The charge is transferred to a drain at the top of the image section. As the CCD is off for most of the time, The CCD is reverse clocked for at least five seconds on initial power-up. When 64, 32 or 8 TDI lines are used the unused portion of the image area is reverse clocked to remove the surplus charge. The reverse clocking can only remove unto the full well capacity of the CCD. This limits the charge that can be used in the 8 and 32 line TDI cases. In the 8 TDI line case, there are 120 reverse lines that the charge is integrated over, so the maximum signal is limited to twice 8/120 of full well or about 26,000 electrons. For Mars imaging this is not a limitation. Figure 11 shows actual signal measured from a single pixel (column) as a function of line number, which is equivalent to time. Useful image data begins after line 168.

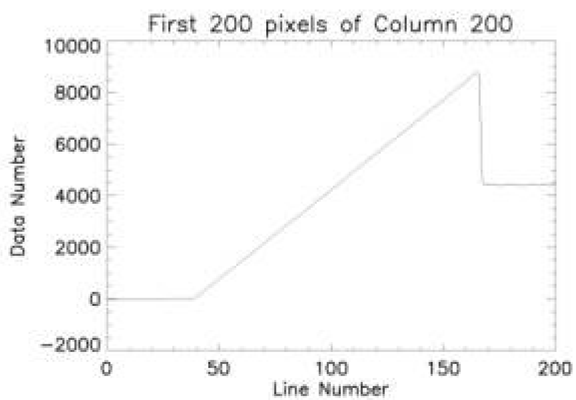

Figure 11: Example of the first 200 lines of a column from a flat field image. First 20 lines are reverse clocked, the next 20 lines are under the mask and the next 128 lines are actual image lines contaminated by reverse clocking.

Imaging Modes

The high resolution mode uses 128 lines of TDI and no binning (bin factor of 1). Because of the limited data return a number of lower resolution modes are available by noiselessly binning pixels in the CCD. The use of different TDI levels ensures that the signal will not saturate the CCD. The combinations of binning and TDI levels are shown in Table 1.

LUTs and Compression

The raw pixel data is 14 bit and is transferred to the onboard memory via a Field Programmable Gate Array (FPGA). Real time compression can be selected in the FPGA via a simple look up table. The lookup table converts the 14 bit data to 8 bits. There are three sources for lookup tables – 28 stored LUTs, uploading a set of new lookup tables and calculating the lookup tables in the instrument controller based on passed parameters. A single lookup table is used for each CCD. (Hence the issue with the offset mentioned above). This compression is performed as the data is transferred to the CPMM memory.

There is another form of compression that can be used as the data is transferred from the CPMM memory to the spacecraft solid state recorder. This is the FELICS hardware compressor that has been added to the spacecraft system**.** This can only accept 8bit data**.**

TDI\Bin	1	2	3	4	8	16
8	All	All	All	All	All	BG
32	All	All	All	All	BG	*
64	All	All	All	BG & NIR	*	*
128	All	All	BG & NIR	BG	*	*

Radiometry Priorities

1	16x16 binning for dark current measurements
5	Low priority
6	Essential
6	Very low priority

Table 1: Imaging modes. All means that all filters can use that combination. BG and NIR mean that only those filter s can use that combination.

HiRISE Camera Electronics Overview

The CCD Processing and Memory Module (CPMM) electronics approach is to minimize the number of active and passive components that contribute to noise. The obligatory analog signal processing chain between the CCD output amplifier and the 80MSPS 14 bit A/D has been designed so that it adds less noise than the CCD, while being radiation tolerant and reasonably low power. The Digital Correlated Double Sample (CDS) scheme is capable of very high pixels rates, in excess of 16 megapixels per second, while sampling a minimum of twice per pixel.
Each of the 14 CPMMs uses a rad-hard Xilinx Virtex 300E FPGA to perform the control, signal processing, Look Up Table compression, data storage & maintenance, and external I/O. The FPGA is SRAM based and uses a Flash Serial Programmable Read Only Memory (SPROM) for configuration upon power-up. The SPROM and FPGA are reconfigurable using JTAG, so design changes during development are extremely simple. The JTAG port is available on an external connector to facilitate last minute design changes, if required.

DATA SYSTEM & OPERATIONS

Target coordinates for an exposure will be uplinked to the spacecraft, which will then translate the target coordinates into the time at which the spacecraft will fly over the area of Mars to be imaged. At the appropriate moment, a block of commands will be executed that will setup-for and then initiate the exposure. Exposure setup parameters include line time, number of lines, binning factor, number of TDI levels, and the lookup table to be used by the CPMM electronics to convert 14-bit pixel data into 8-bit pixel data.

