The X-38 V-201 Flap Actuator Mechanism

Jeff Hagen*, Landon Moore**, Jay Estes**, and Chris Layer+

Abstract

The X-38 Crew Rescue Vehicle V-201 space flight test article was designed to achieve an aerodynamically controlled re-entry from orbit in part through the use of two body mounted flaps on the lower rear side. These flaps are actuated by an electromechanical system that is partially exposed to the re-entry environment. These actuators are of a novel configuration and are unique in their requirement to function while exposed to re-entry conditions. The authors are not aware of any other vehicle in which a major actuator system was required to function throughout the complete re-entry profile while parts of the actuator were directly exposed to the ambient environment.

Introduction

The X-38 Project consisted of multiple unmanned drop test vehicles of various scales and one full-scale unmanned space flight proto-flight Vehicle 201 (V-201). The vehicle shape, derived from the X-24, was a lifting body that lands via a parafoil and skids (Figure 1). The purpose of the project was to perform the development work for an operational International Space Station Crew Return Vehicle.

The configuration, shape, size, and function of the X-38 control surfaces (rudders and body flaps) were chosen to mimic that of the X-23 / X-24 as closely as possible. The X-23 re-entry bodies had fixed rudders and wedge shaped body flaps that replicated only the lower surface of the X-24 flaps while filling the otherwise void area between the top side of the flaps and the lower skin of the aft fuselage. The X-38 flaps more closely resemble the flaps of the X-24 low speed test vehicle, with a void area between the lower skin of the aft fuselage and the top surface of the relatively thin flap. Flight control required the capability to drive the flaps with up to 28,811 N-m (255,000 inch-pounds) of torque - including margins, and to rotate through an angle from zero to forty-five degrees from the lower side of the aft fuselage.

Figure 1. X-38 Family of Vehicles (V131R, V201, V132)

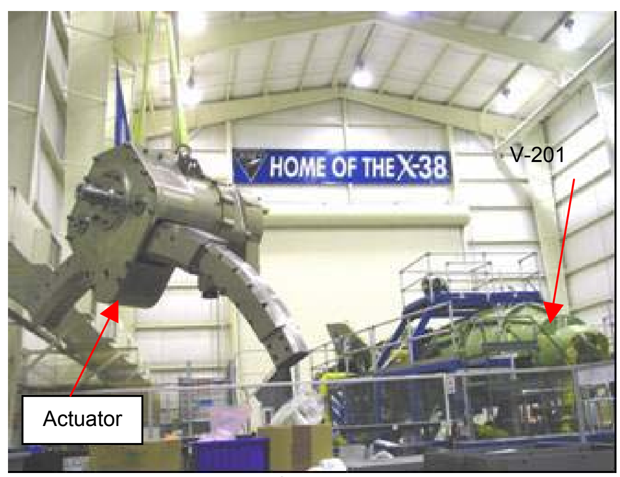

Figure 2. Actuator (foreground) and V201

---

* Lockheed Martin Space Operations, Houston, TX
** NASA Johnson Space Center, Houston, TX
+ Moog, Inc, Motion Systems Division, East Aurora, NY

Proceedings of the 37th Aerospace Mechanisms Symposium, Johnson Space Center, May 19-21, 2004

Due to the exposure of the relatively thin flaps of the X-38 configuration to the full re-entry environment, it was decided to fabricate them from a hot structure material. A conventional cold structure flap covered with thermal protection materials was deemed impractical for the volumetric constraints imposed on the flaps. Furthermore, as a result of the strength limitations of the selected hot structure materials, the X-24 driven geometry of the flaps, and the load environment experienced by the flaps, it was necessary to apply the actuating force to the flaps at a point as near as possible to the center of pressure, which coincided with a location near the center of the roughly square shaped flaps. This geometry dictated that the flap actuator be located in the aft fuselage in the region above the flap and that the linkage connecting to the flap would penetrate through the lower skin of the fuselage to connect to the topside of the flap, thus exposing the actuator linkage to the re-entry environment between the flap and lower skin.

The bulk of the X-38 airframe consists of a cold structure aluminum frame with composite skin panels covered by a thermal protection system (TPS). The flaps mounted on the lower side of the aft fuselage are directly exposed to the flow during re-entry and consequently experience peak surface temperatures that can reach as high as 1649°C (3000°F). The structural layout of the vehicle required that the entire flap fit within a thickness of approximately six inches, which would have been insufficient for a conventional cold structure flap of sufficient stiffness and covered with a TPS of the requisite thickness. Consequently, the flaps were constructed of a hot structure carbon silicon carbide (CSiC) ceramic matrix composite without a TPS. MAN Technologie of Germany supplied the flaps via a European Space Agency / TETRA (TEchnologies for future space TRAnsportation systems) sponsored joint venture with NASA.

To accommodate the interface of the hot structure flaps to the cold structure airframe, MAN developed a high temperature journal bearing material based on CSiC, and designed a system of CSiC beams to support the flap at the hinge line bearings while accommodating the differential thermal expansion ratios of the hot and cold structures. The hot structure beams were surrounded by thermal barriers at the points where they penetrate the TPS of the cold structure. This prevented impingement of re-entry gases on the cold structure. In order to simplify the hinge line interface seals, the flap hinge line bearings and supports were located outside the airframe TPS such that the seals are entirely static and do not flex during rotation of the flaps.

Since the X-38 was an unpowered glider intended for long-term storage on orbit, it does not have an auxiliary power system or hydraulic system such as the Space Shuttle. All of the flight control surfaces are actuated with electromechanical actuators (EMA) that are driven by direct current from Nickel-Cadmium batteries. Prior development programs (EPAD et al) indicated then state of the art EMA technology could be successfully applied as a design solution for X-38. Further confidence was gained through X-38 drop testing (V132, V131R) performed at Dryden Flight Research Center using single string, linear output EMAs. At this point, it was evident that the available pulse-width modulated, brushless DC motor and motor drive technology was adequate to drive the type of power needed to control the V201. The technical concerns shifted to the optimum architecture that should be employed to facilitate human rating. The human rating requirements drove the system to be fully two-fault tolerant meaning that all flight control system requirements for load and rate be met even after two failures. The most fundamental architecture redundancy decision involved the choice between an active-standby system and an active-active system. It was clear that an active standby system would be simpler in terms of software and control, but would result in a more extreme control surface transient during a channel failure. The active-active system promised reduced failure transients at the expense of sophisticated control software that would marshal four highly responsive channels operating simultaneously while forced into lock step by being torque summed through a bull gear. Ultimately, the latter path (active-active) was chosen and successfully developed.

Design Evolution

The original actuator design consisted of a long linear ball screw with the top end anchored to a spherical bearing near the top of the aft fuselage structure and attached at the lower end to the high temperature spherical bearing near the center of the flap (Figure 3). Due to the kinematic layout, the linear actuator would have penetrated the skin at a dynamically changing location, complicating the thermal seal design.

It was assumed that the use of a spherical bearing at each end would prevent bending moments from being applied to the actuator. Aerodynamic analysis determined that skin friction drag parallel to the flap surface contributed only negligible loads to the flap system. The required flap driving moment translated into force of 44,482 N (10,000 pounds) at the chosen EMA attachment point if the maximum flap force were to be experienced.

The concept selection and the kinematic layout were frozen before the detailed design of the actuator or vehicle structure were fully understood due to the long lead time required for fabrication of the flaps. In particular, no satisfactory concept or technology was identified for the thermal seal where the actuator penetrated the fuselage skin. Accommodating such motion within the extremely limited volume and while exposed to the demanding environment led to design problems for the thermal seal that were never satisfactorily resolved for this configuration. Also, as design of the linear actuator matured, it became apparent that insufficient clearance existed between the lower part of the actuator and the fuselage lower longerons, which were already in production. Even after less than optimal modifications to the fuselage structure, there was no room left for growth in the actuator size or for accommodating an adequate thermal seal based on available technology.

A more detailed concept eventually emerged that consisted of a telescoping tube surrounding the actuator lower rod and mounted in a ‘shoe’ that could slide over a slot cut in the fuselage. However, even after extensive surveys of industry, no suitable technologies were found for implementing this sliding and telescoping thermal seal. The principle problem was finding materials capable of surviving the necessary temperature within the extremely limited space available and of accommodating the complex mechanical motions in the high temperature, corrosive environment.

During this time, technology development issues involving the high temperature CSiC bearings of the flap arose. Testing revealed that the bearings exhibited a far higher than anticipated coefficient of friction. This resulted in large bending moments being applied to the linear actuator for which it had not been designed. Reports from MAN Technologie indicated that the bearing coefficient of friction could exceed 0.7 or more. Additional testing later failed to show any meaningful relationship of the coefficient to load, speed, temperature, or other environmental and application factors. Efforts to reduce the friction by modifying the surface treatment or finish also failed to generate usable improvements. MAN Technologie informally reported promising indications of success with a new type of proprietary surface treatment, but the project schedule could not accommodate the necessary technology development effort.

The unexpectedly high friction of the bearing meant that rather than isolating the EMA from bending moments, the bearing could actually apply a large moment at the lower end of the linear actuator EMA. Due to the volumetric constraints on the actuator caused by the layout of the flaps and aft fuselage structure, the EMA configuration was forced to a slender design with a very high length to diameter aspect ratio that could not support bending loads well.

The newly imposed bending moment could only be accommodated with a substantial increase in the diameter of the lower link rod, thus driving up the overall size and weight of the EMA. It would also have been necessary to increase the power of the EMA to meet requirements under the additional load. However, because of the above volumetric constraints, there was inadequate room for the necessary expansion of the EMA and thermal seals. It thus became apparent that the EMA concept required a fundamental redesign. During this time the NASA EMA team began to devise alternative concepts in preparation for the anticipated necessity of changing the design.

Existing aft fuselage primary structure and flap design severely bounded alternative EMA layout concepts. All new design concepts also had to address the difficult thermal seal design problem. The development time line was also a driving issue, as the EMA redesign effort came about very late in the over-all project schedule. This precluded selection of designs or methods that appeared to require a substantial technology development effort.

The first concepts for a new EMA layout resembled the V-132 / V-131R atmospheric test vehicle flap actuators. They were based on a linear ball screw EMA in the aft fuselage, but mounted remotely from the flap, and drove the flap through a four-bar linkage. Only the final link rod would penetrate the fuselage skin and attach to the flap at the same location as the original design (Figure 4). Several variants were explored, trading EMA size, power, location, and minimizing rotation of the final link rod relative to the fuselage lower skin or the flap. Initially, it was hoped that isolating the EMA itself from the flap bearing would eliminate the problem of designing a linear ball screw to operate under a substantial bending load and that penetrating the lower skin with just a simple link rod would increase the space available for the thermal seal and reduce the insulation requirement for the lower rod.

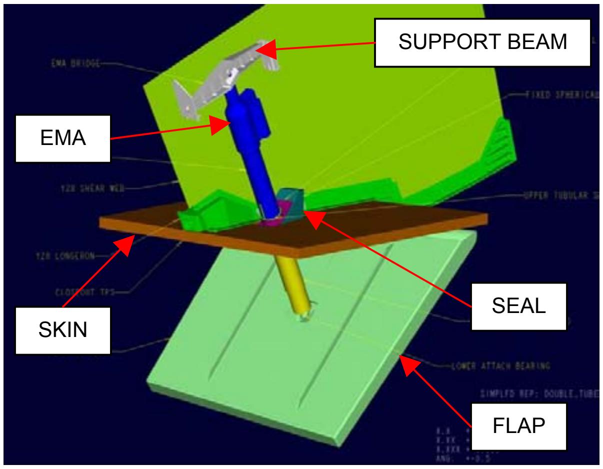

Figure 3. Original linear EMA concept.

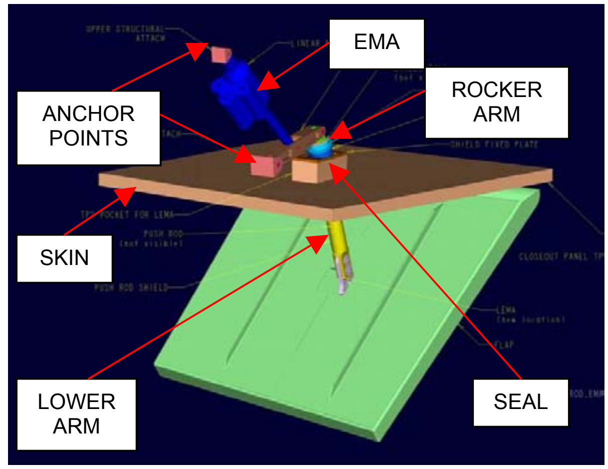

Figure 4. A conceptual 4-bar design

Unfortunately, the various factors did not trade very well. Reducing the amount of rotation required from the flap bearing increased the amount of movement relative to the skin and thus increased seal complexity. Reducing the amount of movement relative to the skin increased the amount of rotation at the bearing. Most of the configurations drove the final link rod to a more disadvantageous angle with the flap, which increased the loads on the flap bearing and the flap hinge bearings to potentially unacceptable levels. Another configuration minimized the rotation at the flap bearing and kept a better angle between the link and flap, but drove the EMA power requirements quite high and pushed it so far up and aft as to be of serious concern to the overall vehicle center of gravity. Additionally, this greatly complicated the thermal seal. Estimates ultimately indicated that all of the configurations would involve a significant weight growth, possibly even double the original design. Stabilizing the complex actuator and linkage train against out-of-plane loads would have introduced additional complexity and weight, which was not accounted for in the conceptual designs and which may not have been achievable. Finally, most of the layouts resulted in a link rod size suitable for the bending loads and relative motion that actually exacerbated the thermal seal design problem. Ultimately, the four-bar linkage concepts failed to address the basic problem of requiring large rotations between the actuator linkage and the flap through the high-friction CSiC bearing.