At the prescribed moment, all DCAs begin clocking simultaneously. When the exposure on the last DCA is completed, clocking ceases and the stored pixel data is then readout sequentially from each CPMM to the spacecraft solid-state recorder, in preparation for downlink to the ground. Science data headers accompany the science data so that it can be properly interpreted. Optionally, the pixel data can be compressed by FELICS, a lossless hardware compressor attached to the solid-state recorder.

The nominal high resolution image is 20,000 pixels by 40,000 lines and can take from 4 to 48 hours of transmission time depending on range to earth and compression factors. Details of operations planning and the HiRISE ground data system are found in Reference 5.

HIRISE RADIOMETRIC PERFORMANCE

The predicted maximum signal is 76,000 electrons for the red channel at 300 km with no binning. Figure 12 shows the expected un-binned SNR capability as a function of spacecraft latitude and regional albedo for the blue-green, red (pan) and NIR bands.

OPERATING LIMITATIONS

Spacecraft Disturbances

Spacecraft disturbances may be a severe limitation to high-resolution operation. To help predict performance of the camera before reaching Mars orbit, two tests will be performed to characterize spacecraft jitter effects. A special ground test will use sensitive accelerometers at the instrument interface to measure disturbances caused by various contributors such as the reaction wheels. The in-flight star calibration(s) will be the first opportunity to characterize disturbances in true flight configuration, although at a different TDI line rate.

Operational solutions will help to mitigate the image degradation. MRO solar arrays will be paused during the most sensitive, i.e., high stability observations. It is also possible, though currently not planned, to pause the high gain antenna tracking during the most critical images.

Thermal Considerations

When image data are being collected or transmitted to the Earth, the high power dissipation in both the FPA and FPE cannot be maintained indefinitely under all thermal environments. Limitations on the type and number of exposures per orbit have been accepted to prevent image degradation or possible hardware damage. It is therefore especially critical to validate the instrument thermal model during instrument and spacecraft thermal-vacuum testing.

EMI/EMC

The HiRISE camera has a very robust thermal design to maintain near diffraction limited optical performance. There are 14 operational control zones in addition to the 3 survival zones under spacecraft control. An undesirable feature of this distributed thermal sensor system is the possibility that it will be susceptible to emissions from other instruments, such as the ground-penetrating radar (SHARAD) or other spacecraft subsystems. A special open loop, or duty cycle, thermal control mode was added to preclude loss of thermal control during intervals of high emissions.

In addition, the sensitivity of the UHF transceiver (Electra) on MRO may be degraded by emissions from HiRISE. Extensive filtering and shielding was added to attenuate emissions, however, whether these measures are sufficient will only be determined by spacecraft-level testing. Operational workarounds may require that HiRISE operations

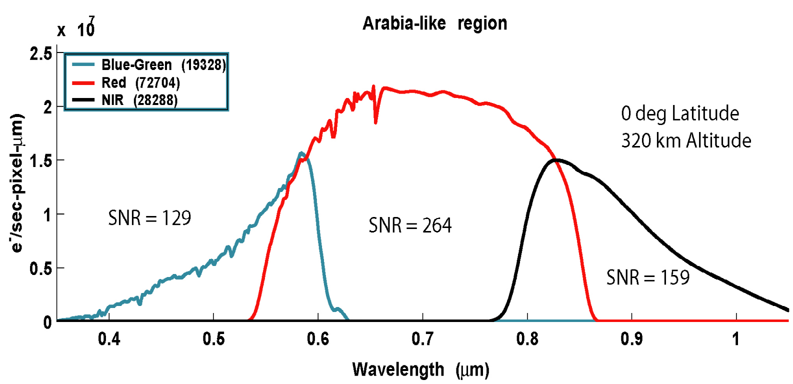

Figure 12: Signal in Electrons as a Function of Wavelength for a Bright Region of Mars

be curtailed when relaying communications from assets on Mars' surface.

Downlink Capacity

The most severe limitation on HiRISE data collection is likely to be spacecraft downlink capacity. Observing scenarios will be optimized through frequent use of LUT compression onboard HiRISE and FELICS compression on the spacecraft. The HiRISE flight software and focal plane electronics have been designed for flexible operation to minimize the science data volume to be transmitted.

Table 2: Subsystem Power and Mass Summary

Item	Orbital Ave Power (W)	Mass (kg)
Thermal Control	35	5
Instr. Controller	13	8
Focal Plane	5	5
Power Supply	7	7
Cables		6
Optics		15
Structure		19
Total	60	65
Allocation	68	65

CONCLUSION

As can be seen in Table 2, the power and mass are remarkably low for such a high resolution camera. HiRISE has set a new bench mark for planetary cameras. In August 2005 the Mars Reconnaissance Orbiter spacecraft will embark on its ambitious journey to explore Mars. The spacecraft will carry an impressive array of instruments including HiRISE, a high-resolution imager that will obtain stunning new images of Mars with a resolution capable of detecting one-meter objects.

ACKNOWLEDGMENTS

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