New Design

Eventually, it became apparent that barring a breakthrough in high-temperature bearing technology, the best chance for success required an actuator concept that eliminated the need for relative rotation between the final actuator link and the flap. Simplification of the relative motion between the actuator and the fuselage lower skin was necessary in order to develop a realistic thermal seal design using available technology. There also remained the requirement for any new actuator design to be compatible with the existing fuselage and flap structures.

<u>Actuator Layout</u>
The most obvious method of achieving the first two goals was to shift the center of rotation of the EMA from inside the fuselage to an axis coinciding with the axis of rotation of the flap, but without requiring any

additional penetration points in the fuselage skin. This was accomplished by changing the final actuator linkage rod to an arc shape analogous to a circumferential segment of a wheel with its center of rotation located on the hinge axis of the flap (Figure 5). The portion of the flap between the hinge bearing and the arc attachment forms one ‘spoke’ of the ‘wheel’ and the fuselage structure between the flap hinge mounts and the anchor points of the EMA forms a second ‘spoke’, which allows freedom of movement of the ‘wheel segment’ (arc) in the tangential direction.

Linkage

The actuator drive unit transmits power to the circular segment link through a pinion gear that drives a gear segment integrated into the periphery of the circular segment. The drive unit consists of three motors that simultaneously apply redundant power to the pinion gear through a cycloidal transmission. The drive unit holds the arc through a series of ‘V’ guide rollers that are integral to the drive unit. Two of the rollers are mounted on the same side of the arc as the pinion and to either side of the pinion, and the third roller is mounted directly opposite the pinion. The third roller is mounted on an eccentric cam that can be adjusted to provide the proper compression of the arc between the opposing rollers and thus minimize free-play without inducing too much friction. The pinion gear is mounted on an adjustable Belleville spring stack to maintain a constant radial force between the pinion and sector gear teeth (Figure 6). The only radial connection fixed relative to the circular segment consists of the flap itself. The drive unit is mounted to the fuselage through a bearing mounted trunion at each end of the axis through the pinion gear to form the other ‘spoke of the wheel’ and fully constrain the actuator location.

Figure 5. Flap Actuator Assembly Before Installation

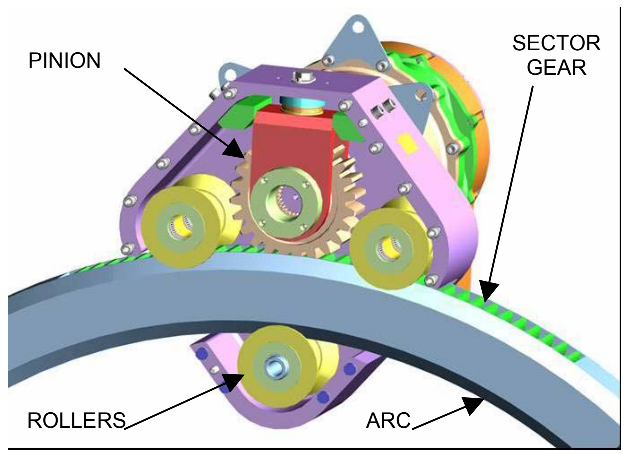

Figure 6. Arc Pinon Gear Drive Assembly

Since the drive unit is mounted completely inside the hull, and since the flap (including the hinges and actuator attach bearing) is mounted completely outside the hull, only the arc penetrates the hull. As the actuator arc moves in and out it remains fixed relative to the flap (since the flap acts as a radius of the circle that the arc moves through). The movement of the arc relative to the skin at the hull penetration is greatly simplified since the circular path of the actuator remains fixed relative to the hull. Since the portion of the actuator that passes through the hull has a constant cross section the seal has a static shape and remains fixed to the hull.

Deflection and Thermal Expansion Relief

Due to differences in the coefficients of thermal expansion of the ceramic flap and aluminum hull structure, the two radial connections to the actuator experience small relative changes through the re-entry profile. By mounting the actuator drive unit on an axis that is free to pivot slightly relative to the vehicle structure, the actuation system can still function even as the two radial fixations vary slightly with respect to each other. Such variations cause a slight displacement of the effective center of the actuator circular segment from the flap hinge line. As the flap rotates, the displacement of the centerlines causes a small rotation of the drive unit about its pivot axis in order to maintain the tangency of the drive unit guide rollers and the circular segment link. This rotation is permitted by allowing the circular segment to rotate relative to the flap at the connection points by very small amounts, which are within the capability of the high temperature bearing. This also causes a small lateral motion of the arc's circular path relative to the skin at the hull penetration, which is accommodated by letting the seal mounting ring float a small amount parallel to the hull skin.

Redesign Constraints and Issues

Due to the pre-existing fuselage structure and flap geometry, major structural elements interfered with the most logical actuator locations, which caused the already large loads on the actuator to be substantially increased. Because of the canted angle of the flap hinge axis and the necessity of keeping the arc axis parallel to the hinge axis, the arc and its path of rotation were canted relative to the fuselage frames. With the actuator attachment point of the flap located under some of the fuselage primary load paths, it was not possible to locate the arc in an orientation that allowed its lower end to match up with the flap attach point. The flap team was able to make some minor modifications and re-orient the assembly of the support structure for the attach point bearing to move the attach point slightly outboard. An offset foot was then designed into the lower end of the arc to shift the attachment point of the actuator inboard sufficiently to connect with the flap. The offset resulted in large out of plane bending loads applied to the arc, which required the arc to be designed with a generous cross section.

This foot became the interface component between the metallic actuator arc and the CSiC flap bearing plates. The actual interface was accomplished with two large diameter, hollow core CSiC pins that connect the metal foot to the CSiC bearing plates that ride on the CSiC bearing of the flap attach point. One of the two holes on the foot side was slightly slotted towards the other hole to allow for the different thermal expansion rates and for manufacturing differences. Both CSiC pins were fitted snugly into the foot holes with individually cut to fit bushings made of PM1000 oxide dispersion stabilized high temperature alloy from Plansee Aktiengesellschaft of Austria. The bushings eliminated concerns of possible chemical incompatibility between the CSiC and Inconel at elevated temperatures. The bushings also provided an additional break in the conduction path and limited the size of the conduction contact patches, which proved important in reducing the amount of heat conducted into the foot by the hot flap. Thin sheets of woven 3M Corporation Nextel ceramic fabric were placed between the non-load bearing adjacent surfaces of the bearing plates and foot to provide additional thermal protection.

In order to carry the large loads while exposed to the very high temperature and highly oxidizing environment, the arc was fabricated from Inconel 718 super alloy and protected by a refractory metal shell. The offset foot was also fabricated from Inconel 718, but was protected by a semi-rigid fabric blanket composed of layers of woven Nextel ceramic fabric, ceramic batting, and Inconel foil.

The portion of the arc exposed below the thermal protection of the fuselage lower skin could reach surface temperatures well in excess of the capability of Inconel to support any meaningful load. Most of the heating resulted from radiation of heat conducted through the thickness of the flap and reflected off the lower fuselage thermal tiles. Also, the recirculating flow in the cavity between the flap and the lower fuselage skin was sufficient to potentially destroy the Inconel by oxidation. Accordingly, the exposed portion of the arc was protected by TPS. Originally it was planned to use the same blanket system that protected the foot, but testing revealed that under operational conditions it would not maintain sufficient strength to resist abrasion as it passed through the thermal seal. The ultimate concept was to wrap the lower arc with a thick layer of ceramic batting insulation that would be surrounded by a thin shell made of Hitemco R-512E silicide coated Oremet Wah Chang C-103 columbium alloy.

The lower part of the Inconel arc was necked down and the external cross section of the shell was identical to the cross section of the upper part of the arc so that the semi-rigid thermal seal would maintain contact with the arc as the flap rotated. The insulated region of the lower arc was bounded by the location of the lowermost drive unit v-guide bearing when the flap was fully retracted, which limited the flap to approximately half extension during the peak of re-entry heating. This met flight control requirements, with the exception of possible very brief extension of the flap to a sufficient angle to expose a very small region of the bare Inconel to thermal radiation. Analysis showed that with a sufficiently polished surface the Inconel could manage these limited exposures. Overall, the peak heat load would have generated a very large thermal gradient between the lower end of the actuator and the more temperature critical elements of the drive unit at the upper end. This gradient was achieved primarily by constructing the arc and foot from ample amounts of Inconel, which acted as a very effective heat sink such that the brief heating period was insufficient to achieve anything close to temperature equilibrium.

The flap and actuator together comprised a system with indeterminate load paths, so the loads applied to each component were determined by applying the flight load to a finite element model of the complete system. Analysis of the actuator components ultimately required a very high fidelity finite element model of the complete arc and foot assembly and took considerable computation time. Applying the flap driving load tangent to, and through a curved beam rather than the original straight beam caused an increase in the total load. The increase resulted from the load vector being closer to parallel to the flap and thus 'loosing' the effectiveness of that portion of the load vector. (The actual EMA load vector was determined by the pressure angle of the pinion gear teeth on the 'arc' sector gear teeth.) The extra load applied in the direction parallel to the flap increased the radial loads felt by the flap hinge bearings, exacerbating the already serious friction problem of the CSiC bearings. Moving the drive unit location farther away from the flap caused the load vector to be more parallel to the flap and worsen the load conditions, but moving the drive unit closer to the flap reduced the length of the arc that could be insulated and thus reduced the amount of flap extension available during the peak re-entry heating. Locating the drive unit attach points far from elements of the fuselage structure applied loads that exceeded the capability of the structure, but locating the attach points close to the structure required clearance cut-outs that weakened the structure unacceptably. After and extensive period of iterative design and analysis, a compromise location for the drive unit was determined that kept the radial loads just within the limits of the flap hinge bearings, allowed just enough room for an insulated section of arc long enough to achieve the necessary flap travel, and kept the structural loads just within acceptable limits without causing any uncorrectable interferences with the existing structures. Fortunately the existing structure was able to accommodate a workable solution.

Although recirculating flow of hot gas between the flap and the lower fuselage skin caused oxidation problems for actuator components, the contribution of convection to the overall heating was fairly minor. Most of the heating in the actuator resulted from radiation from the hot structure flap and lower skin thermal tiles, and from conduction through the foot. With the large masses and thick sections of most of the parts, and the brief duration of maximum heating exposure, the total heat input to the parts was of more importance than particular surface or environment temperatures. The complex nature of the actuator design and its operating environment, along with the transient nature of the thermal state, required a separate high-fidelity finite element model for thermal analysis. Because many portions of the actuator reached temperatures that caused significant weakening of the materials, and because the thermal conduction between parts was largely dependent on the contact pressures between them, it was important to maintain an iterative loop between the thermal and stress analysis. Each of the finite element models required different simplifications, but both were derived directly from the detailed design models by direct computer model translation to achieve sufficient fidelity.

In order to minimize the extra load introduced into the flap (especially launch vibration loads) and to minimize the risk of jamming, the arc needed to be very stiff. Achieving sufficient stiffness was difficult in a curved, end-supported beam with a high length to cross section ratio (a good portion of which was substantially necked down) and which was subjected to a large tangential load. Additionally, the resulting high stresses had to be carried at temperatures that could exceed 538°C (1000°F) degrees in some locations. The result was an integrally machined beam composed of thick sections and dense internal webbing to minimize the length of unsupported spans, especially in the region where the guide rollers applied concentrated loads. (The rollers were slightly crowned to minimize friction caused by slipping as the tapered rollers contacted the angled surface with a radius that varied along their length, which reduced the size of the contact patches.) It was also necessary that the bearing surfaces of the completed arc maintain a precise and constant cross-section and have a very smooth surface to ensure proper passage through the rollers when under load. In order to maintain sufficient strength (roughly 80% of room temperature in the worst areas) and oxidation resistance at the elevated temperatures, Inconel 718 super alloy was chosen. Fabricating the large, complex, precision parts from very hard and difficult to machine material turned out to be quite time consuming and caused extensive, unanticipated delays.

A fully articulated, solid, parametric computer model of the flap and actuator system was constructed with all of the components modeled in complete geometrical detail (Figure 7). The complete model could be moved through the full range of flap rotation and deflections could be introduced that represented thermal expansion of the flap support beams and the main flap in both the longitudinal and vertical directions. By constraining the elements of the design model similarly to the actual hardware, it was possible to verify correct functioning of the design even with the main components in a deflected state before any hardware or mockups had been constructed. Essentially, the drive unit rotates about its mounting axis to maintain the tangency of the pinion gear and sector gear regardless of any shifts in the location of the flap hinge axis. Any such shifts cause the axis through the center of the arc path to no longer coincide with the flap hinge axis, which results in a small change in the normally perpendicular angle of the arc to flap connection. As the flap moves under this condition, the angle between the arc and flap will change minutely and within the limits of the bearing capability. Using the computer design model, it was possible to determine the actual amount of movement at the bearing with the system in a thermally deflected state and verify that the movement was acceptable. The computer model was also used to find and eliminate sources of interference and to verify adequate assembly access in the vehicle. Finally, the computer model proved critical for real-time determination of as-built component locations in the vehicle during the initial integration of the flaps and actuators. Without the articulated 3D model, it would have been very difficult to properly align the parts of the mechanism.

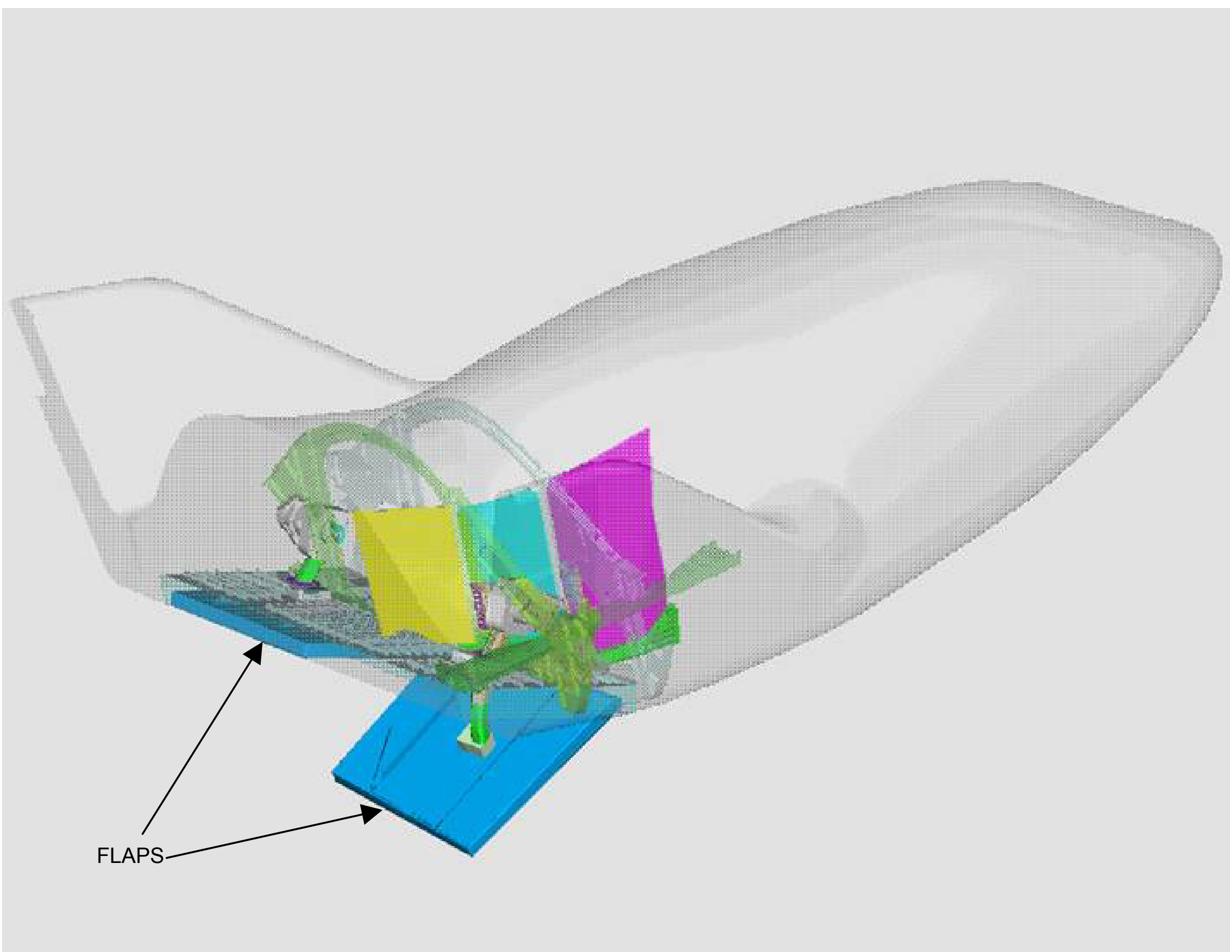

Figure 7. Vehicle Integration CAD Model

The gap formed between the arc and the skin at the location of the penetration through the fuselage required plugging with a thermal barrier or ‘seal’ to prevent the ingestion of hot gas into the aft fuselage (Figure 8). The seal was mounted to the fuselage skin and the arc slides through the seal as the flap rotates. It was comprised of two loops of conventional Nextel spring tube seals stacked one above the other for redundancy, increased footprint, and decreased permeability. (The spring tube seals consist of a loose Inconel wire mesh shaped into a tube, wrapped with braided Nextel ceramic fabric, and filled with Saffil ceramic batting.) The tubes were stitched to the Inconel mounting plate with Inconel safety wire and were also captured between a lip in the fuselage skin thermal protection tiles and a lip in the top of the mounting plate. This design was chosen both for its ability to withstand the temperature and oxidation conditions, and because it provided a soft contact surface with the arc, which was critical to protecting the ceramic coating of the lower arc thermal protection shell from abrasion damage. However, it was also necessary for the sector gear teeth to pass through the seal for the flap to achieve full extension during testing and post re-entry flight. To prevent abrasion damage to the seal by the gear teeth, the portion of the seal contacted by the teeth was covered with an Inconel plate or 'shoe' that covered both tubes and prevented the spring tubes from protruding into the gear teeth. To prevent the 'shoe' from abrading the ceramic coating of the lower arc, the assembly of both spring tubes and the 'shoe' was wrapped with a layer of woven Nextel fabric, with the 'shoe' and over wrap held in place by the Inconel stitching. The seal mounting plate was fixed to a small, removable section of the fuselage skin to aid installation and was mounted with a flange captured between the skin section and a capture plate such that it could slide slightly in the plane of the skin to accommodate dislocations of the arc path caused by thermal expansion and aero loading of the system (Figure 9).

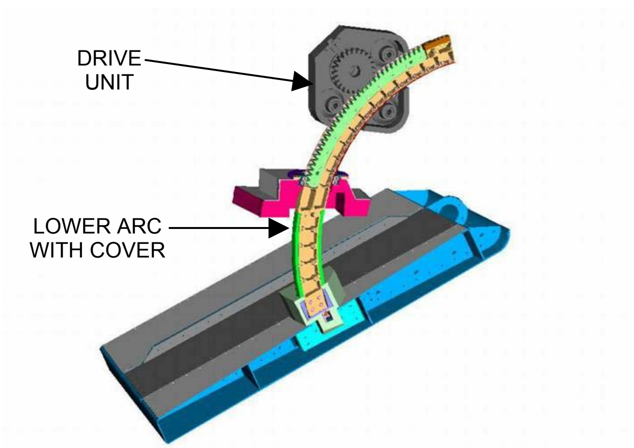

Figure 8. Section through Flap and Actuator System

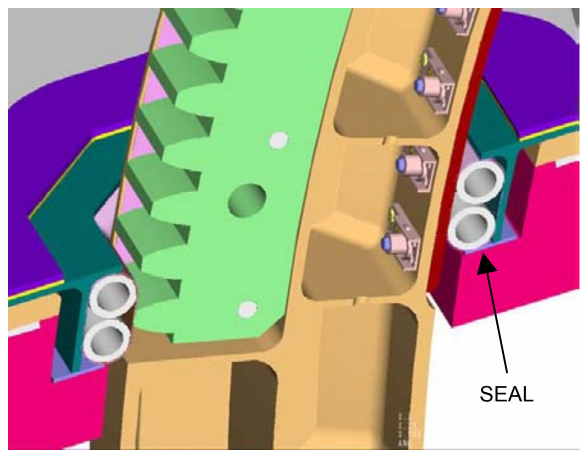

Figure 9. Close-up Section through Thermal Seal

Actuator Drive Unit

The CRV Body Flap Actuation control system (actuator & controllers) was designed, built and qualified by the Systems Group of Moog Inc (Figure 11). It was one of the first space-rated, dual-fault tolerant, electromechanical primary flight control actuation systems.

The actuator was designed to provide full torque and speed performance after any two electrical or controller channel failures. Drive train mechanical redundancy was not provided due to the weight impact and the effect of additional complexity on overall reliability. Instead, large margins of safety and failsafe designs were integrated into the individual components to overcome the reliability concerns.

As shown in the schematic (Figure 10) three redundant motors were torque summed around a common bull gear; which input into a cycloidal drive. The system was designed such that in the event of a motor channel failure the other channel(s) would drag along the failed motor. This eliminated the need for clutches and the problems associated with implementing such devices. The output of the cycloidal drive in turn was connected by a floating crowned spline shaft, which drove the sector pinion gear and subsequent sector gear arc of the flap. The overall actuator drive ratio was 2486.6:1.

Three brushless DC, rare earth, samarium cobalt magnet motors provided power. The rotors were banded with steel sleeves to prevent a rotor / stator jam due to possible failed magnet retention since peak operating speeds of 15300 rpm were possible. A cycloidal drive transmission was chosen because of its low profile, high stiffness, low backlash, high efficiency and drive train robustness. The cycloidal drive has a large number of “lobe” teeth in contact, which minimizes the operating stresses and makes it ideal for high reliability applications. A crowned spline dog bone shaft bridged the output of the cycloidal transmission and pinion gear. This was required because the pinion gear was spring loaded into the sector gear and allowed to float vertically in order to eliminate backlash and absorb thermal variations within the gear mesh.

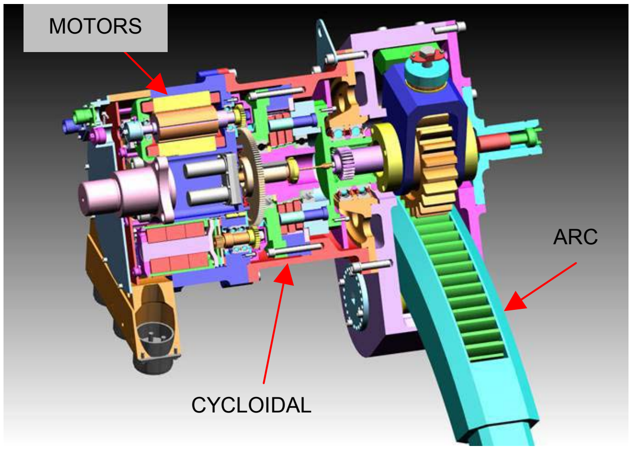

Figure 10. Cutaway View of the Flap EMA Assembly

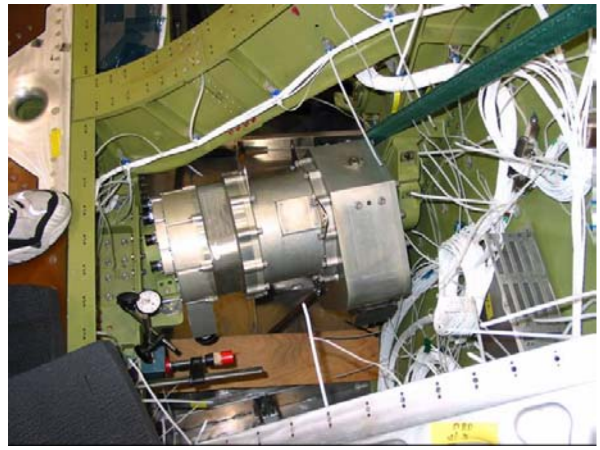

Figure 11. EMA installed.

The offset attachment of the foot to the flap created significant moment loading on the arc that raised concern over gear mesh alignment. To counteract this effect the sector gear was mounted inside an Inconel arc that was supported by three v-shaped conical rollers in a tripod arrangement inside the support box. An eccentric bearing cam pin in the bottom roller was then adjusted to minimize the radial play of the sector arc in relation to the three rollers. The pinion gear was also crowned to accommodate any additional misalignment problems.

The drive unit was mounted on trunions along the axis of the pinion gear. This mounting configuration allowed the actuator to rotate due to the thermal variations in the actuator and flap mounts.

The flap position was monitored by a geared, dual-tandem, brushless resolver (1 model channel / 3 position) that was connected to the pinion gear via a quill shaft arrangement. A triply redundant fail safe brake held the actuator in position during shuttle transport and provided the last defense of control by snubbing the energy and locking the flap in a fixed position in the event of an all out power failure. The brake engaged the same bull gear as the drive motors. The brake incorporated friction plates coated with plasma sprayed chromium oxide for stable operation in a vacuum environment. Dual mechanical limit switches attached to the output of the cycloidal drive protected the actuator when a single channel was active as well as providing an over travel limit.

Vehicle Integration and Testing

Integration of the actuation system with the flaps and completing the associated vehicle attachment proved to be a very challenging process. The process required integrating the flaps which are very large delicate, and expensive, to the vehicle. The axis of rotation of the flaps had to be positioned to extremely tight tolerances, and the rotation needed to be on an axis that was precisely parallel to the axis of the EMA rotating mount. The combination of tight tolerances, close clearances, extremely heavy EMAs (approximately 136 kg (300 pounds) each), and precision alignment provided unique challenges to the integration team.

The total weight of each flap was approximately 61 kg (135 pounds). For their size they are relatively lightweight, but due to the special processes involved in fabrication of them, special handling procedures were required. Additionally, there are several coating processes that are performed on the ceramic that are sensitive to oils, and gloves are required for handling. Special drilling tools are required for drilling ceramic parts. Each body flap cost approximately 14 million dollars to design, fabricate, test, and qualify for space flight. Therefore, great precaution was taken to preclude damage to the flight units.

The scheme that ultimately proved successful for integration of the flaps and EMAs to the vehicle was a "top-down" installation. The EMAs are dropped into place from open access on the top of the fuselage. Precise positioning of the EMA axis was first performed to CAD specs. Using a laser system, a special tool was employed to measure and position the axis of the EMA rotating mount. The positioning of the EMA was based purely on CAD layout drawings. While the EMA mounts were adjustable, adjustments were very difficult with the unit place due to the weight and close clearances with primary structure.

Once the EMAs were positioned as close as possible to CAD specifications, an adjusted flap hinge axis location (relative to the vehicle coordinate system) was derived based on the measured EMA axis. Next, physical positions on the flap (co-linear with the hinge-line) were determined using laser tracking. In order to do this, an empirical flap hinge axis was measured based on actual movements of the flap hinges. This empirical flap hinge axis was then positioned relative to the desired hinge-axis positioning on the vehicle.

In order to accommodate various manufacturing tolerances and tolerance build-up, the flap attachment to the vehicle was adjustable in six degrees of freedom at both hinges. Final alignment of the flaps was made through a trial and error combination of special flap positioning tooling and temporary clamping procedures (Figures 12 & 13). The resulting relative angle between the flap hinge-line and the EMA hinge-line was less than approximately 0.05 degree.

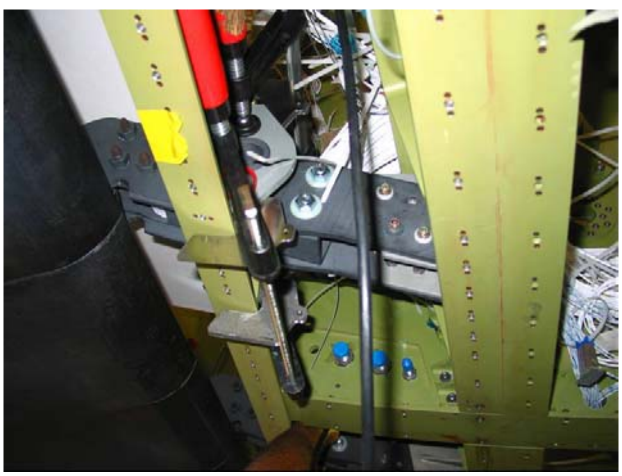

Figure 12. Flap Hinge (aft to left)

Figure 13. Flap Hinge Mounting (aft to right)

Once the alignment was finalized, manual test sweeps of the flap through full angular excursion were made to establish proper integrated function of the flap with the vehicle. With proper positioning established, the flaps were matched-drilled to the vehicle with under-sized holes (to allow potential future fine position adjustment), and fastened temporarily.

The final phase began by attaching the EMA arc to the flap at the foot. This process involved special tooling to protect the flap and aid in the positioning of the heavy moving parts of the EMA. Final mating of the arc to the foot and the flap was followed by manual movement of the entire integrated EMA, arc, foot and flap system while bolted to the vehicle. With successful full-range movement established via these manual sweeps, the temporary fasteners were secured in preparation for powered avionics tests.

Finally, with all parts precisely aligned, checked, and bolted together, the entire vehicle avionics system was powered up, flight software was loaded into onboard computers, and computer controlled movements were conducted. Onboard software provided test-movement sequences similar to those that would have been used in flight. The computer controlled movements included moving the flaps up and down in prescribed sine-wave motions as well as simple step-commands. This testing verified a full functioning, multi-string on-board avionics flight control system with vehicle hardware.

The time from hardware delivery prior to first EMA install to conclusion of full-power avionics testing was approximately six weeks. TPS installation, final drilling of holes for fasteners, and final installation of the flaps were not accomplished before project termination.

Each surface actuation subsystem had four EMA controllers, each controlling three electric motors housed in the EMA subsystem unit. All four controllers are on-line at the same time, and each is talking to each of the three independent motors powering each flap EMA with motor synchronization codes in the controller preventing motors from fighting one another. So for the test with two flaps there were eight controllers and six motors up and running - all cooperating with one another.

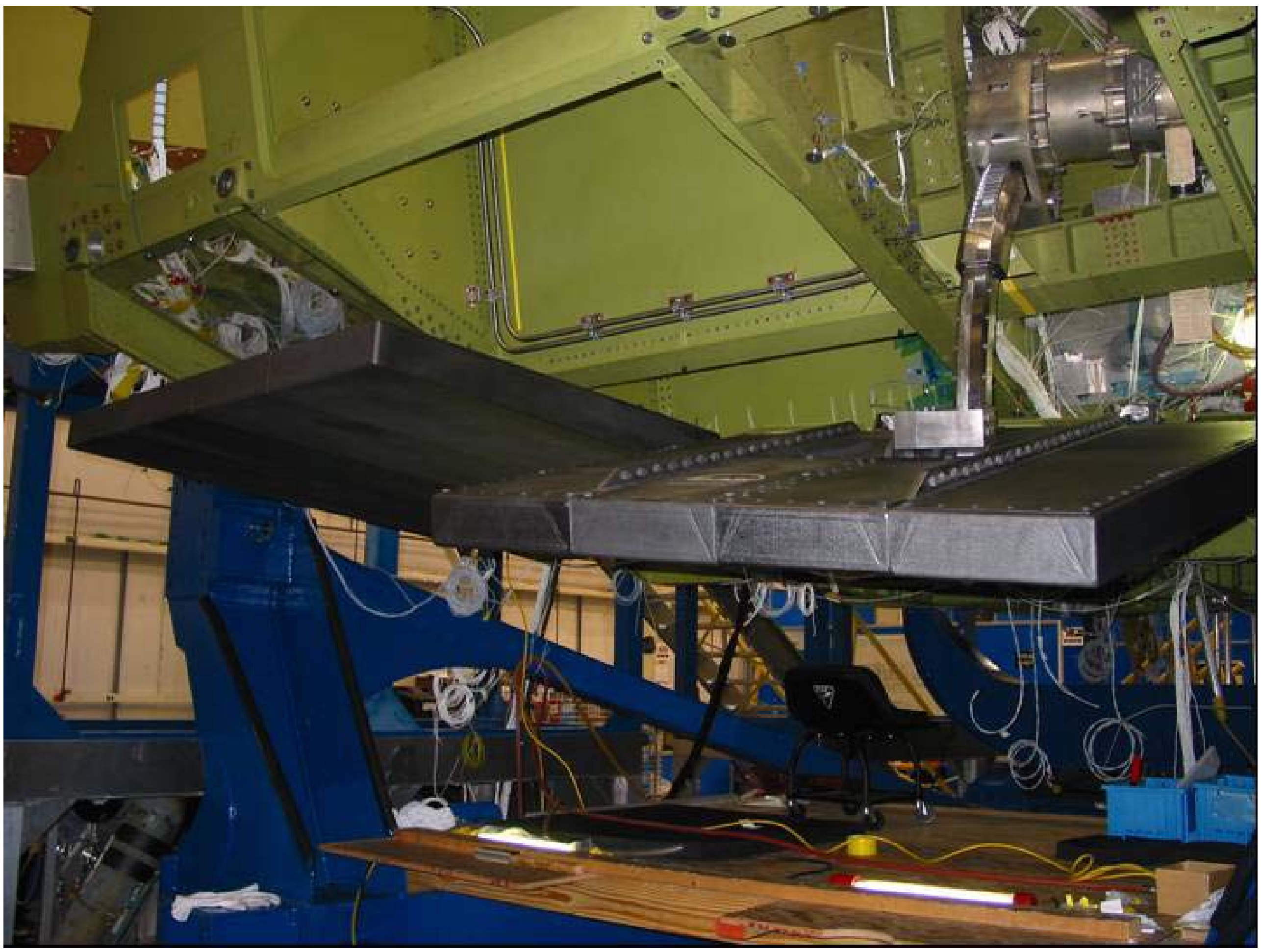

Figure 14. Completed Installation (Starboard Flap Deployed)

The system is capable of having multiple failures of controller components and or motor components, and still functioning fully for flight. It is worth noting that this EMA design is unique in that no other redundancy system for control surfaces provides all motors up and running simultaneously. With other systems, other motors are present, but powered off until needed. With this design, everything there is being used, but if a component fails, the system continues to function and meet requirements with what is left running.

Status

The actuator mechanical units for the flight test vehicle have been fabricated and installed (Figure 14). Some functional testing was successfully completed with the unit installed in the vehicle before the project was terminated. A flight-design actuator qualification unit was successfully tested by the drive unit contractor, Moog, in a loaded test stand that simulated the full aerodynamic load. Initial component level testing of the seal showed promising success, but later testing indicated some seal abrasion problems that remain to be resolved.

The thermal seal system and actuator thermal protection system reached only component level prototype fabrication before the project was terminated. Detailed design of the thermal protection shell for the lower arc and of the seal mount to fuselage skin interface were not completed. The flight unit flaps were installed on the vehicle, but without using the flight design fasteners. The vehicle lower skin was temporarily removed for integration of the flaps and actuators, but was never reinstalled, which would require temporary removal of the flaps and actuators. The flight unit controllers were used for testing, but with temporary mounts and wire routings.

Conclusion

Despite the premature termination of the X-38 V-201 project, the flap actuator development team learned a number of valuable lessons.

Concerns regarding the feasibility of constructing and installing the arc with sufficient accuracy to permit smooth operation of the actuator beset the team from the beginning of the redesign, yet it ultimately functioned perfectly. However, with little experience to serve as a guide, all of the parts were designed and fabricated to the tightest possible tolerances and weeks of effort were spent to align the flaps and actuators as close to the theoretical locations as possible (including offsets for known errors). Also, many elements of the system were devised to reduce free play between the components as much as possible since it was unknown how much would be acceptable. Such extreme efforts added substantial expense and in the future it would be instructive to explore the tolerance for errors of this actuator configuration. In particular, the fabrication of the large Inconel parts became far more difficult than anticipated and only succeeded after multiple failed attempts, the details of which exceed the scope of this paper.

Altogether, creating the actuator system utilized at least five different computer aided design (CAD) systems. Three separate systems were used for the design: one each by the flap supplier, the drive unit supplier, and the vehicle integration team. The integration team alone also used two other systems, one for stress analysis and one for thermal analysis. Keeping all of the different models synchronized was a significant problem that was seriously compounded by the great difficulties of translating between the different parametric, solid model CAD systems. The difficulties of translating between human languages, different systems of measurement, and coordinating across worldwide distances were trivial compared to the problems and the vast amounts of lost time translating between the CAD systems. In retrospect, more effort should have been made to standardize on a single system as the practical state of the art in CAD translation leaves much to be desired.

Somewhat outside the scope of this discussion of the actuator, but worth noting, was the difficulty of installing and aligning the flap system. The hinge design had far too many degrees of freedom between all of the parts and required an inordinate amount of time to integrate on the vehicle. Future projects would do well to reduce the number of degrees of freedom during installation and design into the hardware tooling that allows for optimal mechanical positioning adjustment. Elements as delicate and expensive as hot control surfaces should not be left to hand-clamping systems for mechanical adjustments and installations.

Although incomplete, progress to date on the seal system has identified a dearth of suitable technology for flexible, high temperature and oxidation resistant materials that are sufficiently robust to form elements of a mechanical system intended to operate in the hypersonic environment. Several intriguing concepts were identified, but their development would have exceeded the scope of the available time and resources of this project.

Insufficient understanding of the implications of the extreme operating environment on the actuator design and the absence of adequate detail of the actuator system early enough in the vehicle conceptual design process lead to an unworkable actuator design that was corrected only through a complete redesign of the system late in the project schedule. What would have been a very challenging design under any circumstances became all the more difficult due to the necessity of accommodating the advanced state of maturity in the rest of the vehicle design and operating under a compressed development schedule. In retrospect, the actuator conceptual design should have received higher priority during the overall vehicle layout and particularly in concert with the flap and aft fuselage configuration. As it turned out, only the adoption of an unusual, complex, and “challenging to implement” design and extraordinary effort by the team members along with a great deal of good fortune determined the difference between failure or success of the flap actuator system. Preventing the actuator redesign from becoming a pacing item for the project schedule with such a delayed start required the expenditure of substantially more resources than had been expected for this system.

It is easy to criticize the absence of emphasis on this system in the early project phases in hindsight, but the reality is that at the time, the team lacked the experience necessary to realize the challenges ahead in this system. During the early stages of the project, there was very little recent experience developing re-entry vehicles anywhere in the industry, let alone small re-entry vehicles with active control surfaces. Since the time of the project, the authors have been involved in and observed a number of other, similar projects in the early phases of development and many of them seem to also lack a sufficient early emphasis on understanding the issues of the control surface actuator systems. It must be understood that in a small re-entry vehicle with control surfaces mounted on thin airfoil sections, the actuators of those surfaces will be exposed to environmental conditions unlike anything found in lower speed aircraft and they can not be isolated from those conditions as effectively as would be possible in larger hypersonic vehicles. The actuator development risks simply cannot be estimated by analogy to similar sized airplanes or to the space shuttle, despite the natural tendency to do so.

Acknowledgements

The authors wish to acknowledge the team members that contributed to the system described in this paper: NASA Johnson Space Center (JSC) Engineering Directorate (System Integration, Actuator Mount and Linkage, Thermal Seals), Lockheed Martin Space Operations via NASA/JSC Science, Engineering, Analysis, & Test contract (System Integration, Actuator Mount and Linkage, Thermal Seals), Moog Inc., Motion Systems Division (Actuator Drive Unit), Oceaneering Thermal Systems, (Thermal Seal), MAN Technologie AG Space Technology Division (Body Flap).

References

1. Hagen, Jeffrey D. “Actuator Installation, V-201 Body Flap” SGK51355379 – NASA/JSC Engineering Drawing Control Center.
2. Eckert, A. & Steinacher, A. “X-38 Body Flap Assembly and Integration Procedure” TET-MAN-11-AIP-1013 – MAN Technologie.
3. Layer, C. “E.M. Actuator Assembly Model 176E111” C52002 – Moog Inc., Motion Systems Division.
4. Curry, Donald M. & Handrick, Karin E. “Dynamic and Static High Temperature Resistant Ceramic Seals for X38 Re-entry Vehicle” 53rd International Astronautical Congress – IAC-02-1.3.01.
5. Curry, Donald M. et al. “Further Investigations of Control Surface Seals for the X-38 Re-entry Vehicle” AIAA-2001-3628 – American Institute of Aeronautics and Astronautics.